

Conceptual Design of a Hypervelocity Asteroid Intercept Vehicle (HAIV) and its Flight Validation Mission

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An intercept mission with nuclear explosives is the only practical mitigation option against the most probable impact threat of near-Earth objects (NEOs) with a short warning time (e.g., much less than 10 years). The existing penetrated subsurface nuclear explosion technology limits the intercept velocity to less than approximately 300 m/s. Consequently, an innovative concept of blending a hypervelocity kinetic impactor with a subsurface nuclear explosion has been developed for optimal penetration, fragmentation, and dispersion of the target NEO. A proposed HAIV (hypervelocity asteroid intercept vehicle) consists of a kinetic-impact leader spacecraft and a follower spacecraft carrying nuclear explosives. This paper describes the conceptual development and design of a baseline HAIV system and its flight validation mission architecture for three mission cost classifications (e.g., \$500M, \$1B, and \$1.5B).

I. Introduction

A growing interest currently exists for developing a national plan to protect the Earth from the future possibility of a catastrophic impact by a hazardous asteroid or comet. In a letter on NEOs from the White House Office of Science and Technology Policy (OSTP) to the U.S. Senate and Congress in 2010, the White House OSTP strongly recommends that NASA take the lead in conducting research activities for the development of NEO detection, characterization, and deflection technologies. Furthermore, President Obama's new National Space Policy specifically directs NASA to "pursue capabilities, in cooperation with other departments, agencies, and commercial partners, to detect, track, catalog, and characterize NEOs to reduce the risk of harm to humans from an unexpected impact on our planet." The Planetary Defense Task Force of the NASA Advisory Council also recommended that the NASA Office of the Chief Technologist (OCT) begin efforts to investigate asteroid deflection techniques. Because of such growing national interests, the NEO threat detection and mitigation problem has been identified recently as one of NASA's Space Technology Grand Challenges.

The Asteroid Deflection Research Center (ADRC) at Iowa State University has been developing strategies and technologies for deflection or disruption of hazardous NEOs. As the first university research center in the United States dedicated to such a complex engineering problem, the ADRC was founded in 2008 to address the engineering challenges and technology development critical to NEO impact threat mitigation. For research projects funded by NASA's Iowa Space Grant Consortium and the NIAC (NASA Innovative Advanced Concepts) program of the NASA OCT, the ADRC has been developing space technologies for mitigating the NEO impact threats.¹⁻⁶

Although various NEO deflection technologies, including nuclear explosions, kinetic-energy impactors (KEIs), and slow-pull gravity tractors (GTs), have been proposed during the past two decades, there is no consensus on how to reliably deflect or disrupt hazardous NEOs in a timely manner. Furthermore, due to various uncertainties in asteroid detection and tracking, warning time of an asteroid impact with the Earth can be very short. All of the non-nuclear techniques, including hypervelocity KEIs

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and slow-pull GTs, require mission lead times much larger than 10 years, even for a relatively small NEO. However, for the most probable mission scenarios with a warning time much less than 10 years, the use of higher-energy nuclear explosive devices (NEDs) in space will become inevitable.¹ Direct intercept missions with a short warning time will result in the arrival velocities of 10 - 30 km/s with respect to target asteroids. A rendezvous mission with target asteroids, requiring an extremely large arrival ΔV of 10 - 30 km/s, is totally impractical.

Although a less destructive, standoff nuclear explosion can be employed for direct intercept missions, the momentum/energy transfer created by a shallow subsurface nuclear explosion is roughly 100 times larger than that of a standoff nuclear explosion. However, the existing nuclear subsurface penetrator technology limits the impact velocity to less than about 300 m/s because higher impact velocities destroy the detonation fuzing devices prematurely, although an impact speed limit of 1.5 km/s has been cited for nuclear Earth-penetrator weapons (EPWs).⁹ Also, a precision standoff explosion at an optimal height of burst near an irregularly shaped, smaller NEO, with intercept velocities as high as 30 km/s, is not a trivial task.

Consequently, a proposed *hypervelocity asteroid intercept vehicle (HAIV)* serves as a possible solution which will enable a last-minute, nuclear disruption mission with intercept velocities as high as 30 km/s. This paper presents the details of a baseline two-body HAIV configuration, thermal shielding of a follower spacecraft, impactor targeting sensors and optical instruments, thruster configurations and terminal guidance phase operations, and other secondary optional HAIV configurations. Hydrodynamic simulations are used to assess the mission feasibility and to provide thermal and structural design constraints for the follower spacecraft carrying NEDs. The preliminary development and design of a

Planetary Defense Technology Demonstration Mission

Primary-1. PDT Mission Using "Dummy" Payload

Optional-2. Orbiter S/C for Observation

Optional-3. Sample Return Mission

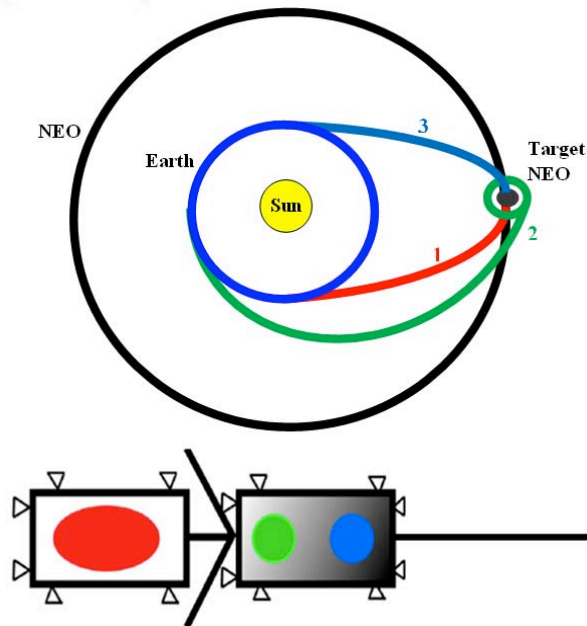


Figure 1. Conceptual illustration of a planetary defense technology (PDT) demonstration mission for validating the HAIV technology.

baseline HAIV exploits system architectures, technologies, and concepts from NASA's Deep Impact, ESA's Don Quijote (canceled), NASA's NuSTAR, and ADRC's Interplanetary Ballistic Missile System (IPBM).⁴ It is envisioned that eventually in the near future, planetary defense technology (PDT) demonstration missions will be considered seriously by an international space community in order to validate the overall effectiveness and robustness of various nuclear options and the associated space technologies. The PDT flight demonstration mission concepts⁵ described in this paper fall into three budget classifications: \$500M, \$1B, and \$1.5B. The ADRC's mission design software tools have been utilized to conduct a search of optimal asteroid targets for a PDT demonstration mission, which would validate asteroid disruption capabilities. For flight validation missions, suitable targets need to be identified. NASA's Near Earth Object Program database was used to construct a list of near-Earth asteroids.⁶ Although comets are also at risk of impacting the Earth, they add unnecessary complexity to the spacecraft design as it must be shielded from the small, hypervelocity dust grains that form the coma. In addition, previous missions such as Deep Impact and Stardust have already flight validated the necessary shielding and targeting capabilities for comets. As such, the asteroid targets identified in this study will allow a demonstration mission to focus on validating deflection/disruption technologies, which should prove equally effective against comets should the need arise.

Figure 1 illustrates the primary and optional mission architectures for a PDT validation mission to a target NEO. An ideal primary objective for a flight demonstration mission is to test and validate the HAIV using a real NED to be employed as in an actual Earth-threatening situation. However, political differences may interfere with this idea in which case a small explosive device or a representative "dummy" payload could be used as an alternative payload option to verify and validate the PD technologies. Other optional missions can be accomplished such as sending an orbiter spacecraft to observe the effectiveness of the disruption mission or collect NEO composition samples and return it to Earth for analysis. Although there are optional mission objectives that can be conducted with this mission, it is imperative that the primary objective of validating the HAIV technology is to remain paramount.

II. Nuclear Disruption Mission Requirements

A practical design solution is required for the delivery of a robust and effective subsurface explosion, using available technology, through a direct intercept trajectory, to mitigate the most probable impact threat of NEOs with a short warning time. Since the warning time is short, a rendezvous mission becomes impractical due to the resulting NEO intercept velocity exceeding 10 km/s. NEDs constitute a mature technology, with well-characterized outputs and are the most mass-efficient means for storing energy with today's technology.^{1,2,7} Nuclear disruption strategies to be employed in a last-minute, direct intercept mission include standoff explosions, surface contact bursts, and subsurface explosions. For each nuclear technique, accurate timing of the nuclear explosive detonation will be required during the terminal-phase guidance of hypervelocity intercept missions. Furthermore, the conventional penetrating NEDs require the impact speed to be less than 300 m/s.

A nuclear disruption mission employs nuclear explosives in three different ways. A nuclear standoff explosion at a predetermined height is often considered as the preferred approach among the nuclear options. A second nuclear option exploits a contact burst on the NEO's surface. The most efficient nuclear option involves a subsurface explosion. The subsurface explosion, even with a shallow burial (< 5 m), delivers large energy that can totally fragment the target NEO. The NED payloads can be categorized into three classes as: i) a 300-kg NED with a yield of about 300 kt, ii) a 1,000-kg NED with a yield of about 1 Mt, and iii) a 1,500-kg NED with a yield of about 2 Mt.

The nuclear standoff burst technique can be used for long-term warning times. The nuclear standoff scenario utilizes the short burst of energy from a nuclear explosive to heat a thin layer of an NEO's surface. As this layer accelerates away from the NEO, its main body recoils in the opposite direction, thus altering its trajectory.⁷ The area of the NEO's surface that is heated by a standoff nuclear explosion depends on the distance between the asteroid and the point of detonation. Also, the depth of penetration depends on the distance between the surface and the detonation point. Thus, detonation close to the surface heats only a small area close to the explosion. At larger distances, the explosion spreads its energy

over a larger area of the asteroid, increasing the angle of effect. As a result, the penetration depth decreases. One advantage of this technique is that it does not require stringent spacecraft maneuvers as might be required for a surface or subsurface explosion.

Numerous studies have been conducted in the past to understand the effect of a standoff nuclear explosion and its ΔV capability. One particular study simulated the effect of a nuclear standoff detonation on homogeneous 1 km-diameter NEOs with densities between 1.91 and 1.31 g/cm³. Approximately 40 seconds after the standoff burst, at 150 m above the NEO's surface, the NEO's speed change ranged from 2.2 to 2.4 cm/s. It was estimated that 97.5% of each NEO from all simulations remained intact, while about 2.5% of its mass was ejected at greater than escape speed by the rebound to the shock wave that passes through the body in reaction to the ejection of heated material.^{7,10} The NEO was held by gravity only and had no tensile strength model. The study concludes that deeper neutron penetration is not dependent on NEO composition.

Another nuclear technique involves the subsurface nuclear explosions. The nuclear subsurface method even with a shallow burial (< 5 m) delivers large energy, potentially disrupting the NEO completely.⁷ One advantage of the subsurface technique is the increased exposed surface area to the NED. A concave surface area absorbs more of the nuclear energy thus allowing a more efficient disruption of the NEO. The major advantage of a nuclear subsurface explosion over a surface or aboveground nuclear explosion is the effectiveness with which energy is transmitted into the NEO. The effectiveness of Earth-penetrating weapons can be used to illustrate the nuclear subsurface method on an NEO.

Nuclear Earth-penetrator weapons (EPWs)⁹ with a depth of penetration of approximately 3 meters captures most of the advantage associated with the coupling of ground shock. According to Fig. 2, the yield required of a nuclear weapon to destroy a deeply buried target is reduced by a factor of 15 to 25 by ground-shock coupling enhancement. The EPW is designed to detonate below the ground surface after surviving the extremely high shock and structural loading environments that result during high-speed impact and penetration. However, its impact speed is limited to approximately 300 m/s. While additional depth of penetration increases ground-shock coupling, it also increases the uncertainty of EPW survival. The ground-shock coupling factor makes the subsurface technique much more efficient than the other nuclear techniques. The ground-shock coupled energy of an EPW approaches 50% with increasing depth of burst (DOB), and is fully coupled at a scaled DOB of about $(2.3)DOB/Y^{1/3}$, where DOB is the depth of burst in meters and Y is the yield in kilotons. Scaled DOB, defined as $DOB/Y^{1/3}$, is a normalization of the actual depth (or height) of a burst based on weapon yield to that for a 1-kt weapon. Thus, the scaled DOB

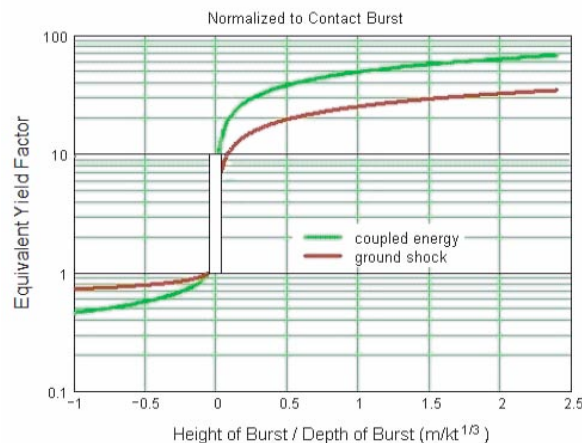


Figure 2. Equivalent yield factors for total coupled energy and ground-shock coupled energy normalized to a contact burst.⁹

and actual DOB are the same for a 1-kt EPW. For example, a 1-kt weapon buried 3 meters has a 3 scaled DOB, whereas a 300-kt weapon buried at the same depth of 3 meters couples its energy to the ground as if it were a 1-kt weapon buried at an actual depth of about 0.45 meter; that is, $3/300^{1/3} = 3/6.67 = 0.45$. For a generic 300-kt EPW at 3-m depth of burst (scaled DOB = $3/(300)^{1/3} = 0.45$), the ground shock-coupling factor is about 20, which is equivalent to a contact burst of about 6-Mt EPW.⁹

Fundamental principles of Keplerian orbital dynamics can be effectively used for examining the effects of the nuclear subsurface explosion under various physical modeling uncertainties.⁸ In Ref. 8, the study considers such a nuclear subsurface explosion with a shallow burial of approximately 5 m for different models of NEOs. In the simulations, the energy source (with 900 kt or 300 kt) region expands creating a shock that propagates through the body resulting in fragmentation and dispersal. While the material representations used have been tested in a terrestrial environment, there are low-density objects, like Mathilde, where crater evidence suggests a very porous regolith with efficient shock dissipation.^{8,10} Shock propagation may be less efficient in porous material, generally reducing the net impulse from a given amount of energy coupled into the surface.

A common concern for such a powerful nuclear option is the risk that the nuclear disruption mission could result in fragmentation of the NEO, which could substantially increase the damage upon its Earth impact. For short warning time missions, the impact mass can be reduced to 0.2% of the initial mass of the NEO, if the intercept disruption occurs nearly perpendicular to the NEO's orbital flight direction.¹¹ Such a sideways push is known to be optimal when a target NEO is in the terminal orbit before the impact.^{8,10,11} The mass of Earth-impacting fragments can be further reduced by increasing the intercept-to-impact time or by increasing the nuclear yield. However, disruption/fragmentation is a feasible strategy only if it can be shown that the hazard is truly diminished. Additional research has been recommended, particularly including experiments on real comets and asteroids, to prove that nuclear disruption can be a valid method.

III. Design and Analysis of a Baseline HAIV Architecture

Current technology and spacecraft concepts from previous NEO missions provide a starting point for the preliminary design of a baseline HAIV. After the success of previous flyby missions to comets such as Stardust, NASA developed the Deep Impact mission to achieve a hypervelocity intercept of a comet, retrieve information on the impact event, and obtain several high resolution images of the comet's interior. The Deep Impact mission employed two spacecraft to study the characteristics of the comet Tempel 1. ESA's Don Quijote mission concept also required two spacecraft to study the effects of a hypervelocity kinetic impactor hitting an asteroid. Unfortunately, the mission was canceled due to mission uncertainty and cost. ADRC's IPBM (interplanetary ballistic missile) concept takes a versatile payload option approach to be used for a variety of deflection/disruption missions. NASA's NuSTAR mission concept is a two-body spacecraft separated by a 10-meter deployable mast. Although its mission is not related to planetary defense, the dynamics and control challenges of a two-body spacecraft have been verified through this mission. These various system architectures are exploited for the preliminary design of a baseline HAIV.

A. A Baseline HAIV Mission Architecture

A baseline system concept has been developed to accommodate the technically challenging aspects of the penetrating subsurface nuclear explosion approach. A baseline HAIV consists of a leader spacecraft (kinetic impactor) and a follower spacecraft carrying an NED for the most effective disruption of a target NEO. The leader spacecraft impacts first and creates a shallow crater in the NEO. Then, the follower spacecraft enters the crater and detonates the NED.^{1,3} The HAIV configurations are shown in Fig. 3 and also the baseline HAIV mission concept is illustrated in Fig. 4.

The primary HAIV carrying a 1000-kg NED payload is delivered by a Delta IV M+ class launch vehicle. The launch vehicle places the HAIV into a direct transfer orbit towards the target NEO. During the transfer phase, the HAIV remains as a single spacecraft by way of the leader staying attached to the follower spacecraft. The HAIV uses a bi-propellant system with a 4,400-N gimballed engine to execute

trajectory correction maneuvers (TCMs). The single spacecraft can be placed in a dormant state, periodically relaying status updates while in transit until the terminal guidance phase.

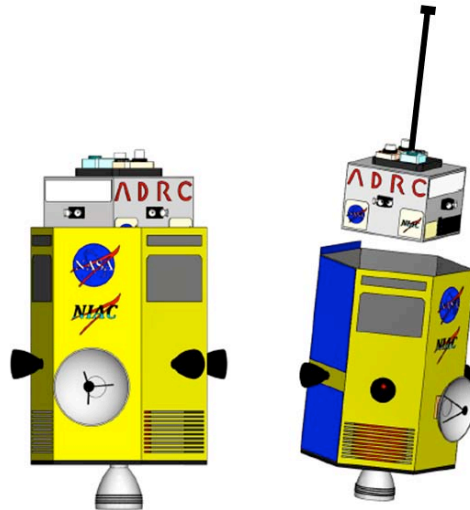


Figure 3. Primary HAIV configuration during terminal intercept phase.

Hypervelocity Asteroid Interceptor Vehicle Terminal Phase Operations

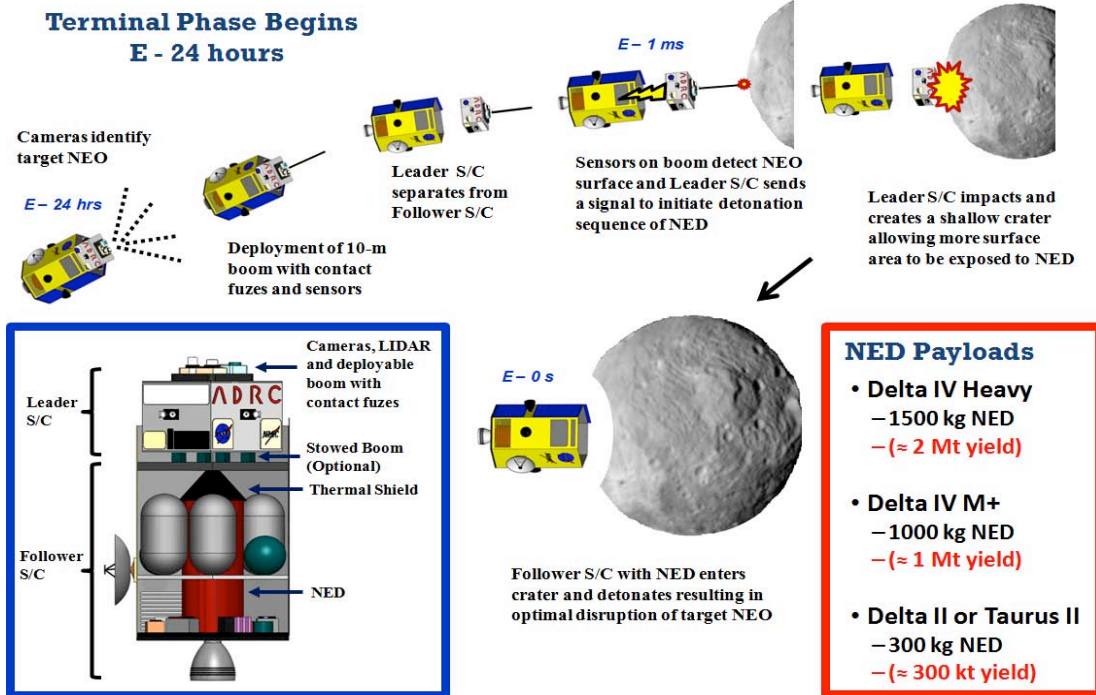


Figure 4. Conceptual illustration of the baseline HAIV mission architecture.

The terminal-phase guidance starts 24 hours before the impact event. Instruments located on the spacecraft detect the target NEO and the subsystems on-board the HAIV become active. Measurements are continued through optical cameras and laser radars located on the leader spacecraft and an intercept location is identified on the asteroid body. The high resolution optical cameras, provides successive images of the NEO to each flight computer where guidance and navigation algorithms are used to guide the impactor and the follower to the intercept location. The computer then uses these calculations to compute the necessary adjustments and instructs the divert and attitude control system (DACs) to execute TCMs. A 10-m boom equipped with contact fuzes and sensors is deployed from the leader spacecraft. Separation occurs between the leader spacecraft and the follower spacecraft and communication is established between the two spacecraft. As the distance between the follower spacecraft and NEO becomes smaller, the triggering system turns on, readying the fuzing mechanisms of the NED payload.

The nuclear fuzing mechanism is initiated by the contact fuzes located at the front of the 10-m deployable boom. Once the boom confirms contact the NEO's surface, it closes the electrical circuit of the contact fuze and the leader spacecraft sends a signal to the follower spacecraft to initiate the detonation sequence. A shallow crater is then created as the leader spacecraft impacts the NEO. Hot ejecta and debris particles result as the leader spacecraft is vaporized at hypervelocity impact. The follower spacecraft is equipped with a thermally resistant, hypervelocity debris shield that protects the NED and triggering system. The shield deforms and melts as it passes through the hot plasma ejecta and the NED detonates. It is assumed that nuclear detonation sequencing requires approximately 1 millisecond of lead time. With a relative speed of 10 km/s, a 10 meter boom connected to the leader spacecraft is assumed to ensure the accuracy of the detonation timing. This timing delay is the most critical part of the disruption mission.

Partitioning options between the leader and follower spacecraft to ensure the follower spacecraft enters the crater opening safely are discussed here. The primary option uses no connection between the two spacecraft. This configuration depends on the instruments, communication, flight computer, and guidance and tracking algorithms to carry out the terminal-phase guidance and impact.

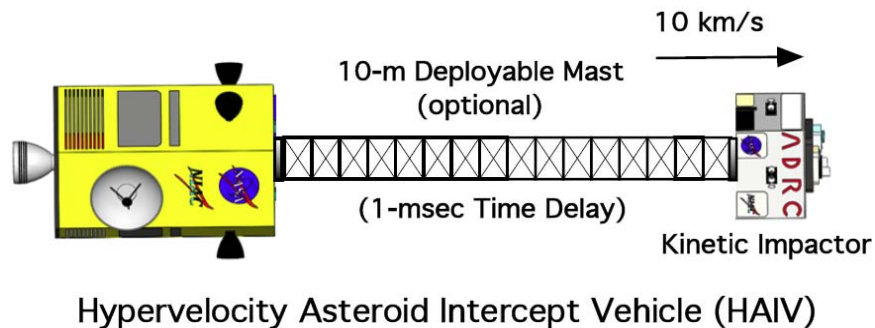


Figure 5. Secondary HAIV configuration employing a deployable mast.

Another option includes the use of a rigid connection between the two bodies through a deployable mast. Figure 5 shows an optional HAIV configuration with a deployable mast. As the mast is deployed and separation distance increases, the center of mass moves from the center towards the front of the follower spacecraft. This new configuration is still treated as a single body but achieves a two-body arrangement. Divert thrusters are pre-positioned at the expected new center of mass location to control the new system as a single body. These large divert thrusters can be gimballed to achieve the desirable thrust

directions. This configuration reduces mission complexity and operations, but is limited to the length of the boom. This is proposed as an optional configuration of the primary HAIV, and it needs further study.

B. State-Of-The-Art Nuclear Fuzing Mechanisms

The NED triggering system is the most vital element of the HAIV. In general, a standard fuzing mechanism ensures optimum NED effectiveness by detecting that the desired conditions for NED detonation have been met and to provide an appropriate command signal to the firing set to initiate nuclear detonation. Terrestrial fuzing generally involves devices to detect the location of the NED with respect to the target, signal processing and logic, and an output circuit to initiate firing. Without the proper selection of a reliable triggering or fuzing mechanism, there is a high risk that the mission can be unsuccessful. Current terrestrial triggering systems such as salvage fuzes, timing, contact, and radar (proximity) fuzes are employed to detonate the NEDs.³

The salvage fuze acts as a contingency fuze which is employed as a failsafe detonation. The fuze “salvages” the bomb and explodes when all other fuzes fail. The salvage fuze serves as a countermeasure to a terminal defense interceptor system and initiates after a detected collision possibility. The NED then explodes as soon as an interceptor comes within a certain range of the NED. Sometimes radar and contact fuzes operate as the failsafe triggers and must function after withstanding extreme deceleration forces and delivery vehicle deformation. In an asteroid direct intercept scenario, the salvage fuze comprised of several contact and radar fuzes becomes activated, and the contact and radar fuzes provide one option for arming and detonating the NED.

Another option for triggering the NED is a timing fuze. The timing fuze operates by using time-to-go, estimated intercept distance, and the rate of the intercept distance. This information is provided to the triggering mechanism by the guidance, navigation, and control instruments and flight computer. The computer activates the timing fuze once the guidance parameters meet specific conditions. However, if the timing fuze proves to be inaccurate, the salvage fuzes (contact and radar fuzes) can restore the arming mechanism of the NED. A salvage fuze is always present to resume the arming of the NED in any such triggering problems.

Proper fuzing systems and operations need to be chosen. For a standoff burst disruption mission, radar acts as part of the primary fuzing system. For the subsurface or contact burst option, coupling of contact and radar fuzes are the primary detonation system. The selection and sequencing of these fuzing options are chosen autonomously and are not dependent on additional hardware or configurations. Contact and radar fuzes can be located on the 10-m boom deployed from the leader spacecraft. This arrangement allows sufficient time to initiate the NED detonation sequence (1 millisecond) before impact.

C. Terminal Guidance Sensors and Instruments

Optical cameras, radar altimeters, and Light Detection and Ranging (LIDAR) are used on the leader spacecraft to accurately identify and track the target NEO and initiate fuzing for the NED. The leader uses a Medium Resolution Instrument (MRI) or Wide Field of View (WFOV) Imager as used on the Deep Impact flyby spacecraft. The WFOV Imager is used to locate the target NEO at the start of the terminal-phase guidance. It is a small telescope with a diameter of 12 cm and takes images with a scale of 10 m/pixel in a spectrum of approximately 700 km. The field of view of the WFOV Imager is approximately 10 deg x 10 deg which allows for more observation of stars and serves as a better navigator for the HAIV during its coasting phase. As soon as possible after acquisition of the target NEO, the WFOV Imager passes information to the High Resolution Instrument (HRI) or Narrow Field of View (NFOV) Imager, which has a field of view of 2.3 deg x 2.3 deg. It is comprised of a 30-cm diameter telescope that delivers light to both an infrared spectrometer and a multispectral camera. The camera has the ability to image the NEO with a scale that is less than 2 m/pixel when the spacecraft is approximately 700 km away. Table 1 shows the properties of each Imager. The Imagers are located on the leading front of the impactor spacecraft. These Imagers are similar to the instruments used on the Deep Impact Mission Flyby and Impactor spacecraft.

Table 1. HAIV Imagers and sensor package properties.

Parameter	NFOV Imager	WFOV Imager
Field of View (deg)	2.3 x 2.3	9.5 x 9.5
Angular Resolution (μ rad)	10	40
Focal Plane Dimension (pixels)	1024 x 1024	1024 x 1024
Estimated Mass (kg)	15	10
Estimated Power (W)	20	10

LIDAR or laser radar measures back-scattered light from a high intensity, short duration output pulse transmitted at the target NEO. It is used in the closing minutes of the terminal phase to calculate the range to the NEO. This information is shared with the fuzing device for detonation of the NED. The LIDAR requires sufficient power to operate over a range equivalent to approximately the last minutes of the terminal phase. The device design would be similar to the ones used on the NEAR and Clementine missions. The LIDAR has a mass of 20 kg and an estimated power consumption of 50 W. Radar altimeters using radio waves are used in conjunction with LIDAR. More study on these instruments is required to meet terminal phase requirements.

D. Thermal Protection and Shield

An in-house hydrodynamics code, which is being developed to accurately study the effects of nuclear disruption missions, is used to estimate the thermal and structural limits experienced by the two-body HAIV. The hydrodynamic code helps to establish a shield design and its configuration on the follower spacecraft. Several different geometries include a flat cylindrical plate, conical shape, spherical cap, and an EPW ogive nose cone. The hydrodynamics code developed by the ADRC is based on a meshless model used previously for asteroid impactor analysis.^{2,10} The initial impact is generated by a spherical shell matching the mass of the leading body, resulting in a field of hot gas and ejecta through which the payload must survive. Figure 6 illustrates this process through snapshots taken from the simulation. It is assumed that most NED designs will experience melting or exceed the maximum allowable structural load in this region. Therefore, a shield design is desirable to mitigate the effects of incident vaporized rock from the leader spacecraft, substantially protect the payload from micrometeorites ejected from the impact, and allow for the maximum depth of burst. Figure 7 shows the peak specific energy of a 0.7 m diameter cylindrical aluminium payload shield as a function of depth for three nominal thicknesses. The horizontal line represents failure of the system to adequately protect the payload, resulting in failsafe detonation.

As observed in Fig. 7, a minimal thickness for this shield is about 10 cm. Above this value, little additional penetration is observed, given the thermal gradient in this region. A complicating factor is the acceleration of the payload. The 10 km/s initial relative speed greatly exceeds the speed of sound in the shield structure, resulting in the equivalent of a standing shock along the shield. Ahead of this shock, the payload measures only minimal interruption. Some initial acceleration due to ejecta impacts and interaction with the gas environment is measurable, but shortly thereafter the maximum structural load is reached. Thickness of the shield has almost no effect on the maximum depth reached before structural failure, making overly thick shields a hindrance rather than a benefit.³

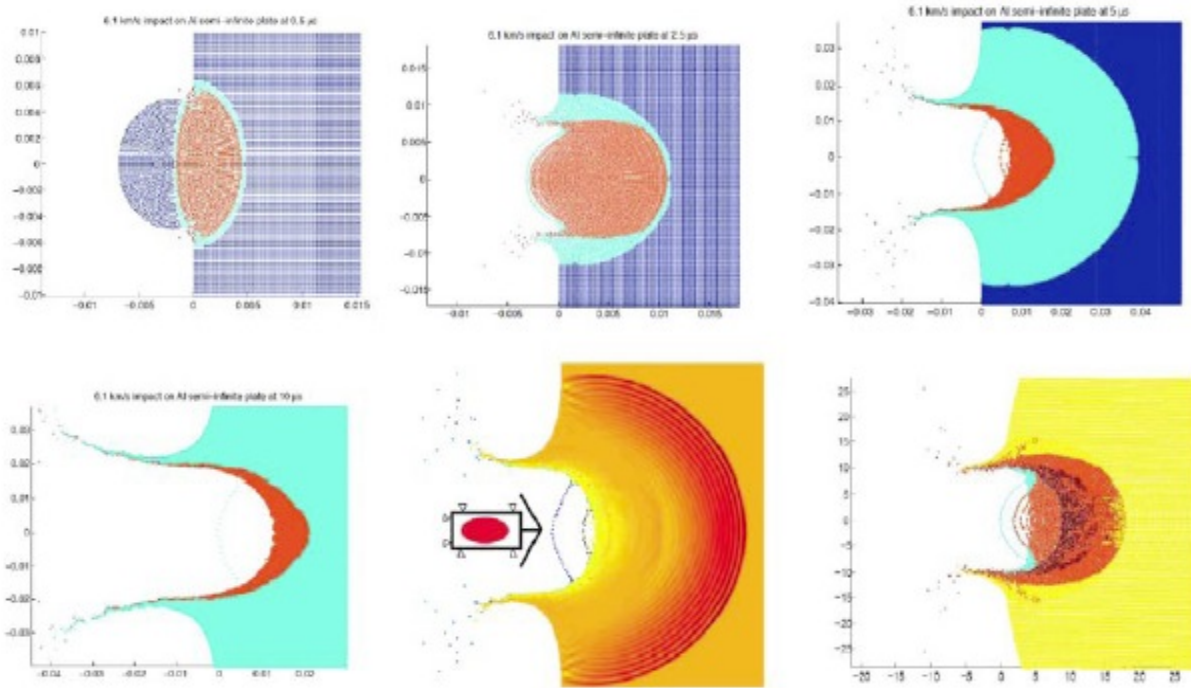


Figure 6. Preliminary simulation study results for the hypervelocity penetrated subsurface nuclear explosion option.

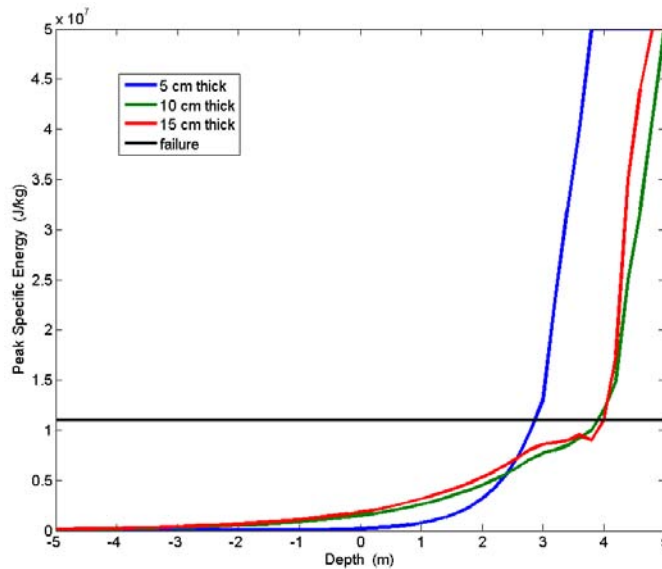


Figure 7. Peak payload specific energy for flat shield design.³

Table 2 shows the results for minimum thicknesses and masses (of aluminum) of the flat, conical, spherical, and ogive nose cone discussed previously. These thicknesses are chosen to allow survival of the payload until the shield experiences structural failure. Further study found these thicknesses to depend very little on the material chosen, other than the mass of the resulting system, as the shape of the shield and the leader spacecraft tend to govern the achievable depth. Also listed in Table 2 is the maximum achieved depth of burst (DOB). Reduced performance can be achieved by using thinner shields, and lowering the required DOB would result in benefits for timing the detonation of the payload.

Based on this initial study, few conclusions can be drawn for the design of the payload thermal shield. First, the primary variables in achievable DOB are the shape, mass, and timing of the kinetic-impact leader spacecraft. Additional analysis must be done to optimize this portion of the mission. Second, given a particular environment, a discontinuous shock to the payload presents challenges in determining how far to allow penetration before detonation. The payload cannot survive a direct impact at this speed, so it must be triggered using a combination of sensor and optical data at an appropriate data rate. Third, geometry of the shield seems to present a greater influence on DOB than any other variable. Adding thickness to the thermal shield in excess of the minimums presented do not result in further penetration, since both shields experience high structural loads at the maximum DOB. Finally, these results appear to be independent of the materials tested, as the limiting factor is the acceptable structural loads on the payload. However, significant mass can be saved by utilizing lighter alloys or materials for the thermal shield.

Table 2. DOB based on thickness parameter and shield geometry.³

Shield	Thickness (cm)	Mass (kg)	DOB (m)
Flat Cylinder	9.4	97.7	3.8
Conical	10.1	105	4.1
Spherical	8.8	76.8	5.3
Ogive	10.5	116.1	4.6

E. Optional HAIV Configuration Employing a Deployable Mast

The leader and the follower spacecraft can be separated and connected by a deployable mast to ensure that the NED payload follows the leader spacecraft safely and reliably. The mast/boom must be sufficiently rigid to avoid oscillatory motion of the two bodies. A deployable mechanism is preferable compared to a fixed structure due to volume constraints in the launch vehicle fairing. The connection mechanism can be divided into four categories, hinged, telescoping, an articulated mast system, and carbon fiber reinforced plastics (CFRP).

A hinged deployable boom consists of a hinged truss structure that is collapsible in storage and when deployed, locks into place and is held firm. ATK, the manufacturer of such trusses, reports 12.4 m and 6.2 m length trusses both with bending stiffness of $1.5 \times 10^6 \text{ N}\cdot\text{m}^2$, although mechanical properties are dependent on component materials. Depending on the materials of the components for the system, the mass cost of such a system could be high. Most such trusses are planned to be retractable which adds a level of complexity that is unnecessary for the HAIV application. ATK has manufactured many systems that have been tailored to specific mission requirements, and provides a favorable flight history. Another option that can also be classified as a hinged deployable boom is the folding hinged boom. ESA has been developing such systems and are much like the hinged truss. This particular option does not have the flight history as reported by ATK but mostly because it is highly tailorable to each application, making comparison difficult. The mass and mechanical properties of the hinged booms are strongly tied to the

material selected. Composite materials may be lighter but more expensive and metals would be heavier but easier to manufacture.

ATK also provides a telescoping system which is also meant to be retractable. ATK reports a 5.5 m (deployed) boom with a bending strength of 72,000 N that weighs 20 kg. Unfortunately, when the boom is deployed, the diameter of the next telescoping section is reduced in order to be efficiently stored.³ This option has a high mass cost and is used primarily for larger spacecraft applications.

The articulated mast system is designed and manufactured by ATK and is used for deploying critical spacecraft payloads. It can be tailored for specific mission requirements and has efficient stowage volume. Its deployment capability has a high push force with or without active controls. It has lengths up to 60 m with a bending load capacity of 8,100 N·m and a bending stiffness of 5.76×10^8 N·m². The articulated mast system has had successful deployment on multiple ISS/STS missions and is being used on NASA's NuSTAR mission.

Solar sails have previously employed deployable booms made of carbon fiber reinforced plastics (CFRP). The booms (which can be made up to 20 m) have a unique cross section. When the cross section is flattened, by pulling horizontally, the material can be coiled. The CFRP wraps in an "S" coil which occupies more volume than anticipated. The dimensions of the structure are designed to fit specific loads. There are also similar models that are pressurized (inflatable) to increase strength and stiffness. More research is needed to choose the boom that meets the requirements of connecting the two bodies of the HAIV.

F. Mass Budget Summary

The proposed baseline HAIV, as shown in Fig. 3, takes the form of a box-shaped impactor spacecraft equipped with thrusters and targeting instruments. It connects to a hexagon-shaped follower spacecraft equipped with 4 divert thrusters, a high-gain antenna, a thermally resistant shield, and an NED. The HAIV has a total length of approximately 6.7 m and a circular base of 4 m. The follower spacecraft incorporates a shelf that holds the leader spacecraft and the optional stowed booms. The leader spacecraft is also equipped with a boom to be deployed before impact. Sensors and contact fuzes are located on top of this boom which detects the surface of the NEO giving accurate detonation timing delay. In addition, the leader spacecraft separates from the follower spacecraft by pyrotechnic spring attachments.

The HAIV is configured by using unscaled dimensions of commercial off-the-shelf components and materials such as ATK's fuel tanks, bi-propellant engine, optical instruments, etc. These dimensions and mass properties accurately reflect a preliminary configuration of an innovative HAIV.

Table 3 shows the mass breakdown of a baseline primary HAIV carrying a 1000-kg NED payload. The leader spacecraft has a wet mass of 315 kg and the ability to provide a total ΔV of 270 m/s which is similar to what the impactor spacecraft used in the Deep Impact mission during its terminal phase. The follower spacecraft has a dry mass of 1,170 kg carrying an NED payload of 1,000 kg. Depending on the material selected, the thermal shield and the optional deployable booms are estimated at an average of 135 kg and 55 kg, respectively, which correspond to previous space missions (Deep Impact Mission and Space Shuttle). The follower spacecraft also has the propellant necessary to execute trajectory correction maneuvers (ΔV of 550 m/s) during the transfer orbit and adjustment maneuvers (ΔV of 270 m/s) during the terminal phase. The mass of the HAIV upon arrival at the target NEO is estimated at 2,710 kg.

A mass margin of 30% is used to account for uncertainties, thus making the total wet mass at launch approximately 4,238 kg. Without using an upper stage or orbital transfer vehicle, the Delta IV M+ has the capability to deliver the HAIV in a direct C3 trajectory towards a target NEO. The propellant system on the HAIV uses a bi-propellant feed system of dinitrogen tetroxide (N₂O₄) coupled with MMH attitude thrusters. The N₂O₄ propellant system has a restartable engine capable of producing 4,400 N of thrust at a specific impulse of 326 seconds, making it favorable for executing TCMs. The MMH attitude thrusters are used for attitude adjustments and terminal adjustment maneuvers. The leader and follower spacecraft are equipped with small MMH attitude thrusters.

Table 3. Mass breakdown of the baseline HAIV.³

Vehicle	Description	Mass (kg)
Leader	Dry Mass	285
	MMH Propellant	30
	Wet Mass	315
Follower	Dry Mass	1170
	NED Payload	1000
	Thermal Shield	135
	Deployable Boom (Optional)	55
	Total Dry Mass	2170
	N ₂ O ₄ Propellant	775
	Wet Mass	2945
Total Spacecraft	Dry Mass	2455
	Wet Mass at Launch	3260
	Mass at NEO	2710
	Mass Margin (30%)	978
	Total Mass w/ Margin	4238

Other secondary options of the HAIV exist depending on ΔV demand, mission budget, and NEO characteristics. A Delta II class launch vehicle in conjunction with an upper stage or orbital transfer vehicle can be used to launch a smaller HAIV (1,543-kg) that is capable of carrying a 300-kg NED payload. Likewise, a Delta IV Heavy launch vehicle class can deliver a scaled up version of the HAIV (4,242-kg) able to carry a 1,500-kg NED payload.^{3,5} NED payloads and fuel tanks can be interchanged easily with slight modification to the HAIV or to accommodate different launch vehicles. This design process is explained in a detailed flow chart and is used to consider all feasible mission scenarios.

G. Pre-Mission Design Process

The pre-mission design software tool is comprised of several functions and subroutines calculating payload capacity of launch vehicle classes, propellant mass and tank size for an orbital transfer vehicle (OTV), and dimensions of the payload configuration in the fairing.⁵ The software tool takes user-inputs such as the masses of the HAIV and NED payload and mission ΔV or C3 needed to reach the target NEO to calculate several different feasible solutions. A flowchart of the pre-mission design process is provided in Fig. 8. The beginning of the design algorithm takes inputs about the HAIV and NED payload, mission parameters on the target NEO, and launch vehicles to be considered for the mission. The user must also specify whether the mission is a direct C3 orbit injection or if there is a required ΔV from a 185 km altitude circular parking orbit. For the C3 orbit injection missions, all other launch vehicles and only the three-stage Delta II launch vehicles are considered due to their C3 payload capabilities. If a ΔV is to be required, an OTV is included in the design and fuel masses are computed.

With all the given inputs, the program checks specific parameters that might indicate the need of an OTV. If an OTV is not necessary for the mission, the HAIV mass and dimensions are analyzed against the fairing sizes of the launch vehicle classes to ensure it can be carried to the specified orbit. If there is a need for an OTV, the amount of ΔV needed enables the program to calculate the mass of the bi-propellant fuel. The two types of bi-propellant being considered are liquid oxygen/liquid hydrogen (LOx/LH₂) and

N₂O₄/Hydrazine. Based on the choice of fuel type, the mass and capability of the fuel can be calculated. From there, the HAIV plus OTV configuration is then checked against the launch vehicle fairing sizes to see if the entire payload can fit inside. If the HAIV or HAIV /OTV configuration does not fit within the specified class of launch vehicles' fairings, then a new class of launch vehicles will need to be specified for analysis. If the HAIV does fit within one of the launch vehicle fairings, then the algorithm has found a possible solution.

With a set of solutions obtained by the HAIV/OTV design algorithm, each solution has to be analyzed to ensure its viability. The user enters the design-loop at this point, deeming a solution as either acceptable or not, and potentially restarting the entire design process if necessary. If a viable design is found from the resulting set of solutions then it can be taken and used to design the corresponding mission to a specified target NEO.

If an OTV is deemed necessary for a particular mission, it would consist of another spacecraft bus and motor carrying propellant attached to the HAIV, to be used for the purposes of orbit injection and/or TCMs. For the required ΔV , the mass of the fuel and oxidizer will be calculated and tanks will be sized to fit the corresponding amounts of propellant. Given that the necessary change in velocity and final mass of the HAIV/OTV is known, a simple rocket equation is solved for the final mass. Therefore, the difference between the initial and final spacecraft masses is the propellant mass. Since both fuel options are

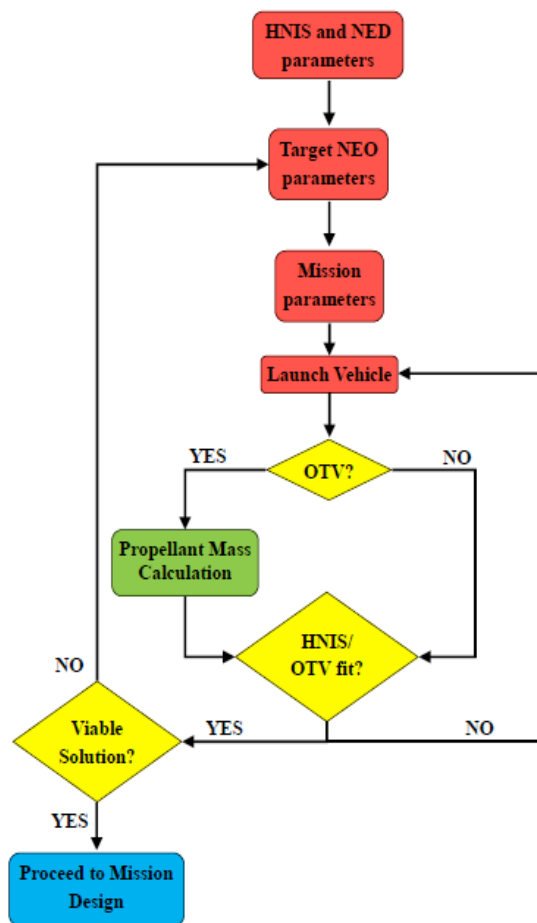


Figure 8. Flowchart illustration of the pre-mission design process.⁵

bipropellant fuel sources, the appropriate amount of fuel and oxidizer needs to be calculated. Given the propellant mass needed for the OTV, the ratio of fuel to oxidizer can be used to find how much mass and volume of each propellant.

The design and arrangement of the OTV and its tanks is important to determine if a mission is feasible. Given the amount of propellant needed, the process of finding the correct size tanks may warrant several iterations of the design-loop before a solution is finally found. The key variables in the design-loop include the desired ΔV , the mass of the propellant, the size of the tanks, and the number of tanks. If the size and number of tanks do not fit within the fairing of the specified launch vehicle, the solution set would need to be revised, either by changing the number of tanks in the OTV design, or as a last alternative, choosing a different launch vehicle to find a feasible solution.

IV. A Target Selection Process for PDT Demonstration Missions

For the purposes of this study, only asteroids in the near-Earth asteroid (NEA) groups such as Apollo, Aten, and Amor were considered. Asteroids in these groups all have perihelion distances of 1.3 AU or less, and many of them also cross the Earth's orbit at some point. Asteroids in these groups are relatively close to the Earth, and have low ΔV requirements to achieve intercept. As such, objects in these groups are the most likely candidates for an asteroid deflection/disruption demonstration mission. Apollo and Aten class asteroids are characterized by asteroids with orbits that intersect that of the Earth, which could potentially lead to lower ΔV requirements for a mission. On the other hand, this same fact means that any significant perturbation in the object's trajectory could cause it to later impact the Earth. While unlikely, a demonstration of deflection technologies could cause this to happen. The ESA also had this in mind when they selected the asteroids 2002 AT4 and 1989 ML from the Amor group for the Don Quijote mission concept.⁶ With that in mind, the Amor group shall be the focus for determining suitable candidates in this paper.

Table 4. Optimal targets with corresponding early and late launch dates, mission duration, minimum total ΔV , and diameter.

Launch Vehicle	Asteroid	Launch Date	Transfer Time (days)	Minimum ΔV (km/s)	Diameter (m)
<i>Early Launch Window</i>					
Delta II	1998 SB15	5/5/2017	159	3.34	330
	2007 FS35	2/4/2015	272	3.47	620
Atlas V	2003 GA	12/3/2015	111	3.52	300
	2009 TB3	9/22/2017	202	4.69	300
	1998 SB15	5/5/2017	159	3.33	330
Delta IV Heavy	2006 SJ198	3/17/2015	337	4.60	1,200
	2007 FS35	2/4/2015	272	3.47	620
	2004 GY	6/30/2015	365	4.35	480
	2003 QC	1/1/2015	331	4.48	400
<i>Late Launch Window</i>					
Delta II	1998 SB15	6/22/2017	104	3.33	330
	2009 TB3	1/28/2018	97	3.60	200
Atlas V	2007 FS35	10/3/2019	289	3.96	620
	1989 ML	11/17/2018	120	4.03	630
Delta IV Heavy	2006 SJ198	3/16/2018	340	4.69	1,200
	2007 FS35	10/3/2019	289	3.96	620
	2004 GY	5/23/2018	285	4.40	480
	1989 ML	11/17/2018	120	4.03	630

The Amor asteroid group is characterized by asteroids that approach the Earth, but do not actually cross its orbit. By definition the perihelion distances of these asteroids lie between 1.017 and 1.3 AU. As the entire orbit is outside that of the Earth, any disturbance in the trajectories of these asteroids is even less likely to cause them to later impact the Earth. As of 10/20/2011, there are 3,084 Amor class asteroids listed in NASA's Near Earth Object Program database.⁶ This number is first reduced by only considering asteroids that are at least 100 m in diameter. This is done by only considering objects with an absolute magnitude (H value) of 22 or lower. Assuming that the asteroids albedo falls within the presumed 0.25 to 0.05 albedo range, this H value corresponds to an object at least 110 to 240 m in diameter. Applying this minimum size limit reduces the number of asteroids to be considered to approximately 2,200.

While asteroids as small as 100 m are studied, optimal candidates will have a diameter between 300 m and 1 km. This large diameter requirement is utilized due to constraints imposed by current targeting technologies, and a necessity to assess the effectiveness of nuclear fragmentation on larger, threatening objects. Should the mission successfully disrupt a larger object, it will prove equally effective on smaller sized asteroids as well. A limit on the ΔV required for intercept is due to the limitations imposed by current launch vehicle and spacecraft capabilities. This limit also takes into account the requirement of a relative closing velocity of approximately 10 km/s. This is enforced in order to simulate a situation with a short warning time of Earth impact. The limit on total ΔV results with a similar upper limit for the Don Quijote mission selection process. This number was chosen due to the total ΔV capabilities based on maximum payload masses for each current available launch vehicle.

The majority of the data used to evaluate target asteroids was generated using a FORTRAN 90 program, which executed a grid search approach for potential launch dates spanning a period of twenty-five years (Jan. 1, 2015 to Jan. 1, 2040) in conjunction with various transfer durations up to a maximum of five years. **Error! Bookmark not defined.** Ephemeris files for 2,140 Amor asteroids were automatically downloaded via a program written specifically to access NASA's Horizons system via TELNET. Using this information, each asteroid was searched using a three day time step for both the launch date and mission length. Only direct transfer orbits were considered in this program. This search was parallelized using OpenMP to utilize each core on the workstation, and required a run time of approximately 20 hours. Although data was generated up until the year 2040, only the results for the first five year time span (Jan. 1, 2015 to Jan. 1, 2020) and a maximum mission length of one year were analyzed in greater detail.⁶

From extensive analysis, there is no benefit to looking at mission lengths beyond that of a year for most targets in terms of ΔV . While there are some possible mission designs at the very edge of the maximum mission length, they would not be any lower than the minimum ΔV pockets found between 100 and 150 days. The data was then inserted into a cost function based on the hyperbolic excess velocity and the arrival burn magnitude to ensure a 10 km/s closing velocity. Ten asteroids that minimized this function were selected as optimal targets to be studied in greater detail. Some of the asteroid diameters in Table 4 are slightly outside the desired range of 0.3-1 km. Without knowing the albedo, there is some uncertainty in either direction for these diameters. There is not too much concern for asteroids with diameters greater than 1 km, but for those which have a diameter less than 300 m, the targeting accuracy of the instruments may not be high enough to reasonably ensure an impact. As such, it will be left as a potential target to assess targeting capabilities of future spacecraft technologies, but will not be seriously considered as a target for any of the mission configurations used in this study.

The recommended targets for each launch vehicle configuration for both the earlier and later launch windows are given in Table 4. Up until this point, only the total ΔV requirement was examined to match launch vehicles with potential targets. Now the different diameters are matched with corresponding NED sizes. There is some overlap between the categorization for asteroids with diameters that could be suited for different size NEDs meaning that either configuration could be used with the target. Should the albedo of these asteroids be known with greater certainty, a more accurate diameter can be calculated, which could then reduce the amount of overlap. It should be noted that this does not have to be strictly followed. If desired, it is entirely possible to use the larger size NEDs on targets with smaller diameters than paired with in the table or vice versa.

A. Asteroid 1998 SB15

The target 1998 SB15 is the only asteroid that the Delta II launch vehicle configuration can reach. This asteroid is one of the smallest selected in this study, and will most likely not require the larger size NEDs to disrupt. Its small size will also be used to test the limits of the terminal phase guidance technology. The launch date for the minimum ΔV takes place on 5/5/2017 with a mission length of 159 days.

As can be seen in Fig. 10, the orbit of 1998 SB15 is contained entirely within the orbits of the Earth and Mars. Unlike many asteroids whose orbits go beyond that of Mars, missions to this target do not have to wait until the close approach date. The trajectory results in an impact approach angle of 19.82 degrees

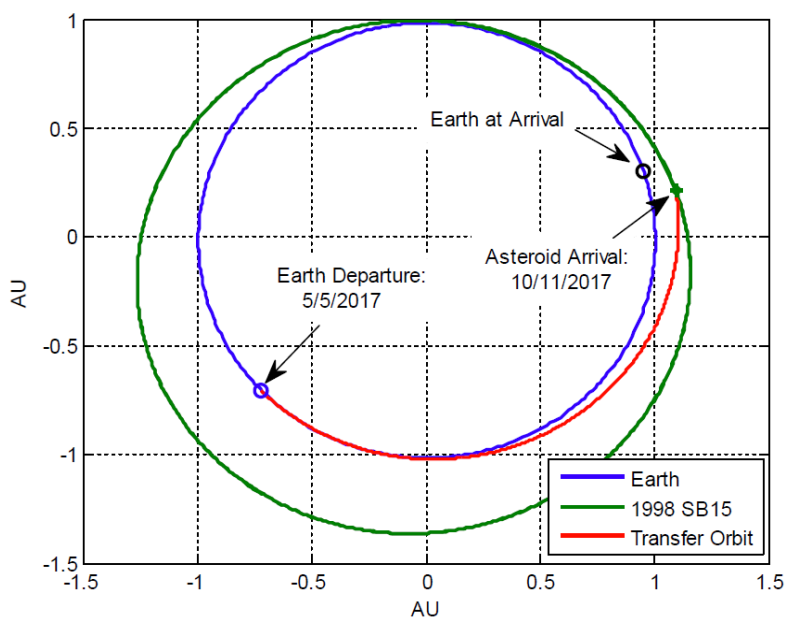


Figure 10. Mission trajectory from Earth to 1998 SB15.⁶

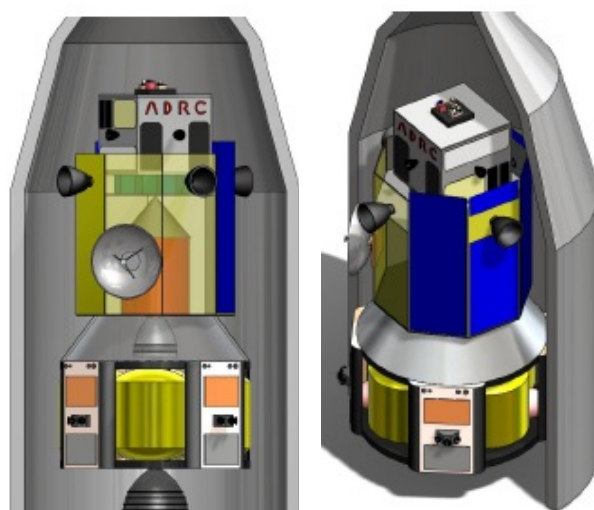


Figure 11. Baseline HAIV and OTV configuration within a Delta II fairing.⁵

and a Sun-S/C-Earth angle of 44 degrees.

The mass of the HAIV configuration to be used for the 300-kg NED mission to 1998 SB15 becomes 1,843 kg, with NED payload mass and the added mass margin included. A two-stage Delta II launch vehicle equipped with an additional OTV is utilized in order to achieve the mission requirements. The total payload mass for the Delta II launch vehicle becomes 5,868 kg. Allocating additional propellant, the OTV implements a ΔV of 3.35 km/s to inject the HAIV configuration into the appropriate direct transfer orbit towards 1998 SB15. Figure 11 shows a preliminary design of the HAIV/OTV configuration, within the fairing of a Delta II rocket.⁵

Again, these demonstration missions can be used to exercise the space technologies needed for a short warning time disruption mission. Therefore, the spacecraft can be modified to test the NED fuzing system and the hypervelocity kinetic impactor to provide experimental information without using a real NED. This information would prove to be invaluable for an actual NEO disruption mission.

B. Asteroid 2006 SJ198

As the largest asteroid in this study, 2006 SJ198 and 2011 BX10 was paired with the Delta IV Heavy launch configuration as it could carry the largest NED. With these asteroids being 4x as large as 2003 GA, the chances of failing to impact are greatly reduced. The minimum ΔV launch date for 2006 SJ198 takes place on 3/17/2015 with a mission length of 337 days. The designed trajectory for this mission initially follows closely to that of the Earth, and extends out beyond Mars as illustrated in Fig. 12. Towards the arrival date, the spacecraft approaches almost from behind the target asteroid. Upon arriving at the target asteroid, the trajectory in Fig. 12 results in an impact approach angle of 23.78 degrees and a Sun-S/C-Earth angle of 17.7 degrees.

For the most expensive and complex mission, with the largest NED and HAIV mass, it was decided that the two best asteroids to target were 2011 BX10 and 2006 SJ198. However, the high C3 energy to get to 2011 BX10 is infeasible for current launch vehicle technology. Since asteroid 2006 SJ198 has much more feasible mission requirements, it is chosen as the primary target for this mission. Despite the large cost for this mission, all the launch vehicles within the Delta IV and Atlas V classes are analyzed.

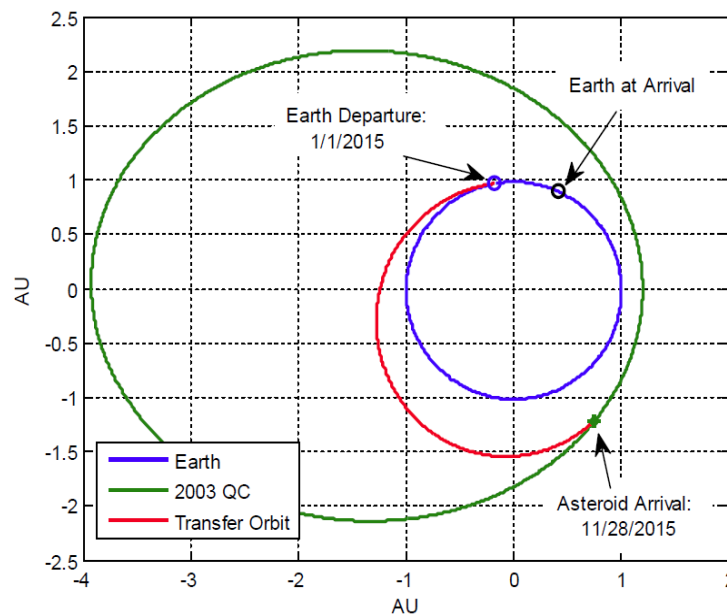


Figure 13. Mission trajectory from Earth to 2003 QC.

C. Asteroid 2003 QC

Asteroid 2003 QC has an estimated diameter of 400 meters, a required C3 of $29.14 \text{ km}^2/\text{s}^2$, and a launch date on January 1, 2015, an appropriate target for the 1000-kg NED mission. Based on the C3 energy requirement, it can be assumed that the amount of energy needed to be provided to the HAIV far exceeds that which can be provided by a Delta II rocket, in either a three stage or two stage plus OTV configuration. Regardless, both Delta II configurations were tested to see if the asteroid could be reached. With an HAIV mass of 4,251 kg, with NED and mass margin included, no three-stage Delta II configuration launch vehicle could carry such a massive payload to the required C3 orbit. The required HAIV/OTV mass to be placed in LEO to reach the target NEO would be approximately 7,435 kg, over 1,400 kg beyond the maximum carrying capacity of the Delta II 7920H launch vehicle. Therefore, a more powerful launch vehicle must be used to achieve the desired orbital injection and transfer requirements.

Since the launch vehicles within the Delta IV and Atlas V classes can support the size of payload to the desired C3 orbit, an OTV is not needed in the design of this mission. With the high C3 mission requirement, the Delta IV Heavy is the launch vehicle of choice for the mission to 2003 QC. However, in the case that the mass decreases after construction and fabrication, a smaller launch vehicle like the Atlas V 551 could potentially be used to perform the same mission. Figure 13 shows the mission trajectory of the HAIV from the Earth to asteroid 2003 QC, and the orbit tracks of both the Earth and asteroid over the transfer time of the spacecraft.

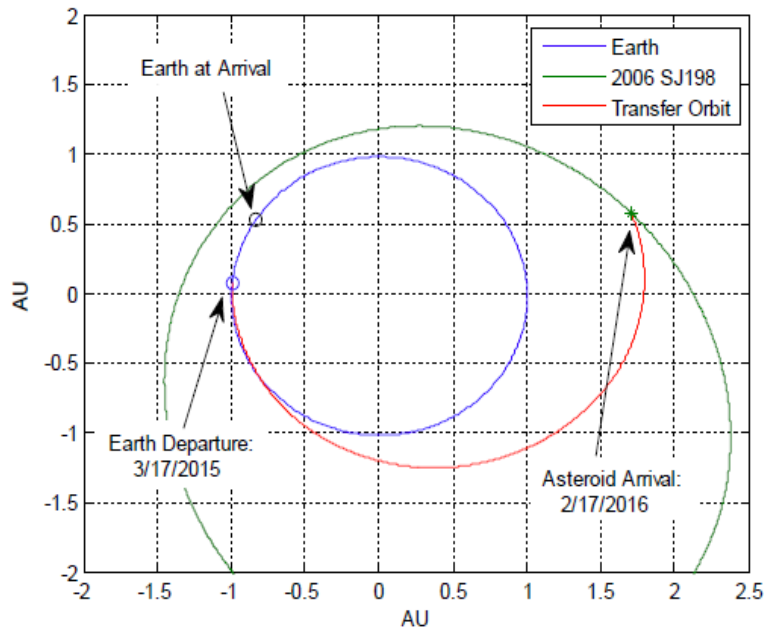


Figure 12. Mission trajectory from Earth to 2006 SJ198.⁶

V. Mission Cost Estimation

Mission cost estimation to design and fabricate the missions is an important task necessary for an early assessment of the mission viability and feasibility. The final total cost of each mission is given as a combination of the cost for the launch vehicle, the HAIV/OTV system, and any fuel for the OTV, if utilized. Initially, the maximum costs for the three mission scenarios were assumed as: \$250M, \$500M,

and \$1B; however, based on the designs of the HAIV/OTV that would be required for the selected target asteroids, those initial cost estimates have been found to be rather modest.

A cost estimation algorithm was developed to determine the costs associated with constructing the HAIV, based on a number of previous spacecraft missions with similar goals and parameters. Spacecraft such as Deep Impact, Stardust, and Dawn were researched to find the cost of developing their spacecraft and a linear polynomial fit was applied to the data to come up with an analytic formula relating spacecraft mass and cost. It is important to note that the mass/cost of the NED was not included when the estimations were made. In addition, the total mass margin was left intact when estimating the cost of the HAIV development, in order for the estimate to be thought of as a relative maximum. The total mass of these spacecraft for the three different demo missions, without NED payloads, are 1,543 kg, 3,251 kg, and 4,242 kg, respectively. Running these masses through the cost estimation algorithm gives spacecraft development costs of approximately \$411M, \$823M, and \$1,057M, respectively. Table 5 shows a cost breakdown for each NED mission, along with a total mission cost. The cost of each mission is limited to:

Table 5. Cost breakdown of three baseline PDT demonstration missions.⁵

	1998 SB15 Mission	2003 QC Mission	2006 SJ198 Mission
Launch Vehicle	Delta II 7920H	Delta IV Heavy	Delta IV Heavy
HNIS Mass (kg)	1543	3251	4220
Launch Vehicle Cost (\$)	100M	325M	325M
HNIS Cost (\$)	411.7M	823.9M	1058M
OTV Cost (\$)	2M	0	0
Total Costs (\$)	513.7M	1148.9M	1383M
30% Cost Margin (\$)	154M	344M	415M
Total Mission Costs (\$)	668M	1.5B	1.8B

i) the launch cost of the specified launch vehicle, ii) the HAIV fabrication, and iii) the OTV fabrication and fuel.⁵

A similar cost analysis was also run using NASA's Advanced Mission Cost Model (AMCM)⁵, to get an estimate of the costs of these three missions. The estimates from the AMCM for each HAIV came out to be \$616M, \$979M, and \$1,149M, respectively, in 2004 US dollars. These estimates are ballpark approximations, mostly due to the fact that these HAIV designs don't exactly fit into a single mission category from the available choices. However, the estimates at least verify that the estimates shown in Table 5 are in the appropriate cost range. Given the total cost estimates discussed above and taking the results from the AMCM into consideration, the revised mission costs would be approximately \$668M, \$1.5B, and \$1.8B, respectively, accounting for mission operations costs by adding 30% of the estimated total costs.⁵ A more detailed discussion on cost estimates as well as technical assessments of a variety of NEO disruption missions can be found in Ref. 5.

VI. Conclusion

A concept of using a fore body (a leader spacecraft) to provide proper kinetic-energy impact crater conditions for an aft body (a follower spacecraft) carrying nuclear explosives has been investigated in this paper as a technically feasible option for the most probable impact threat of NEOs with a short warning time (e.g., much less than 10 years). Requirements of a nuclear disruption mission prove to be challenging due to direct intercept speeds of greater than 10 km/s, nuclear disruption technique, impact speed limit of 300 m/s for state-of-the-art NED fuzing mechanisms, and structural and thermal loads acting on the spacecraft. The development of the HAIV has been discussed which includes thermal shielding simulations, selection of fuzes and optical instruments, terminal-phase guidance operations incorporating a 1 millisecond time delay for NED detonation, and other secondary HAIV configurations. Preliminary designs and analyses of flight demonstration missions with cost estimations have been presented for a baseline hypervelocity nuclear interceptor system carrying 300-kg, 1,000-kg, and 1,500-kg NED. Although an ideal primary objective for a PDT demonstration mission should be to test and validate the HAIV using a real NED, a small explosive device or a representative “dummy” payload could be used as an alternative payload option to verify and validate the planetary defense space system technologies. Other optional mission goals can be accomplished such as sending an orbiter spacecraft to observe the effective disruption/deflection or collect NEO composition samples and return it to Earth for analysis. A list of potential asteroid targets for a PDT demonstration mission has been shown for launch windows between the year 2015 to 2020. The current as well as planned studies at the Iowa State ADRC would enable an important step forward for this area of emerging international interest, by finding the most cost effective, reliable, versatile, and technically feasible solution to the NEO impact threat mitigation problem, which is now one of NASA’s Space Technology Grand Challenges.

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