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Interplanetary Ballistic Missile (IPBM) System Architecture Design for Near-Earth Object Threat Mitigation

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Abstract

This paper presents a preliminary conceptual design of an interplanetary ballistic missile (IPBM) system architecture for deflecting and/or disrupting a near-Earth object (NEO) that is on a collision course towards the Earth. The proposed IPBM system basically consists of a launch vehicle (LV) and an integrated space vehicle (ISV). The ISV consists of an orbital transfer vehicle and a terminal maneuvering vehicle carrying a nuclear explosive device (NED). A Delta IV Heavy lift vehicle is chosen as a baseline LV of a primary IPBM system for delivering two, 750-kg NED payloads to a target NEO. Secondary IPBM systems using a Delta IV M+ and a Taurus II with a smaller ISV are also proposed. The proposed IPBM system architectures will be applicable with minimal modifications to a wide range of NEO deflection missions with varying requirements and mission complexity.

Nomenclature

ACS = Attitude Control System
AOI = Apophis Orbit Insertion
AU = Astronomical Unit
C3 = Earth Escape Energy
CDHS = Command and Data Handling System
EOL = End of Life
GNC = Guidance, Navigation, and Control
GT = Gravity Tractor
GTO = Geostationary Transfer Orbit
IPS = Ion Propulsion System
ISV = Integrated Space Vehicle
KI = Kinetic Impactor
LiDAR = Light Detection and Ranging
M = Mean Anomaly
MMH = MonoMethyl Hydrazine
NFOV = Narrow Field of View
NEO = Near-Earth Object
NED = Nuclear Explosive Device
NSTAR = NASA Solar Electric Propulsion Technology Application Readiness
OTV = Orbital Transfer Vehicle
RCS = Reaction Control System
TMV = Terminal Maneuvering Vehicle
WFOV = Wide Field of View
YE = Yarkovsky Effect
Ω = Right Ascension of Ascending Node
ω = Argument of Periapsis

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I. Introduction

Asteroids and comets have collided with the Earth in the past and are predicted to do so in the future. These collisions have a significant role in shaping Earth's biological and geological history, most notably the extinction of the dinosaurs 65 million years ago. One recent event is the 1908 Tunguska impact in Siberia, which released an explosion equivalent to approximately five to seven megatons of TNT. This explosion had enough power to destroy a 25 km radius of forest. It has been estimated that an impact from the asteroid 99942 Apophis would release approximately 900 megatons of energy, over 130 times the Tunguska event. The results of a collision of this magnitude in a highly populated area would be catastrophic. To date, an estimated 20,000+ potentially hazardous asteroids orbit within the vicinity of the Earth, with more being discovered every year.

Now is the time to further consider actual development of space system architectures for an inevitable space mission of deflecting and/or disrupting a near-Earth object (NEO) that is on a collision course towards the Earth. An NEO is defined as an asteroid or comet that comes near to or crosses the Earth's orbit. Detailed descriptions of the previous work on the detection, characterization, and mitigation of NEO impact threats can be found in [1-5].

This paper describes a preliminary conceptual design of an interplanetary ballistic missile (IPBM) system architecture for deflecting and/or disrupting an NEO that is on a collision trajectory towards the Earth. An IPBM system basically consists of a launch vehicle (LV) and an integrated space vehicle (ISV). The ISV consists of an orbital transfer vehicle and a terminal maneuvering vehicle carrying a nuclear explosive device (NED). The proposed IPBM system architecture utilizes the nuclear standoff or buried (subsurface) explosions for deflecting and/or disrupting an NEO [6-10]. It can also carry non-nuclear payloads (e.g., kinetic impactor or gravity tractor). The Delta IV Heavy is chosen as a baseline LV of a primary IPBM system for delivering a total 1500 kg NED payload to a target NEO. Two secondary IPBM systems and a smaller ISV are also described in this paper.

II. Asteroid 99942 Apophis

In this paper, asteroid 99942 Apophis is used as an illustrative example for conceptual mission analysis and design of the IPBM systems.

NEOs are asteroids and comets that have been nudged by the gravitational attraction of nearby planets into orbits that allow them to enter the Earth's neighborhood. Most NEOs, such as Apophis, have been identified spectroscopically as an Sq-type asteroid. This means the composition is comprised of magnesium-silicates (stones) mixed with metals. These types of asteroids are found in the main asteroid belt within 2.2 AU and 3 AU [1, 2].

Apophis was first discovered in 2004. There has been a great deal of discussion over the threat posed by Apophis before and after 2029. This possible threat and its comparable composition to other NEOs make Apophis an ideal reference model for an IPBM deflection mission study. Tables 1 and 2 show the physical parameters and orbital elements of Apophis, respectively. The diameter of Apophis is estimated at 270 meters and its orbital period is 323 days.

Apophis will pass within geostationary orbit on April 13th, 2029. There is a 1 in 45456 chance that Apophis will pass through a 600-m "keyhole" during its 2029 Earth encounter. If this should occur, Apophis would enter a resonant orbit and impact the Earth on April 13th, 2036. Keyholes are very small regions of the first encounter b-plane such that if an NEO passes through them, it will have a resonant return impact with the Earth.

An extremely small amount of impact ΔV (approximately 0.05 mm/s) in 2026 will be sufficient to move Apophis out of a 600-m keyhole area by approximately 10 km in 2029, if it is going to pass through a keyhole, to completely eliminate any possibility of its resonant return impact with the Earth in 2036. However, a recent study in [11] shows that the Yarkovsky effect and solar radiation pressure can cause 20 - 740 km of position change of Apophis over the next 20 years leading into the Earth flyby in 2029. It was also found in [11] that small uncertainties in the masses and positions of the planets and the Sun can cause up to 23 Earth radii of prediction error for Apophis by 2036. Consequently, any NEO deflection effort must produce an actual orbital change much larger than predicted orbital uncertainties from all sources.

The proposed IPBM system architectures are to be applicable with minimal modifications to a wide range of NEO deflection missions with varying requirements and mission complexity, such as a worst-case scenario of deflecting asteroid 99942 Apophis after 2029 [12-13]. Such unmanned deflection missions for Apophis might also be appropriately coordinated with a crewed mission to Apophis in 2028-2029 described in [14].

Table 1: Physical Parameters of Apophis

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

Table 2: Orbital Elements of Apophis at Epoch 2455000.5 (2009-Jun-18.0) TDB

Orbital Elements	Value
Semi-Major Axis (AU)	0.9224
Eccentricity	0.1912
Inclination (deg)	3.3314
Ω (deg)	204.4425
ω (deg)	126.4042
M (deg)	117.4684

III. NEO Deflection Mission Concepts

Although Apophis is used as a reference asteroid throughout this paper, it is important to note that all information regarding deflection options and terminal phases can be extended to any other NEO deflection missions. In other words, the techniques to be described are not only applicable for Apophis.

A. Deflection Options: NED, GT, KI, YE

There are a number of options that exist for perturbing the orbit of an NEO. This study focuses on four, main deflection options. The first option utilizes a nuclear explosion at a specified standoff distance from the target NEO to cause its velocity change by ablating and blowing off a thin layer of the surface. The nuclear standoff explosion could also fragment part of the asteroid, depending on the detonation altitude of the NED. The surface or subsurface use of nuclear explosives is assessed to be more efficient, although they may run an increased risk of fracturing the target asteroid [9, 10].

A kinetic impactor (KI) could also be used as a deflection option. This deflection mission relies on crashing a spacecraft into the surface of the asteroid. By doing so, the resulting linear momentum transfer from the impact would essentially alter the asteroid's orbit. To make a KI mission more effective, the impactor mass as well as the relative impact speed should be maximized [12, 13].

The next deflection option incorporates the use of a gravity tractor (GT). A GT mission uses the gravitational attraction between the asteroid and a body to slowly change the asteroid's orbit. The GT maintains a certain distance from the asteroid, and uses a propulsion system to counteract perturbations from the asteroid. This technique is done very slowly however, and the mission requires a longer lifetime to complete its objective [4, 5].

The last technique to be discussed is the manipulation of the Yarkovsky Effect (YE). The YE is a force that results from the thermal radiation of a rotating body in space. Since different areas of the rotating body radiate at different strengths due to the uneven thermal gradient, a perturbation force is produced that affects the orbit of the asteroid. By controlling the YE on an asteroid, the path of the asteroid can be altered. This method can be implemented by deploying a solar sail to keep the sun's radiation from hitting the asteroid's surface. This changes the asteroids thermal properties; hence, changing the force produced by the asteroid's radiation. The resulting momentum change can slightly perturb the asteroid's path [4, 5].

This paper will look at the application of these techniques with respect to Apophis. Each deflection technique is executed during a mission's terminal phase. Direct intercept or rendezvous option makes up the terminal phase category.

B. Direct Intercept Mission

Direct intercept missions theoretically require no ΔV maneuver at arrival near a target asteroid. However, once approaching an NEO, fuel may be needed for trajectory correction maneuvers. These maneuvers ensure a successful intercept of the asteroid. Only two deflection missions perform a direct intercept. A KI mission always performs a direct intercept. An NED mission can also employ a direct intercept; however, this deflection option will become more reliable with a rendezvous mission.

A direct intercept mission can be useful since the mission requires less ΔV than a rendezvous mission. This means that a direct intercept mission requires less propellant, which leaves more available mass for the deflection mission payload. A direct intercept mission is also executed more quickly than a rendezvous mission. This is because the relative speed between the impacting object and the asteroid is maximized in order to produce a successful deflection.

C. Rendezvous Mission

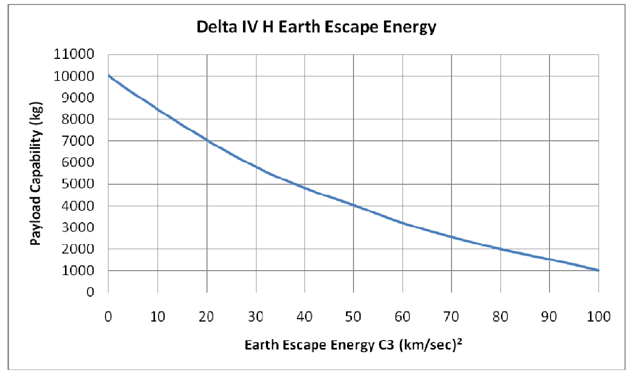
A rendezvous terminal phase can include a number of maneuvers in which the ISV does not directly impact an NEO. For an NED rendezvous, the ISV performs proximity operations to trail behind or fly ahead the target NEO. This allows the ISV to more easily control the NED detonation process.

Rendezvous missions, unlike direct intercept missions, require additional ΔV for maneuvers once the target NEO is reached. Also, rendezvous missions in general require a longer mission lifetime; however, the rendezvous terminal phase provides additional deflection options for consideration. A rendezvous terminal phase can be used for a NED, GT, or YE missions.

D. Launch Vehicle Options

In regards to an NEO deflection mission design, launch vehicle selection is an important process. Since there are many to choose from, it is important to examine the trade-offs that occur from making a selection. Smaller launch vehicles are less costly and more readily available, but the payload capability is very limited. Larger launch vehicles such as the Delta IV family help to increase the payload capabilities, but are rarely available for urgent launches. For the reference mission to Apophis, the Delta IV H has been chosen as a baseline. This launch vehicle provides the highest mass and volume capabilities to accommodate the ISV with a primary NED payload.

A trade-off study to determine the maximum arrival mass at Apophis obtainable from a Delta IV H launch has been performed. Various parking orbits and C3 values were used in the trade-off study. The C3 capabilities of the Delta IV H can be seen in Figure 1(a) and the maximum mass delivered to commonly used orbits for the Delta IV H is listed in Figure 1(b). As shown in Figure 1(a), it is possible for the Delta IV H to provide the ISV with any C3 that is equal to or lower than the minimum C3 value found at any launch window. Also, starting from a parking orbit in LEO forces the ISV to use a massive upper stage to achieve an interplanetary trajectory. On the contrary, launching from a GTO allows the ISV to utilize its own propellant to attain the same trajectory. Throughout this study, it has been determined that departing from a GTO allows for the most usable mass to be delivered to Apophis. This allows for the use of larger payloads and adds more flexibility to the mission.



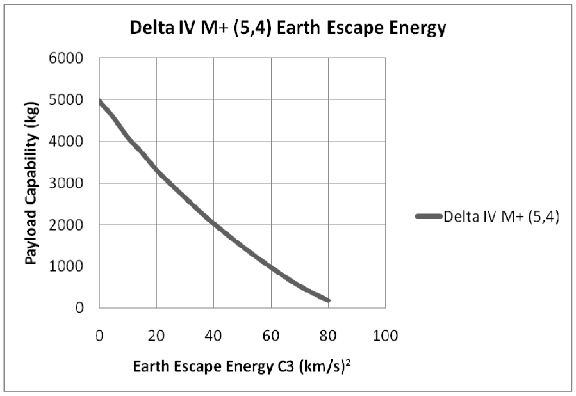
(a) Delta IV Heavy's Payload Mass versus C3.

Mission	Orbit	Delta IV Heavy
LEO	407 x 407 km 28.7 deg inclination	22592
GTO	35,786 x 185 km 28.7 deg inclination	13000
C3 (0.0 km/sec)^2	185-km (100-nmi) Perigee 28.7 deg inclination	10000
C3 (10.0 km/sec)^2	185-km (100-nmi) Perigee 28.7 deg inclination	8425

(b) Delta IV Heavy's Payload Mass Capabilities (in units of kg)

Figure 1: Delta IV Heavy's C3 Values and Launch Mass Capabilities [15].

The Delta IV M+ (5,4) is considered as the first of the two secondary IPBM systems. The Delta IV M+ can carry a smaller ISV with a primary KI payload. The (5,4) represents the five meter diameter fairing and four strap-on graphite epoxy motors. This is the largest available fairing for the Delta IV M+, which allows us to maximize the size of the ISV. The same trade-off study is used for the secondary IPBM design system using Delta IV M+. The C3 values are analyzed as well as the launch mass capabilities to commonly known orbits. The most deliverable mass to Apophis is achieved by means of the same GTO orbit. The resources used in the tradeoff study are shown in Figure 2. The secondary IPBM system is described in more detail later.



(a) Payload Mass versus C3.

Mission	Orbit	Delta IV M+ (5,4)
LEO	407 x 407 km 28.7 deg inclination	13354
GTO	35,786 x 185 km 28.7 deg inclination	6500
C3: 0.0 $(km/sec)^2$	185-km (100-nmi) Perigee 28.7 deg inclination	4986
C3: 10.0 $(km/sec)^2$	185-km (100-nmi) Perigee 28.7 deg inclination	4114

(b) Launch Mass Capabilities

Figure 2: Delta IV M+ Launch Mass Capabilities [15].

The Delta II launch vehicle is also a feasible option for small-payload deflection missions or for asteroid reconnaissance missions. The Delta II is capable of carrying a 1250-kg payload to an interplanetary trajectory. However, the Taurus II will soon replace the Delta II. The Taurus II carries a larger payload, while attaining the same desired trajectory. For this reason, the Taurus II is a better fit for small-payload deflection or reconnaissance missions. As seen in Figure 3, the Taurus II launch vehicle can carry approximately 1400 kg while attaining an interplanetary trajectory. This is made possible by using an enhanced second stage instead of the standard, solid-motor second stage. This enhanced second stage is a liquid fuel stage, which uses a methane and LOX fuel mixture. The enhanced second stage Taurus II provides a more cost-effective option than the Delta IV M+. Although the Delta IV M+ is capable of carrying a larger payload mass, the Taurus II can launch a smaller ISV on a direct intercept mission, which can still achieve successful deflection.

Illustrative drawings of the Delta IV Heavy, Delta IV M+ (5,4), and the Taurus II launch vehicles can be seen in Figures 4 through 6.

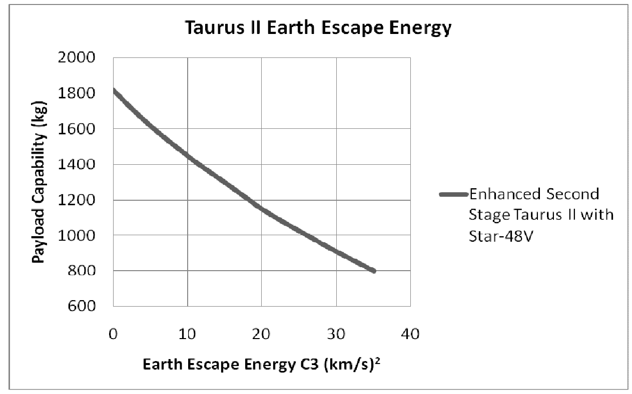


Figure 3: Taurus II Payload Capability.

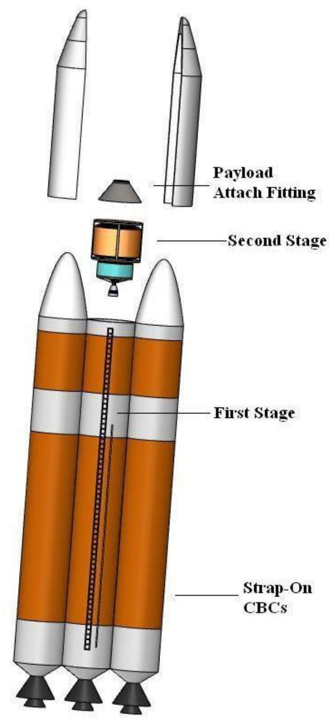


Figure 4: Delta IV Heavy Launch Vehicle.

III. Mission Analysis and Design

Preliminary mission analysis and design have been performed using software developed in-house at the ADRC. The minimum ΔV for each departure date is found by calculating the necessary ΔV for arrival dates ranging from 200 to 600 days after departure. The calculations are carried out with the use of ephemeris data and a computer program written to solve Lambert's problem for each launch and arrival date combination. For the approach taken, the use of two-body orbital dynamics and impulsive maneuvers are assumed. Due to the complexity and large computation time, phasing orbits have not been considered.

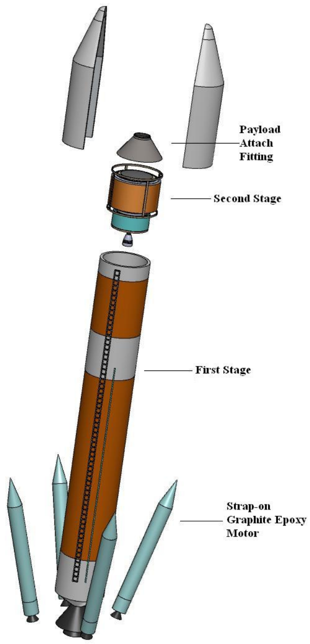


Figure 5: Delta IV M+ (5,4) LV.

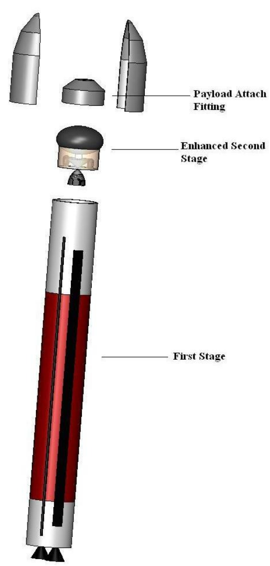


Figure 6: Taurus II LV.

To allow sufficient opportunity for the ISV development, only launch windows after 2018 are considered. Also, to ensure sufficient time for a secondary GT or YE mission, launches after 2024 are not considered. This time span corresponds well to the launch windows found. Launch windows are found by estimating the total ΔV to 3.5 km/s. This helps to allow for a sufficient margin of error, 0.25 km/sec. As seen in Figure 7, May 5, 2020 and April 13, 2021 are two launch dates yielding the smallest total mission ΔV.

A. Launch Window Analysis

Analysis of Figure 7 results in selecting a range of dates in which the total ΔV is less than or equal to 3.5 km/s. Figure 8 shows ΔV plots for dates between 2020 and 2022. During these two years, the total ΔV is a minimum at two different dates. These two dates are chosen for launch windows. Each optimal launch date is shown in Figure 9 and illustrates the minimum ΔV opportunity for each launch window. Each ISV must be capable of wide range of Earth departure ΔV (1.258-2.007 km/s) and Apophis arrival ΔV (0.366-1.767 km/s). These ΔV requirements demand a restartable, bi-propellant fuel system on the OTV.

Launch dates other than those shown in Figure 9 were also found, but didn't meet the launch and arrival date requirements. Launch windows appeared to come in sets within a two to three year range. Each set appears to only occur during periods when the relative Earth-Apophis distance is less than one AU. Figure 10 shows the Earth-Apophis distance that has a minimum approximately every 7.8 years. This means the next set of launch windows, with relatively small ΔV requirements, would not occur until the 2028-2030 ranges. Any GT, KI, YE mission occurring only one to two years prior to the Apophis-Earth encounter on April 13, 2029 has almost no chance of perturbing Apophis out of any gravitational "keyhole." For this reason, the launch dates found are the last launch windows for a GT, KI, or YE mission to Apophis. Therefore, only launch dates earlier than 2024 were considered for analysis

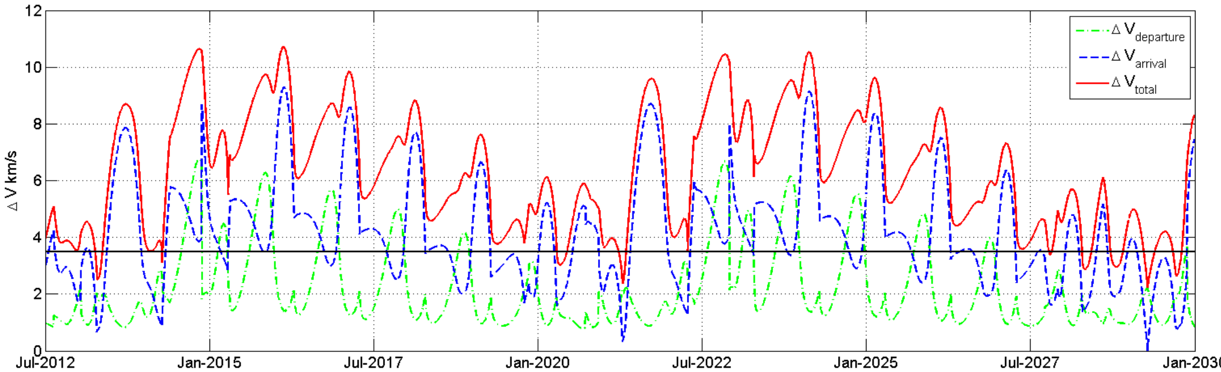


Figure 7: Departure, Arrival, and Total ΔV for Minimum ΔV Trajectories to Apophis.

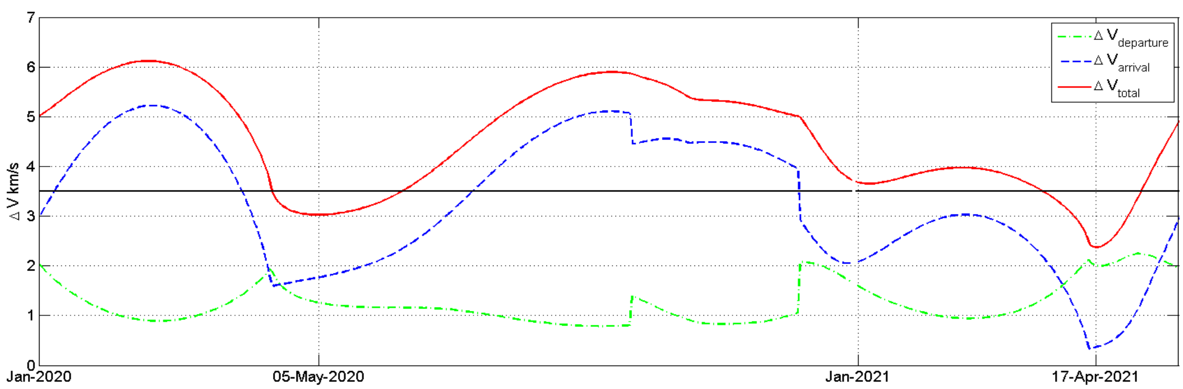
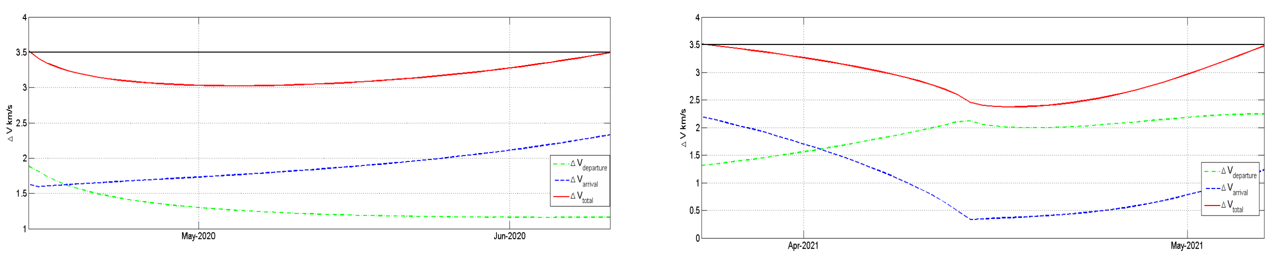


Figure 8: Required ΔV plot. (Similar to Figure 5, but expanded to clearly show each launch window.)



(a) Launch Window 1

(b) Launch Window 2

Figure 9: Asteroid Deflection Mission Design Launch Windows.

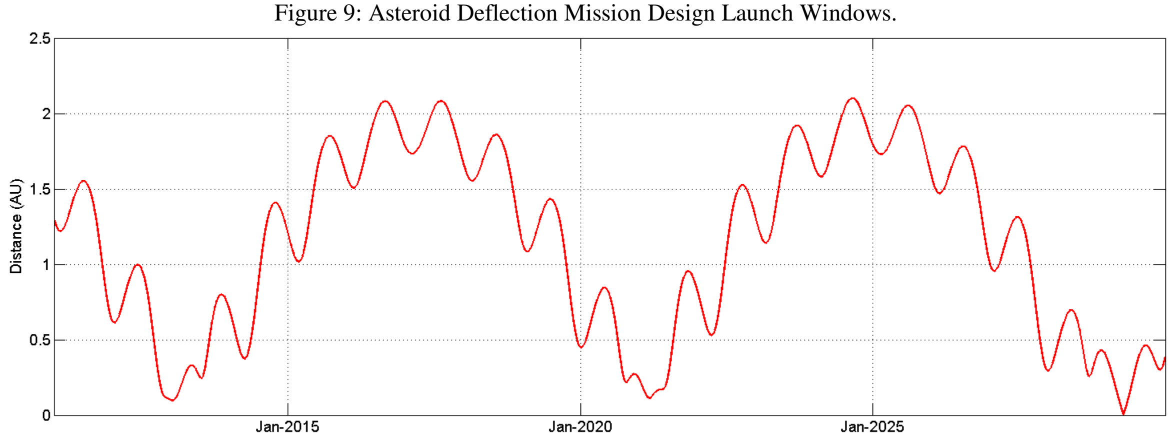
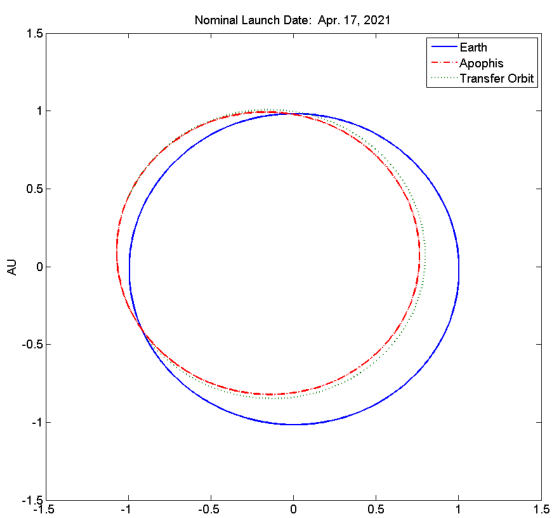


Figure 10: Earth-Apophis distance for January 2012 through January 2030.

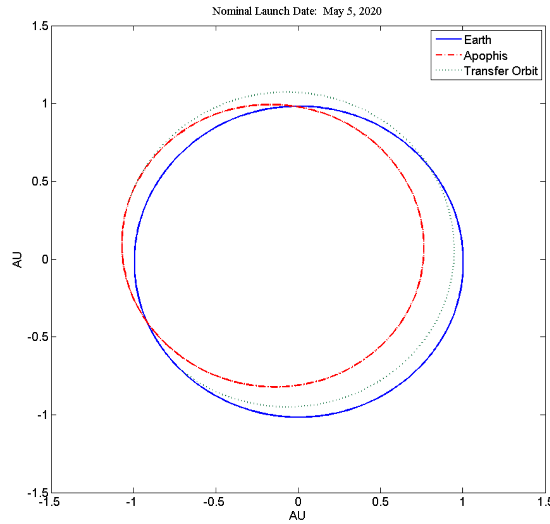
The trajectories for launch windows 1 and 2 are shown in Figure 11. Figure 11(a) illustrates the trajectory for launch window 1, which follows the Earth's orbit closely before the Apophis rendezvous. However, in Figure 11(b), the trajectory for launch window 2 follows Apophis's orbit closely before the final rendezvous. Typical interplanetary cruise times range from 8 to 9 months.

Table 3: A Rendezvous ΔV Mission Profile for Launch Windows 1 & 2.

Mission Information	Launch Window 1	Launch Window 2
Earth Departure		
Departure Date	5-May-2020	17-Apr-2021
Departure C3	11.024	28.808
Departure ΔV (km/s)	1.258	2.007
Transfer Orbit		
Semi-Major Axis(AU)	0.968	0.940
Eccentricity	0.132	0.169
Inclination(deg)	2.555	3.176
Ω(deg)	225.122	207.375
ω(deg)	115.692	122.253
Departure ν(deg)	244.357	237.779
Arrival ν (deg)	179.262	185.997
TCM ΔV (km/s)	0.189	0.301
Apophis Arrival		
Arrival Date	3-Jan-2021	13-Jan-2022
Arrival C3	3.121	0.134
Arrival ΔV (km/s)	1.767	0.366
Totals		
ΔV Margin (km/s)	0.126	0.201
ΔV (km/s)	3.339	2.874
C3	14.145	28.941



(a) Mission Trajectory 1



(a) Mission Trajectory 2

Figure 11: Baseline Mission Trajectories.

B. Rendezvous Mission ΔV

The baseline IPBM system design does not require a separate upper stage for the Earth departure burn. The OTV's bi-propellant system provides the required ΔV to transfer from GTO to an interplanetary trajectory. Trajectory correction maneuvers (TCM) during the interplanetary transfer orbit were assumed to be 15 percent of the Earth departure ΔV, which allows for a sizeable error margin during high-thrust burns. The total ΔV per launch window is designed to have a margin of error of 10 percent of the departure ΔV. This margin represents reserve fuel, room for error, and allows for launches over a wide range of days for each launch window. Additional analysis of each deflection mission must be performed to ensure the ΔV margin estimate is sufficient. Table 3 shows the ΔV breakdown for a rendezvous mission with each launch window.

Launches on days other than the optimum dates consume a portion of the ΔV margin. Arrival dates for launch windows 1 and 2 are given in Table 3 as January 3rd, 2021 and January 13th, 2022. The transfer times for launch window 1 and 2 are 300 and 242 days respectively.

The Delta IV H, primary IPBM system and the Delta IV M+, secondary IPBM system use the same ΔV mission profile. This allows the Delta IV M+ IPBM system to be launched at the same time as the primary IPBM system or replace the launch of the primary IPBM system.

The baseline IPBM mission is designed for a rendezvous with Apophis using launch window 1. In launch window 1, the Delta IV H delivers the ISV to GTO. From here on, the OTV performs all of the remaining ΔV maneuvers. This includes the Earth departure burn (1.258 km/s), TCMs (0.189 km/s) and the rendezvous burn (1.767 km/s). In all rendezvous scenarios, the OTV performs a burn equal to its arrival ΔV. The total ΔV required for this baseline mission is 3.339 km/s and the total ΔV required by the OTV is 3.213 km/s. Figure 12 illustrates the ΔV breakdown for launch windows 1 and 2. Although launch window 2 requires less total ΔV, we have chosen the worst-case scenario for the baseline. For this reason, the baseline design includes all other launch windows.

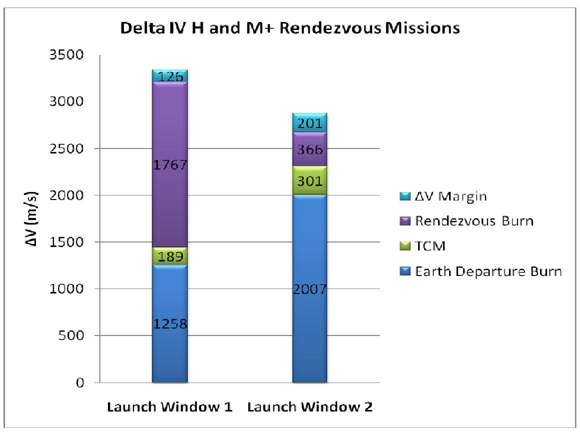


Figure 12: ΔV Breakdown for a Rendezvous Terminal Phase.

IV. Preliminary IPBM System Design

A. Integrated Space Vehicle

The ISV is made up of two, separable space vehicles. The ISV design includes a modification of the existing Dawn satellite into a TMV, as well as the creation of a suitable OTV for interplanetary transfer. The two vehicles are connected by a releasable interstage system allowing the TMV and OTV to separate when necessary.

In order to tailor the TMV to an Apophis mission, some changes in the Dawn satellite design must be made. Starting with the technology used on Dawn, a TMV subsystem design and mass budget is examined.

B. Dawn-based Technology and Subsystems

The Dawn mission is designed to study Ceres and Vesta; the two largest asteroids located within the asteroid belt [16]. This spacecraft is unique in that it utilizes a flight-proven Ion Propulsion System (IPS). This IPS is an expanded version of the ion engine used on the Deep Space 1 spacecraft.

The overall spacecraft design is a modified version of Orbital's STAR-2 series. The bus is constructed of a graphite composite cylinder and is surrounded by aluminum and composite panels for mounting hardware. Dawn's IPS consists of three, 30 cm diameter NSTAR engines that were first tested on Deep Space 1. Approximately 425 kg of pressurized Xenon is used as fuel for the IPS and is housed in the center cylinder. Each engine is two-axis gimbaled and can be throttled at various levels to match power restrictions [16].

At Dawn's farthest distance from the Sun, approximately three AU, the power system needs to provide adequate power for the IPS's operations. This is accomplished by using two, highly efficient solar arrays having a combined area of 36 m^2. A 35 Ah NiH_2 battery is also used to power the satellite systems during launch. The battery also supplements the two power buses during IPS thrusting. Two power buses are needed to provide the IPS with high voltage power.

The attitude control system (ACS) uses star trackers supplemented with gyros for attitude estimations. Reaction wheels, or the reaction control system (RCS), are then used to control the spacecraft attitude. The hydrazine RCS, which is fully redundant and uses a total of twelve 0.9 N thrusters, can also be used for quick orbit-correction maneuvers.

For communication purposes, Dawn uses traveling wave tube amplifiers and four antennas. The antennas include a 1.52-m high gain antenna and three low gain antennas, all of which are compatible with NASA's Deep Space Network.

Fully integrated, Dawn has a dry mass of 725 kg and a wet mass of 1240 kg. Dawn's subsystems serve as a guide for the baseline design of the TMV.

C. Terminal Maneuvering Vehicle

The Terminal Maneuvering Vehicle (TMV) is the vehicle that contains the primary deflection payload. It closely mirrors the Dawn satellite, and contains subsystems nearly identical to Dawn's. A bi-propellant system, independent of the OTV's system, provides the TMV with the ability to perform correction maneuvers at Apophis. The TMV propellant tanks are sized to provide a ΔV of 250 m/s for these maneuvers, which is based on a similar impactor spacecraft of NASA's Deep Impact mission.

The TMV contains all the necessary subsystems to function without the OTV once at Apophis. Mass estimates for subsystems such as the ACS, RCS, TCS, CDHS, and the telecommunications system mirror Dawn's systems. Major changes can be seen in the mechanical structure system, electrical power system, and propulsion system. While a detailed mass estimate has been put together, the systems mentioned are the ones, which pose the largest geometric constraints. Thus, only these systems are broken down in more detail.

The structure has been modeled after the Orbital STAR-2 series, which is the same design used by Dawn. This structure offers space for mounting hardware between the center cylinder and the outer panels as well as the propellant tanks. The TMV's size must be large enough to hold the necessary bi-propellant fuel tanks. However, the dimensions must not be larger than what is allowed inside of the fairing. The total height must also be within limits of the fairing.

Unlike the Dawn spacecraft, the TMV uses a bi-propellant propulsion system. This system is responsible mainly for correction maneuvers at Apophis using RCS thrusters. The bi-propellant system uses MMH as the main propellant and nitrogen tetroxide (N_2O_4) as an oxidizer. A helium pressure tank is also needed to control propellant flow.

Because the ISV may encounter different levels of solar intensity throughout its mission, the solar arrays need to be able to produce the required power at the farthest point from the Sun. For material selection, there are several choices that are available, the most common being Silicon and Gallium Arsenide. Dawn however, used a more efficient triple-junction solar cell that minimizes the area of the solar array. The triple-junction cells, manufactured by Emcore Corporation, have a minimum efficiency of 27.5 percent. These solar cells are radiation resistant and also have a low degradation rate (approximately 0.5 percent). These panels are folded up during launch and deployed once the Earth departure maneuver is completed.

D. Primary NED Payload

The baseline payload consists of two, NED payloads attached to (or located inside of) the TMV. The assumed NED is based on the B83 nuclear weapon. Its size and volume are unknown. However, the device is assumed to fit comfortably in the TMV bus. Since there may be additional room left in the fairing from the baseline design, adjustments can be easily made to the size of the TMV bus if necessary. It should also be noted that a larger payload can still be used if launch window 2 is selected. This is due to the reduced propellant mass, which leaves additional mass that can be applied to the NED.

E. Orbital Transfer Vehicle

The baseline Orbital Transfer Vehicle (OTV) has the largest wet mass at launch. Most of this mass comes from the bi-propellant fuel. Again, the bi-propellant fuel is a combination of MMH and N_2O_4. This is because the OTV acts like a third stage, in that it is designed mainly to transport the TMV to Apophis. In a trade-off study, it has been determined that the OTV is a better fit for this mission than an ATK solid fuel, third stage [17]. This is because more mass can be delivered to Apophis and more room becomes available inside the fairing. The OTV comprises of propellant tanks, an engine/nozzle, and mechanical structure, including an interstage that connects to the TMV. The OTV operates on its own Dawn-like subsystems but is capable of separating from the TMV once the propellant has been spent.

F. Primary IPBM System Design Example

The primary IPBM system consists of a Delta IV H launch vehicle that can deliver a 13,000 kg ISV to a 185 by 35,786 km GTO. The ISV is launched on launch window 1 and uses its own fuel to achieve an interplanetary transfer orbit towards Apophis. During the transfer orbit, system checks and trajectory correction maneuvers are executed. On arrival, the OTV completes a rendezvous burn. After the arrival burn, the OTV is jettisoned and the TMV conducts proximity operational maneuvers until it has reached a desired detonation altitude. The TMV utilizes its own fuel to reach the desirable altitude or possibly landing on Apophis. Once the correct distance is reached, the NED payload is detonated resulting in the deflection/fragmentation of Apophis.

During the interplanetary cruise phase for all terminal phases, the ISV deploys the solar arrays and performs a test of each subsystem prior to tracking Apophis. All correction maneuvers are performed using reaction control system (RCS). Once the ISV is in range of Apophis, tracking is performed with a WFOV camera, moving to a NFOV camera when necessary. Determination of the correct altitude before detonation is accomplished with a LiDAR system and the use of cameras.

The primary IPBM system design is broken down into a preliminary mass estimate table. The baseline ISV consists of an OTV with a dry mass of 800 kg, bi-propellant fuel at 8444 kg, a TMV of 2361 kg, and extra mass of 1395 kg. The TMV holds two, 750 kg NED payloads along with reserve propellant of 284 kg. Table 4 shows the primary ISV mass breakdown. Again, the primary IPBM system is a baseline for the worst-case scenario. More mass is available when selecting other launch windows and/or terminal mission profiles. Four tanks chosen from ATK hold the 8444 kg of propellant fuel needed for launch window 1 [17]. The MMH tank is approximately 3648 L, while the N_2O_4 tank is 3609 L. The combined mass of the tanks is approximately 300 kg. A conceptual configuration of the primary IPBM and ISV can be seen in Figures 15 and 16.

To produce 1 kW at the EOL, a solar cell area of 4.22 m^2 is required. To allow for a margin of error, the solar arrays are designed to have a total area of 4.5 m^2. This generates an estimated 1.07 kW of power at EOL. The ISV is equipped with two solar arrays, each having an area of 2.25 m^2. Each array consists of an accordion folded style panel. A more accurate power budget of the ISV system will be performed in a future study.

Mechanical structure is scaled up due to the unknown size of the NED payload. A mass of 200 kg is only an estimate for the TMV and the OTV but will be revised after more studies. The extra mass will be allocated throughout the ISV once a more accurate mass budget is created.

Launch Window 1: May 5, 2020; Rendezvous	**Delta IV Heavy Primary ISV**	
Vehicle	**System**	**Mass (kg)**
OTV	Mechanical Structure	200
	Propellant Tanks	300
	EPS	123
	ACS	37
	RCS	14
	TCS	44
	CDHS	21
	Telecom	28
	Balance	13
	Uncertainty	20
	Total Dry Mass	**800**
	Total Bi-Propellant	8444
	Total Wet Mass	**9244**
TMV	Mechanical Structure	200
	Propellant Tanks	40
	EPS	123
	ACS	37
	RCS	14
	TCS	44
	CDHS	21
	Telecom	28
	Balance	13
	NFOV	10
	WFOV	12
	LiDAR	15
	Uncertainty	20
	NED Payload	1500
	Total Dry Mass	**2077**
	Total Bi-Propellant	284
	Total Wet Mass	**2361**
ISV	Mass To Apophis	2361
	Wet Mass at Launch	**11605**
	Mass Margin	1395

Table 4: Primary-ISV Mass Budget.

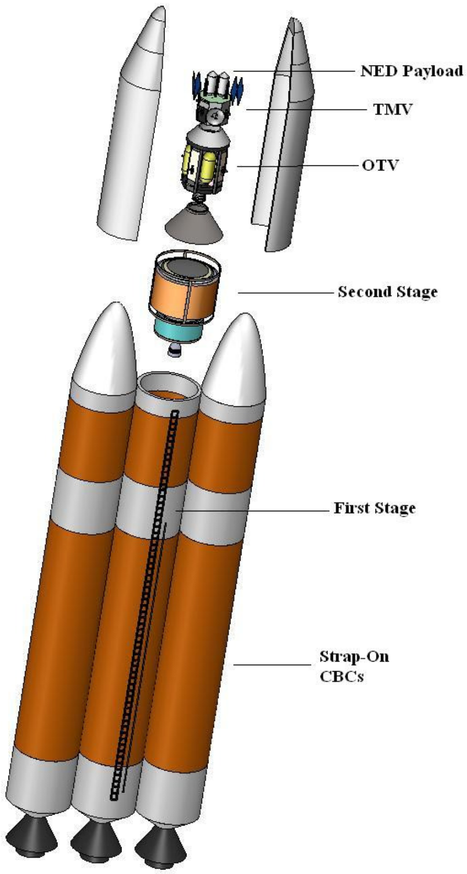


Figure 15: Primary IPBM and ISV Design.

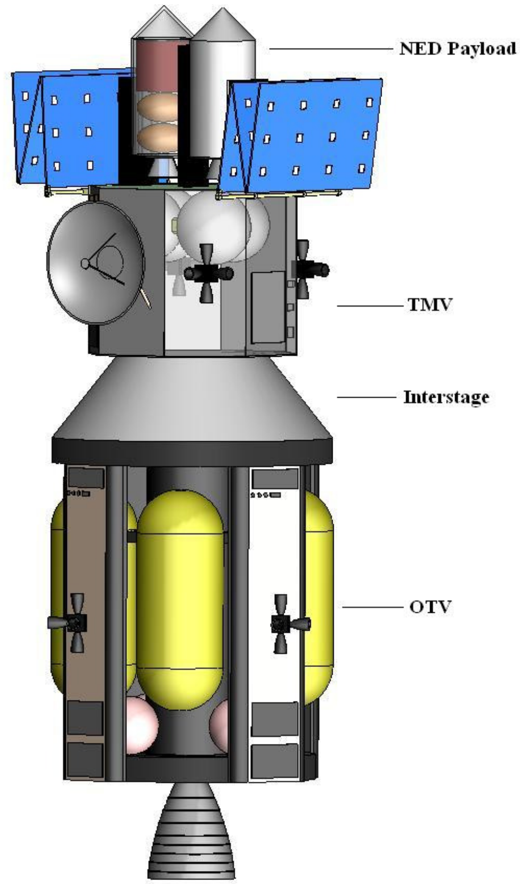


Figure 16: Conceptual ISV Configuration.

G. Secondary IPBM System Design Example

The secondary IPBM system employing the Delta IV M+ can launch 6500 kg into the same GTO as the Delta IV H. The baseline ISV is tailored to fit in the Delta IV M+ fairing. The secondary ISV has smaller propellant tanks in the OTV allowing the ISV to fit in the fairing. The TMV size is also scaled down but has sufficient room for a KI payload.

The secondary option follows the direct intercept terminal phase on launch window 2. The ISV follows the same mission profile as the primary IPBM system but with no rendezvous burn. Once close to Apophis, the ISV again uses cameras and LiDAR to track the asteroid.

The Delta IV M+ IPBM system includes an OTV with a dry mass of 650 kg, 3543 kg of fuel, a TMV of 1755 kg, and extra mass of 552 kg. The TMV consists of a dry mass of 577 kg with a 1000 kg KI payload, and 178 kg of fuel. The secondary OTV is much smaller due to the decrease in propellant mass. Since the propellant amount is much less, smaller tanks are used. The MMH tank is sized at 1531 L and the N_2O_4 tank is sized at 1514 L. This decrease in mass leads to a smaller structural design of the baseline OTV. Table 5 lists the mass breakdown of the secondary ISV. Table 5 shows the mass breakdown of the Delta IV M+ secondary IPBM system.

Launch Window 2: April 17, 2021; Direct Intercept	Delta IV M+ ISV	
Vehicle	**System**	**Mass (kg)**
OTV	Mechanical Structure	200
	Propellant Tanks	150
	EPS	123
	ACS	37
	RCS	14
	TCS	44
	CDHS	21
	Telecom	28
	Balance	13
	Uncertainty	20
	Total Dry Mass	**650**
	Total Bi-Propellant	3543
	Total Wet Mass	**4193**
TMV	Mechanical Structure	200
	Propellant Tanks	40
	EPS	123
	ACS	37
	RCS	14
	TCS	44
	CDHS	21
	Telecom	28
	Balance	13
	NFOV	10
	WFOV	12
	LiDAR	15
	Uncertainty	20
	NED Payload	1000
	Total Dry Mass	**1577**
	Total Bi-Propellant	178
	Total Wet Mass	**1755**
ISV	Mass To Apophis	2357
	Wet Mass at Launch	**5948**
	Mass Margin	552

Table 5: Delta IV M+ Based Secondary ISV Mass Budget.

The other, secondary IPBM system utilizes the Taurus II launch vehicle which can launch 1400 kg to an interplanetary trajectory with a C3 of 11 $(km/s)^2$. The Taurus II deflection mission is limited to only direct intercept terminal phases and launch window 1, due to the limited launch capabilities. This ISV design is unique because it does not require the use of separable vehicles, such as an OTV and TMV. Instead, the ISV is a single satellite which houses the fuel tanks, mission payload, and subsystems. This secondary configuration allows it to fit inside the Taurus II fairing and carry a 600 kg deflection payload or reconnaissance payload. Overall, the Taurus II mission is limited by mass, terminal phase, and launch window.

The Taurus II IPBM system consists of an ISV with a dry mass of 470 kg, 180 kg of bi-propellant fuel, and a NED payload of 600 kg. The secondary ISV mass estimates mirror the Dawn spacecraft.

The MMH tank is sized at 68 L and the N_2O_4 tank is sized at 77 L. Since the tanks are much smaller than the primary IPBM system, two vehicles are not needed. This configuration is necessary due to the geometric and mass constraints of the Taurus II. Table 6 lists the mass breakdown of the Taurus II, secondary ISV.

Launch Window 1; May 5, 2020; Direct Intercept	Taurus II ISV	
Vehicle	**System**	**Mass (kg)**
ISV	Mechanical/Structure	108
	EPS	123
	ACS	37
	RCS	14
	TCS	44
	CDHS	21
	Telecom	28
	Balance	13
	Propellant Tanks	25
	Uncertainty	20
	NFOV	10
	WFOV	12
	LiDAR	15
	Total Dry Mass	**470**
	Total Bi-Propellant	174
	NED Payload	600
	Total Wet Mass	**1244**
	Mass on Arrival of Apophis	989
	Extra Mass	156

Table 6: Taurus II Based Secondary ISV Mass Budget.

V. Secondary Payload Options

Secondary payloads are further discussed here for a GT, KI, or YE mission. GT and YE can only be implemented through the use of a rendezvous mission. The structure of the TMV remains unchanged for the secondary payloads, but reconfiguration of the thrusters and bi-propellant fuel is needed. A brief description of changes for each payload option is provided below.

A. Kinetic Impactor

The KI mission employs a direct intercept terminal phase. The direct intercept phase requires less fuel for the OTV to reach the target NEO and as a result, more mass is allocated in the TMV. For this type of mission, it is important to maximize the arrival mass of the ISV at Apophis. To increase the effectiveness of the KI mission, the relative velocity between Apophis and the ISV should be maximized [12, 13].

B. Gravity Tractor

A rendezvous mission is needed in order to perform the GT mission. It is undetermined at this time if the OTV and the TMV will remain together after the rendezvous burn. The GT utilizes the ion propulsion system (IPS), which provides all necessary thrust to counteract the gravitational attraction of Apophis. The TMV structure is equipped with three NSTAR thrusters, each having an operational lifetime of more than two years at full throttle. By carrying 425 kg of fuel and operating only one thruster at a time, a GT mission can last six years. This provides sufficient time to properly perturb the orbit of Apophis. The GT propellant feed-system is placed inside the TMV structure. The RCS system provides all the necessary stationkeeping controls. Like most missions, the total lifetime is limited only by fuel [4, 5].

C. Yarkovsky Effect

To use the Yarkovsky Effect as a deflection method, a rendezvous terminal phase is required. One approach proposed in the literature uses a large carbon nanotube solar sail. This is deployed from the TMV in order to block the surface of Apophis from the sun. Theoretically, this alters the asteroid's thermal properties. The radiation force of the asteroid is changed, and the orbit of Apophis is altered slightly over time. The solar-sail payload is housed within the TMV structure. More research is currently being performed on the effectiveness and practicality of this deflection mission [18].

VI. Conclusions

The purpose of this paper was to describe practical, IPBM system architectures for various NEO deflection missions in the future. A baseline or reference system architecture has been designed using asteroid Apophis as a reference target so that many different mission options can be utilized by making minor changes in the proposed baseline architecture. The Delta IV Heavy launch vehicle was considered as a primary launch vehicle. However, many other mission options exist. The Taurus II launch vehicle, different mission payloads such as a KI, GT, or YE, and a direct intercept terminal phase highlight some of the available secondary options for IPBM system design. The baseline IPBM system architecture described in this paper was intended to be applicable with minimal modifications to a wide range of NEO deflection missions with varying requirements and mission complexity.

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