

Spacecraft Mission Design for The Mitigation of The 2017 PDC Hypothetical Asteroid Threat

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Abstract

This paper presents a detailed mission design analysis results for the 2017 Planetary Defense Conference (PDC) Hypothetical Asteroid Impact Scenario, documented at <https://cneos.jpl.nasa.gov/pd/cs/pdc17/>. The mission design includes campaigns for both reconnaissance (flyby or rendezvous) of the asteroid (to characterize it and the nature of the threat it poses to Earth) and mitigation of the asteroid, via kinetic impactor deflection, nuclear explosive device (NED) deflection, or NED disruption. Relevant scenario parameters are varied to assess the sensitivity of the design outcome, such as asteroid bulk density, asteroid diameter, momentum enhancement factor (β), spacecraft launch vehicle, and mitigation system type. Different trajectory types are evaluated in the mission design process from purely ballistic to those involving optimal midcourse maneuvers, planetary gravity assists, and/or low-thrust solar electric propulsion. The trajectory optimization is targeted around peak deflection points that were found through a novel linear numerical technique method. The optimization process includes constrain parameters, such as Earth departure date, launch declination, spacecraft/asteroid relative velocity and solar phase angle, spacecraft dry mass, minimum/maximum spacecraft distances from Sun and Earth, and Earth/spacecraft communications line of sight. Results show that one of the best options for the 2017 PDC deflection is solar electric propelled rendezvous mission with a single spacecraft using NED for the deflection

Keywords: planetary defense, asteroid deflection, mission design, astrodynamics

1. Introduction

In this paper we present the methodology and results for our analysis of mission options for responding to the 2017 PDC hypothetical asteroid impact scenario. The scenario parameters are documented at <https://cneos.jpl.nasa.gov/pd/cs/pdc17/>. We consider space-based response options including reconnaissance of the asteroid prior to an in-space mitigation attempt, the mitigation mission itself (deflection or disruption of the asteroid), and missions to monitor the asteroid during and/or after the mitigation attempt. We study two candidate technologies for asteroid deflection: kinetic impactor (KI) spacecraft and nuclear explosive devices (NEDs). KI spacecraft collide with an asteroid to change the

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asteroid's heliocentric velocity via conservation of linear momentum. NEDs are detonated near the asteroid's surface so that the radiation released by the NED ablates a thin layer of asteroid surface, which then blows off rapidly, acting like a brief but very forceful rocket thrust on the asteroid. Both the KI and NED detonation techniques are impulsive in nature, meaning that they induce nearly instantaneous changes in the asteroid's velocity. The change in the asteroid's velocity alters its heliocentric orbit, causing it to miss Earth rather than collide years later.

The study of this hypothetical asteroid impact scenario is motivated by the need to be prepared to respond to an actual asteroid impact threat whenever one is discovered. At present, 16106 near-Earth objects (NEOs) have been discovered, and 1797 of them are classified as Potentially Hazardous Objects (PHOs), meaning that each of them has a Minimum Orbit Intersection Distance (MOID) with Earth ≤ 0.05 au and absolute magnitude, H , ≤ 22.0 (corresponding to diameter ≥ 140 m, assuming a geometric albedo of approximately 0.14). NEOs are defined as asteroid or comets whose orbits have a perihelion < 1.3 au. Current NEO population models indicate that while just over 90% of the 1000 m diameter and larger NEOs have been discovered, over 10000 NEOs with diameters between 100 m and 1000 m remain to be discovered, and at least several million NEOs between 10 m and 100 m remain to be discovered. These model predictions, combined with the 0.5 MT Chelyabinsk asteroid impact that injured ~ 1600 people in Russia on February 15, 2013, indicate that the threat of a devastating NEO impact on Earth is quite real, and that we should be prepared to take effective defensive action on short notice.

In this hypothetical asteroid impact scenario, the near-Earth asteroid (NEA) designated 2017 PDC is discovered on March 6, 2017. Its H is calculated to be 21.9 ± 0.4 , but its geometric albedo is unknown. Because the albedo can, in principle, be in the range of 0.03 to 0.6, the asteroid's diameter could be anywhere from 60 m to 385 m, with the most likely range of possible diameters being 100 m to 250 m. It is on a rather eccentric orbit, with a perihelion of 0.88 au and aphelion of 3.60 au. Thus, the asteroid's orbit period is approximately 3.35 years. Additionally, the asteroid's orbit plane is inclined by 6.3° to Earth's orbit plane. The high eccentricity of the asteroid's orbit actually aids deflection, allowing it to be deflected further from Earth for a given amount of momentum change. However, the asteroid's appreciable orbit inclination and long orbit period make it both difficult to observe from Earth and challenging for a spacecraft to rendezvous with.

Based on initial observations, the asteroid has a small probability of Earth impact (1/40000) in July of 2027. After another month of observations, the probability rises to 0.2/100. By this time the asteroid has reached its point of closest approach to the Earth, 0.13 au, and won't pass that close to Earth again until the possible impact time in 2027. The asteroid is too far away to be detected by Goldstone radar, and is too far south to be detected by Arecibo radar. By early May 2017, about 2 months after discovery, the probability of Earth impact in July 2027 has risen to about 1%. In our analysis we assume that the probability of Earth impact continues to rise for another couple of months, reaching an actionable level by the beginning of August 2017. We needed to make this assumption in order to derive the earliest allowable date of spacecraft launch in our mission analyses.

2. Study Outline

The planetary defense scenario from 2017 PDC allows for a trade-off between different mission options. As a impact simulation scenario, this paper assumes that space assets currently available will also be at the mission's disposal. With this, different system options are available for the mission design, such as: launcher, launch date, deflection method, trajectory type, and propulsion system. This section outlines the advantages and disadvantages of the aforementioned systems for the mission design, as well as how these options are included in the trade-off process. Each trajectory is designed to support a robust mission, with the basic requirement of avoiding the Earth impact on July 21, 2027. The trajectories take into account realistic mission constraints that are outlined and justified in Table 1. In addition to the trajectory constraints, the mission is required to perform a survey of the target asteroid before the deflection. A survey mission prior to a deflection launch is important to characterize 2017 PDC and improve knowledge on its trajectory as precise navigation and targeting are key to this mission.

The Earth impact in 2027 can be avoided by changing the asteroid's trajectory, which for an asteroid with the size and orbital geometry of 2017 PDC can only be made by changing its velocity. A change of velocity imparted to the asteroid is small, orders of magnitude smaller than the asteroid's velocity. However, if done years in advance, the velocity change leads to a relatively large difference in position that is sufficient to avoid an impact at the Earth's close approach point. Therefore, independent of

Table 1: Mission constraints

Constraint	Value	Reason
Launch date	after Aug. 1, 2019	2 years after the asteroids probability of Earth impact rises to 10%.
Launch declination	± 28.5	Declination bounds for the Kennedy launch complex.
Asteroid encounter phase angle	≤ 120	Upper limit to have enough of the asteroid illuminated for the spacecraft's terminal guidance system.
Sun minimum distance	0.7 A.U.	Lower limit for the spacecraft design to handle the more aggressive thermal and radiation environments.
Sun maximum distance	3.5 A.U.	Upper limit to design a large spacecraft (complicated) enough to handle power generation and Earth communications at greater distances is probably not compatible with a rapid spacecraft build timeline.
Earth Angle at asteroid encounter	≥ 3	Lower limit for the Deep Space Network to guarantee a viable RF link with the spacecraft.

the trade-off performed by the different available systems, the trajectory design has the fundamental objective of changing 2017 PDC's velocity in such way that it maximizes Earth's close approach distance - currently an impact. Details of how the optimization is performed is out of the scope of this mission design paper. In short, the close approach distance is the objective to be maximized and it can be described as the asteroid's radius of the perigee on the Earth-spacecraft two body frame,

$$J = r_p = a(1 - e) \quad (1)$$

where, a and e are the semi-major axis and eccentricity of the Earth-spacecraft two body problem, respectively. The optimization tool then calculates the best time and direction that a deflection needs to be made in order to obtain the maximum value in Eq. 1.

2.1. Deflection methods

This study considers two basic deflection methods: kinetic impactor and nuclear explosive device (NED). These methods are considered the most effective based on current technology and lead times less than 25 years. As time from launch to impact is relatively short, gravity tractor deflection type missions are not considered.

The kinetic impactor method consists in changing the asteroid's velocity by intercepting the target with a spacecraft. This imparted velocity change is based on the momentum transfer, which depends on the spacecraft's arrival velocity and mass, as described on Eq. 2.

$$\Delta \mathbf{v} = \mathbf{v}_\infty \beta \frac{m_{S/C}}{m_{S/C} + m_{asteroid}} \quad (2)$$

where, $\Delta \mathbf{v}$ is the velocity change imparted on the asteroid by the spacecraft's impact, \mathbf{v}_∞ is the spacecraft's arrival velocity relative to the asteroid, β is the momentum enhancement factor which encompasses the plasticity of the impact (β can be 1 or higher), and $m_{S/C}$ and $m_{asteroid}$ are the spacecraft and asteroid masses, respectively. To make a conservative assumption of the impact, $\beta = 1$ is assumed. A higher β would be beneficial because it would mean that the impact generates a higher Δv , which, in turn, provide a higher deflection.

The NED method consists of detonating a nuclear explosive near or at the surface of the asteroid to generate a velocity change. A explosive device adds a new level of complexity on the problem as it generates ejecta and depending on the asteroid's composition, can break apart the body, which may result in an insufficient deflection. To avoid this issue, the incident radiation of the NED on the asteroid's surface needs to be constrained, which results in a balance between the size of the explosive and the distance of device detonation. For cases where the deflection is not possible and obliterating the asteroid is the best option, a surface or subsurface detonation can also be considered. That said, asteroid

disruption via standoff detonation is practical for a wide variety of scenarios. A detailed calculation for the size of the nuclear explosive and the distance from the asteroid at which it should be detonated is beyond the scope of this paper. Velocity imparted on the asteroid is dependent on the deterministic escape velocity, as well as its orbital characteristics. The trajectory portion of this study outlines the various velocity changes that each method is required to provide.

2.2. Launcher

Different launchers are considered in this study: The Atlas V family, the Delta IV Heavy, and the SLS Block 2B. Other launchers that will begin operations in the next years would only improve the solution space. The launcher selection has implications on the launch mass and escape velocity. In general, a higher escape velocity results in a better deflection campaign, as the vehicle will be able to reach the asteroid faster and with more velocity (which can be beneficial for a kinetic impactor). Note, higher arrival velocities do not ensure mission success, since the specific energy might exceed 100 J/kg, which is assumed as an upper bound for the asteroid's cohesion energy. The spacecraft launch mass is free to vary as one of the optimizer parameters. However, the arrival mass is constrained to guarantee a viable size for the spacecraft; a minimum bound is placed on the spacecraft arrival mass of 1900 kg for NED and 500 kg for survey and kinetic impactor. The spacecraft mass will vary during the trajectory if propellant is used. When launching kinetic impactors the deliverable mass is launch vehicle limited.

2.3. Trajectory type

Three types of trajectories are considered in the trajectory design: ballistic trajectories, trajectories that utilize Deep Space Maneuvers (DSMs), and low-thrust trajectories. In the ballistic trajectory, the vehicle mass remains constant and the arrival conditions are dictated by the launch - there is no deterministic maneuver during the transfer. In the DSM type, one deterministic chemical maneuver is made during the transfer, and the direction and time of the maneuver is solved for by the optimizer. Low-thrust designs change the trajectory by propulsively accelerating the spacecraft. A solar electric propulsion system is used in this work. This system has its maximum thrust capacity varying with the vehicle's distance from the Sun, number of thrusters and throttle logic. For all trajectory types, Earth Gravity Assists (EGAs) were explored as a way to potentially improve the trajectory solutions.

Combined with the different trajectory types, two arrival types are also considered: flyby and rendezvous. A flyby option, in general, requires less propellant because the spacecraft does not need to match the asteroid's velocity. This is especially useful for targets with high orbital eccentricity and/or inclination. However, an intercept trajectory can only approach the asteroid within a limited relative geometry, which, in turn, limits the ability to control the direction of the velocity change imparted to the asteroid by either a kinetic impactor or nuclear device. Additionally, the maximum asteroid-relative velocity at intercept that can be successfully accommodated is limited by the onboard Guidance, Navigation, and Control (GNC) system; in this study we limit the relative velocity at asteroid intercept to be no higher than 10 km/s. The rendezvous option provides a comfortable condition for the navigation system (low approach speed), but there will generally be fewer opportunities for rendezvous missions, due to the effects of relative orbit geometry and limited onboard propellant. Additionally, a rendezvous mission for asteroid deflection is only practical if a nuclear device is to be employed in standoff mode. Clearly, a kinetic impactor will have little effect on an asteroid if it is not approaching the asteroid at hypervelocity.

Finally, this paper also explores the use of single or multiple spacecraft. Single vehicle missions are less complicated to operate and less costly, but have the added complexity of having a single spacecraft performing both the asteroid reconnaissance and deflection portions of the campaign. That likely requires enhanced spacecraft reliability. Moreover, for the kinetic impactor method, this means that all the deflection has to be delivered by a single spacecraft. On the other hand, a multiple vehicle mission can perform the reconnaissance and deflection mission separately. This may simplify the mission logistics to some extent, while involving greater cost (more launches). However, overall mission campaign reliability may suffer if multiple spacecraft must all operate reliably in order to achieve overall mission campaign success.

3. Deflection Approximation

Traditionally, this type of mission analysis is performed by grid searches. Grid search methods provide a comprehensive understanding of the solution space, but can require long computational times, which can be impractical. This study developed a fast semi-analytical approximation to find the peak deflection points for the mission. The asteroid's trajectory is divided into segments (enough to generate

a trend line) from August 1, 2019 to the Earth impact day July 21, 2027. For each point, a deflection is applied to the asteroid towards and against the velocity vector - these two directions guarantee the maximum energy change in the orbit. A perturbed asteroid trajectory is then propagated with both Keplerian and n-body dynamics for comparison. After propagating the perturbed asteroid, unperturbed asteroid, and Earth, the resulting distance from Earth is measured. Both propagations use a stepped evaluation to find the new close approach point. Moreover, the Keplerian propagation makes use of linear correctors - the detailed description of the correctors is out of the scope of this paper.

The results of the deflection survey is shown in Figure 1. Note that the linear corrector applied to the Keplerian propagation matches the best times for deflection, shown by the peaks, and provides a good estimate of how much the asteroid is deflected away from Earth (at the time of closest approach to Earth). This approximation is fast and reasonably accurate, providing a good understanding of the deflection trade space in a only a few minutes of computational time on a modest laptop computer.

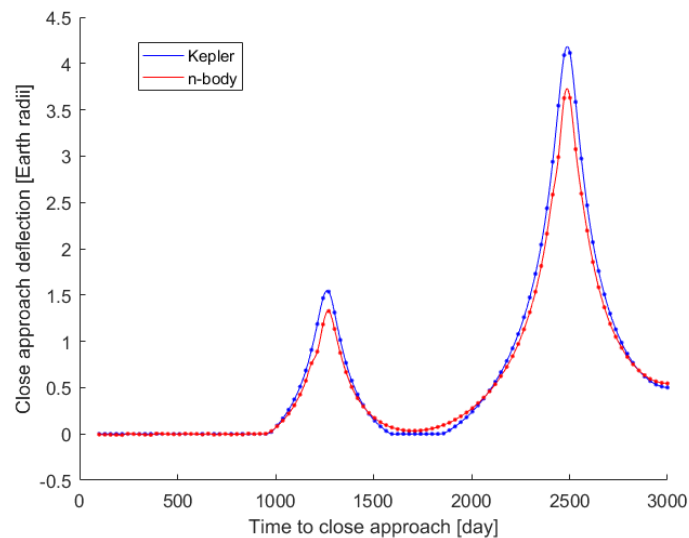


Figure 1: Deflection time survey

Having this information no longer requires a lengthy grid search approach. The peak target deflection points can be approximately determined by this method in Figure 1. However, the timing at which the peak deflection occurs is more significant than the actual deflection value. It should be noted that there may be scenarios in which the deflection cannot be approximated adequately using this Keplerian-with-correctors approach. Deflection peak time ranges identified are used as a initial estimation for the trajectory optimizations that will follow. It is clear that 2017 PDC has a limit number of opportunities for a meaningful deflection. The trajectories will then target a arrival near these dates, which greatly decreases the search space for the optimizer.

4. Mission Analysis

In this section, ballistic, DSM and low-thrust trajectories are designed considering mission constraints. Deflection dates are targeted to the two peaks found in Section 3. It is important to note, that it is not necessary to perform survey and deflection with the same trajectory type different combinations. Ballistic, DSM and low-thrust are possible for a mission concept where the survey and deflection are performed by different spacecrafts.

This mission study is performed for different asteroid sizes and densities. Spacecraft trajectories are analyzed by using asteroid parameters as described in Table 2. Case number one is the biggest and most massive asteroid, and due to this, it yields the upper limit of the deterministic escape velocity. The escape velocity represents the velocity required for a particle, generated by the explosion or impact, to escape the gravitational grip of the asteroid. As presented in the outline, this study considers the total change of the asteroid's velocity to be less than or equal to 10% of the deterministic escape velocity. Cases two to five represent smaller asteroids with a fixed density. The last column of the table shows the maximum velocity change that the NED can possibly impart to the asteroid. Naturally, the higher the mass results in a lower amount of velocity which can be imparted by the device. Note that this maximum

Table 2: Asteroid parameters and NED max deflection velocity change

Case	Asteroid Diameter (meters)	Asteroid Density (g/cm ³)	Asteroid Escape Velocity (cm/s)	Max Δv from NED (cm/s)
1	385	2.6	23.2099	6.123
2	100	1.5	4.579	63.808
3	150	1.5	6.8685	26.918
4	200	1.5	9.158	14.542
5	270	1.5	12.3633	7.623

Table 3: Pre-deflection observation missions (BF: Ballistic Flyby, BR: Ballistic Rendezvous, A401: Atlas V 401, D: Delta IV Heavy)

Mission Type	Launch Vehicle	# of Launches	Launch C3 (km ² /s ²)	Launch Date (MM-DD-YYYY)	Ast. Arrival Date (MM-DD-YYYY)	S/C-Asteroid Arrival ΔV (km/s)	S/C Mass @ Arrival (kg)	Rel. Speed at Asteroid (km/s)	Rend. Propellant mass frac.
BF	D	1	88.459	7/22/2021	10/5/2022		1176	7.866	
BF	A401	1	53.976	6/12/2021	4/13/2023		515	4.903	
BF	D	1	96.739	5/18/2022	8/21/2023		903.7	9.697	
BF	A401	1	53.837	8/26/2022	8/11/2023		519.1	11.91	
BR	D	1	53.976	6/12/2021	4/13/2023	4.903	583.9		0.8113

value will violate the escape velocity; therefore, the NED will have to provide less energy to the system for a deflection. Otherwise, the device will disrupt the asteroid.

4.1. Ballistic solutions

Ballistic case trajectories consider only one maneuver, which takes the spacecraft from its current heliocentric orbit to the point of desired asteroid intercept. Doing so results in trajectories that are not feasible for certain launch vehicles or cannot deliver enough mass with a single spacecraft. When this is the case, multiple launch vehicles are considered, but do not include delay times between launches nor varying trajectories. All spacecraft are assumed to be launched with the same launch vehicle and same trajectory. Effectively, this is a discrete mass multiplier to any given spacecraft mission trajectory.

Along with deflection mission, pre-deflection and post-deflection ballistic observation missions are considered. These missions only consider one spacecraft as well as a minimum final mission mass of 500 kg. For simplicity and cost effectiveness, each observation mission is launched after its previous mission (when considering deflection and post deflection missions) is successful. However, this is not needed when differing missions are not required to be completed before its proceeding mission launches (staggered launches).

Figure 2 outlines the possible trade space for the deflection and observation missions considering the asteroid's discovery time and a three year mark. The mark is intended to give reference to the time that it would take to prepare for such mission. This value may be as low as two years or even zero, if a pre-constructed deflection system has been developed.

4.1.1. Kinetic impactor

Kinetic impactors are used to impart a change in velocity to an intercepted asteroid. This is done by a momentum transfer when collision occurs. Each case provided in Table 2, with corresponding case missions in Table 4, are investigated for plausibility. Many of the mission scenarios considered launch independently, as well as after its preceding mission's completion. This results in deflection missions that must reach the asteroid in a shorter amount of time. Doing so causes the spacecraft's minimum solar distance to be less than 0.7 AU, which is undesirable. A remedy for this situation is to have a staggered launch sequence. Solutions for launches with longer time-of-flights are given for each case in Table 4. When staggered launches are considered, any combination of launches can be completed from Tables 3, 4 and 5.

When an asteroid's mass becomes larger, more spacecraft mass is needed to impart the required asteroid change in velocity. However, as shown in Table 4, some missions require an undesirable amount of kinetic impactors, but meet deflection velocity change, specific energy (< 100 J/kg), and deflection distance requirements. As the asteroids become less dense and smaller in diameter, it can

Table 4: Deflection observation missions (BK: Ballistic Kinetic Impactor, BN: Ballistic NED, A401: Atlas V 401, A421: Atlas V 421, A551: Atlas V 551, D: Delta IV Heavy, S: SLS Block 2B)

Cases	Mission Type	Launch Vehicle	# of Launches	Launch C3 (km ² /s ²)	Launch Date (MM-DD-YYYY)	Ast. Arrival Date (MM-DD-YYYY)	Asteroid ΔV (cm/s)	S/C Mass @ Arrival (kg)	Minimum Solar Distance (AU)	Rel. Speed at Asteroid (km/s)	Impact Specific Energy (J/kg)	Escape Velocity Fraction (DV/Vesc)	Deflection (Earth Radii)
1	BK	D	23	21.719	10/25/2022	2/17/2024	1.835	146051	0.7	9.759	89.52	0.079	1.034
	BK	S	4	21.719	10/25/2022	2/17/2024	1.788	142311	0.7	9.759	87.22	0.077	0.983
	BK	D	20	17.274	11/9/2021	11/9/2023	1.800	138255.5	0.7	10.117	91.07	0.078	0.933
	BK	S	3	17.274	11/9/2021	11/9/2023	1.487	114153.6	0.7	10.117	75.19	0.064	0.609
	BN	S	1	13.837	10/15/2022	2/17/2024	1.857	6118	0.7	4		0.080	1.114
	BN	D	1	13.837	10/15/2022	2/17/2024	1.857	7365	0.7	9.523		0.080	1.114
2	BK	A401	1	21.719	10/25/2022	2/17/2024	1.864	1500	0.7	9.759	90.93	0.407	1.066
	BK	A401	1	17.274	11/9/2021	11/9/2023	1.932	1500	0.7	10.117	97.73	0.422	1.072
	BK	A421	1	30.77	8/11/2023	2/7/2024	1.440	820.5226	0.5	13.784	99.25	0.314	0.529
	BK	A421	1	53.537	9/10/2023	2/17/2024	1.544	948.9111	0.5	12.783	98.71	0.337	0.682
	BK	A551	1	72.5	9/20/2023	2/17/2024	1.506	896.96	0.5	13.188	99.31	0.329	0.577
3	BK	D	1	21.719	10/25/2022	2/17/2024	1.835	4983.266	0.7	9.759	89.52	0.267	1.034
	BK	A551	1	21.719	10/25/2022	2/17/2024	1.470	3993.51	0.7	9.759	71.74	0.214	0.643
	BK	A401	2	17.274	11/9/2021	11/9/2023	1.543	4043.16	0.7	10.117	78.05	0.225	0.666
	BK	A421	1	17.274	11/9/2021	11/9/2023	1.246	3265.35	0.7	10.117	63.04	0.181	0.372
	BK	A551	1	17.274	11/9/2021	11/9/2023	1.669	4373.13	0.7	10.117	84.42	0.243	0.796
	BK	D	1	30.77	8/11/2023	2/7/2024	1.448	2785.14	0.5	13.784	99.82	0.211	0.537
	BK	A551	1	30.77	8/11/2023	2/17/2024	1.448	2785.14	0.5	13.188	99.82	0.211	0.537
	BK	A421	1	30.77	8/11/2023	2/17/2024	1.236	2377.11	0.5	13.188	85.20	0.180	0.334
	BK	A551	3	72.5	9/20/2023	2/17/2024	1.506	3027.23	0.5	13.188	99.31	0.219	0.577
	BK	D	2	72.5	9/20/2023	2/17/2024	1.506	3027.232	0.5	13.188	99.31	0.219	0.577
	BK	D	1	53.537	9/10/2023	2/17/2024	1.509	3128.74	0.5	12.783	96.44	0.220	0.646
	BK	A551	2	53.537	9/10/2023	2/17/2024	1.557	3228.74	0.5	12.783	99.52	0.227	0.695
4	BK	D	2	21.719	10/25/2022	2/17/2024	1.973	12700.09	0.7	9.759	96.25	0.215	1.186
	BK	A401	3	21.719	10/25/2022	2/17/2024	0.835	5377.42	0.7	9.759	40.75	0.091	0.024
	BK	A551	3	21.719	10/25/2022	2/17/2024	1.861	11980.52	0.7	9.759	90.79	0.203	1.063
	BK	A401	6	17.274	11/9/2021	11/9/2023	1.953	12129.48	0.7	10.117	98.79	0.213	1.094
	BK	A401	3	17.274	11/9/2021	11/9/2023	1.953	6064.74	0.7	10.117	49.39	0.107	0.122
	BK	A421	3	17.274	11/9/2021	11/9/2023	1.577	9796.06	0.7	10.117	79.78	0.172	0.701
	BK	A551	2	17.274	11/9/2021	11/9/2023	1.408	8746.26	0.7	10.117	71.23	0.154	0.53
	BK	D	1	17.274	11/9/2021	11/9/2023	1.113	6912.78	0.7	10.117	56.30	0.122	0.245
	BK	D	2	30.77	8/11/2023	2/7/2024	1.440	6562.18	0.5	13.784	99.22	0.157	0.528
	BK	A551	2	30.77	8/11/2023	2/7/2024	1.447	6594.42	0.5	13.784	99.71	0.158	0.535
	BK	A551	6	72.5	9/20/2023	2/17/2024	1.516	7223.73	0.5	13.188	99.31	0.166	0.587
	BK	S	1	72.5	9/20/2023	2/17/2024	1.506	7175.66	0.5	13.188	99.31	0.164	0.577
	BK	A551	2	30.77	8/11/2023	2/7/2024	1.447	6594.42	0.5	13.784	99.71	0.158	0.535
	BK	D	3	53.537	9/10/2023	2/17/2024	1.544	7591.29	0.5	12.783	98.71	0.169	0.682
	BK	A551	4	53.537	9/10/2023	2/17/2024	1.557	7650.86	0.5	12.783	99.49	0.170	0.695
5	BK	D	5	21.719	10/25/2022	2/17/2024	1.835	29062.4	0.7	9.759	89.52	0.148	1.034
	BK	D	4	21.719	10/25/2022	2/17/2024	1.203	19050.13	0.7	9.759	58.68	0.097	0.368
	BK	S	1	21.719	10/25/2022	2/17/2024	1.788	28318.19	0.7	9.759	87.22	0.145	0.983
	BK	D	4	17.274	11/9/2021	11/9/2023	1.810	27651.1	0.7	10.117	91.53	0.146	0.942
	BK	D	3	17.274	11/9/2021	11/9/2023	1.357	20738.33	0.7	10.117	68.65	0.110	0.48
	BK	D	3	30.77	8/11/2023	2/7/2024	1.407	15776.37	0.5	13.784	96.95	0.114	0.497
	BK	S	1	72.5	9/20/2023	2/17/2024	1.261	14787.2	0.5	13.188	83.18	0.102	0.345
	BK	D	6	53.537	9/10/2023	2/17/2024	1.544	18677.42	0.5	12.783	98.71	0.125	0.682

Table 5: Post-deflection observation missions (BF: Ballistic Flyby, A401: Atlas V 401, D: Delta IV Heavy)

Mission Type	Launch Vehicle	# of Launches	Launch C3 (km ² /s ²)	Launch Date (MM-DD-YYYY)	Ast. Arrival Date (MM-DD-YYYY)	S/C Mass @ Arrival (kg)	Rel. Speed at Asteroid (km/s)	Phase Angle at Asteroid Arr. (deg)	Years until Earth Encounter (Days)
BF	D	1	81.043	5/17/2024	5/27/2026	1473	2.992	39.15	420
BF	A401	1	52.168	6/6/2024	10/4/2026	568	2.794	42.73	290
BF	D	1	53.804	12/23/2026	5/12/2027	3108	9.604	54.52	70
BF	D	1	53.352	7/31/2025	8/25/2026	3143	11.825	3.3	330
BF	D	1	85.956	8/20/2025	8/25/2026	1270	9.806	6.87	290
BF	D	1	65.492	7/21/2025	1/22/2027	2292	6.222	2.96	180

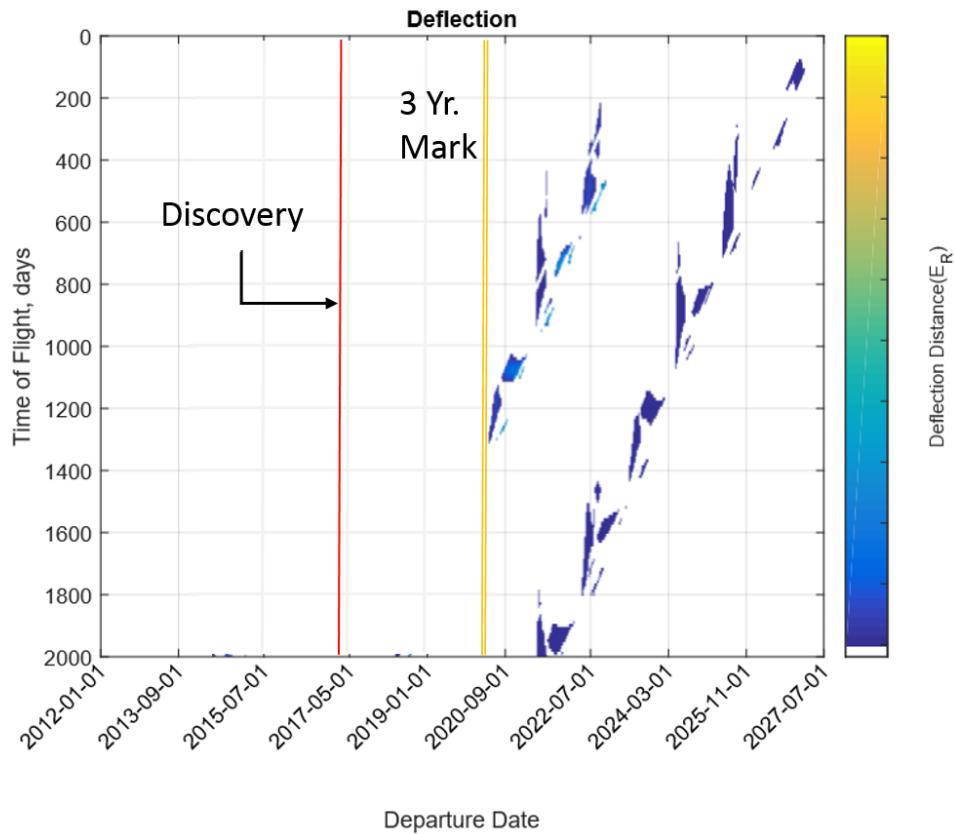


Figure 2: Ballistic case example of trade space

be seen that the escape velocity fraction requirement is not met. In contrast, if the imparted change in velocity is reduced to include the escape velocity requirement, the deflection from Earth is reduced drastically.

4.1.2. Nuclear Explosive Device (NED)

Another deflection option that is considered is delivering a nuclear explosive device. A NED is more mass efficient than a kinetic impactor. However, much like kinetic impactor scenarios, the change in velocity imparted to the asteroid cannot exceed ten percent of the asteroid's deterministic escape velocity. Since a NED's yield is determined before hand, the stand off distance for detonation can be varied to impart the desired asteroid velocity change.

Spacecraft relative speed to the asteroid is a major concern when conducting a mission. If the speed is too large, the error in detonation distance might be insufficient to ensure deflection, or the device might not detonate before impact. Figure 3 illustrates the timing needed for a given detonation distance accuracy for specified relative speeds. As it can be seen, slower velocities are more desirable.

When the spacecraft is approaching the asteroid at a speed that exceeds a certain threshold, the spacecraft may slowdown. This however, requires the use of propellant mass. Figure 4 depicts the fraction of mass after maneuver completion compared to that of mass prior to the maneuver. For the spacecraft to slowdown, sufficient mass, which is not lower than 1900 kg, must remain. This ensures that there is enough structure for housing the nuclear device as well as other instruments and operational implements.

Each deflection mission in Table 4 may be converted into a NED mission. However, the final mass for each spacecraft must be greater than 1900 kg, and the NED device's max change in velocity cannot exceed the required kinetic impactor velocity change. New missions can be determined using Figures 3 and 4 as well as Table 2 and 4. This information will help determine: if the device can deliver the needed change in velocity, the end mass after a burn, and the asteroid's deflection from Earth. However, as was stated in the previous section, smaller objects potentially cannot withstand change in velocities that yield a sufficient deflection from Earth. When this is the case, asteroid obliteration is the only option.

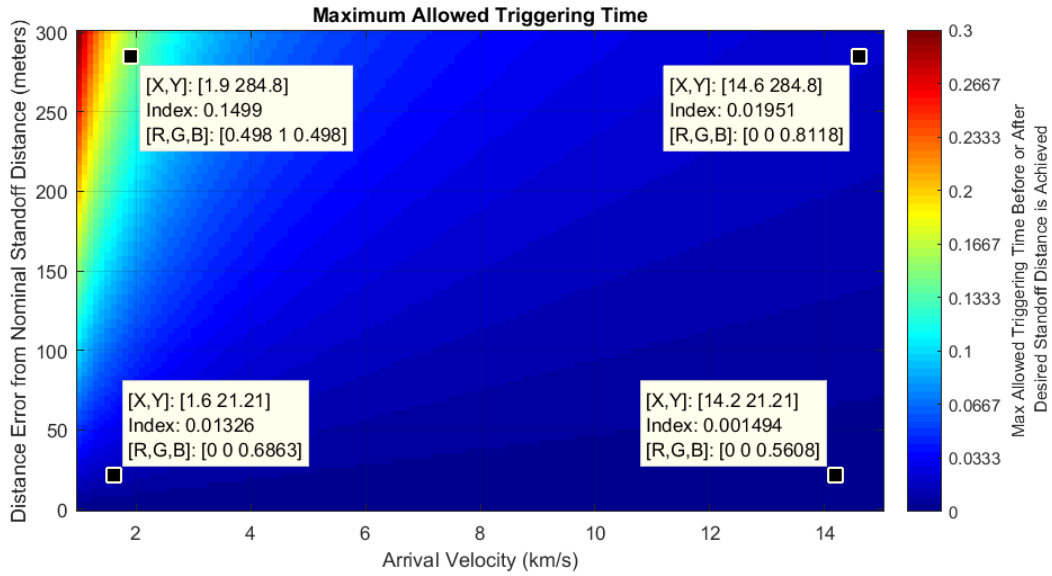


Figure 3: Detonation timing required for specific S/C-asteroid relative speed for a desired detonation distance error

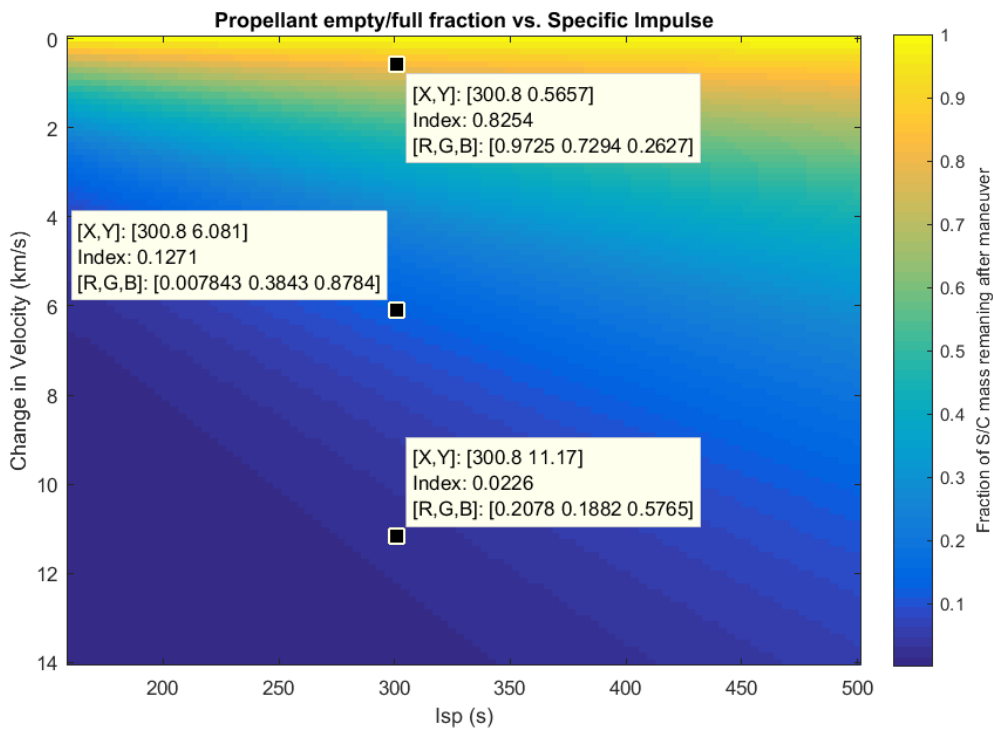


Figure 4: Percent of S/C mass remaining when a decrease in relative velocity is required

4.2. DSM solutions

DSM solutions are designed utilizing a Non-Linear Programming Method combined with Multiple Basin Hopping. Deflection dates are bounded by the two peaks found in Section 3. The objective of the DSM is to change the ballistic trajectory with one impulse of $I_{sp} = 200$ s changing the vehicle's trajectory and providing a better deflection than a pure ballistic solution. Due to the large number of spacecraft needed for an successful kinetic impactor mission, only the NED concept will be used here. With this it is possible to realize the mission with a single spacecraft performing the survey and deflection, or two spacecraft (one surveying the asteroid prior to launching and a second one to perform the deflection).

With a single spacecraft concept, the same vehicle has to perform both survey and deflection. Figure 5 shows a solution for flyby. The DSM is not able to rendezvous since only one maneuver is explored. Note, the flyby option can only deflect the asteroid at the second peak, the spacecraft cannot reach the asteroid for the second time at the first peak. Moreover, it requires a large launch mass and C_3 that can only be provided by the Delta IV Heavy launcher - the Atlas V launcher is incapable of reaching the escape conditions. However, as it will be seen further, the low-thrust propulsion will enable double flyby missions with Atlas V, as well. Rendezvous options cannot be obtained with this trajectory type. Changes in the trajectory have to be greater than a chemical impulse can provide. As it will be seen in the next section, low-thrust propulsion can provide the necessary conditions to rendezvous with 2017 PDC.

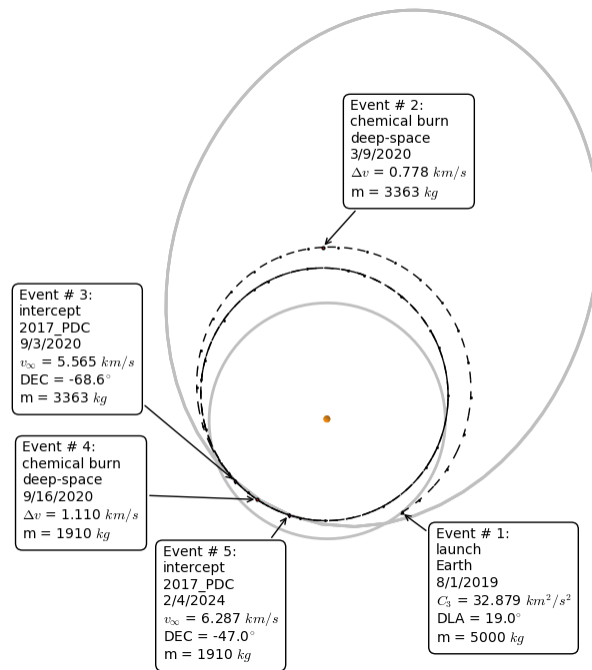


Figure 5: DSM single spacecraft with two flybys launched by Delta IV Heavy

With a double spacecraft concept, the first vehicle needs to survey the asteroid before the second spacecraft is launched. This option requires only flyby cases to be explored since rendezvous solutions only require one vehicle. Figures 6 and 7 show, respectively, solutions for the survey and flyby deflection. Both solutions show a valid mission profile with the deflection happening at the second peak.

4.3. Low-thrust solutions

Low thrust solutions are designed utilizing a Non-Linear Programming Method combined with Multiple Basing Hopping. The objective of the low-thrust trajectory is to allow the spacecraft to steer the asteroid as far as possible from the Earth on the impact date. As discussed in the ballistic results, the kinetic impactor concept is not applicable for particular asteroid sizes; therefore, only the NED concept will be used hereafter.

The spacecraft's propulsion system is composed of a two NEXT TT11 high-thrust engine, with a throttle logic mode of minimum number of thrusters on. A duty cycle of 90% is used to leave margin for missed thrust and trajectory corrections due to secondary order effects. Considering a 1 A.U. basic power level, the spacecraft bus consumes 0.8 kW, and its power system is set to 20 kW. An initial mass of the spacecraft is allowed to vary based on the launcher with its minimum delivered mass set to 1900 kg to account for the NED, subsystems and propellant.

With a single spacecraft concept, the same vehicle has to perform both survey and deflection. Figures 8 and 9 show solutions for flyby and rendezvous. Note, the flyby option can only deflect the asteroid after the second peak, where the deflection is too small to avoid an Earth impact. Utilizing the rendezvous option, the spacecraft arrives after the first peak, but before the second peak. This allows a comfortable time for the survey campaign and a precise detonation once time reaches the second peak.

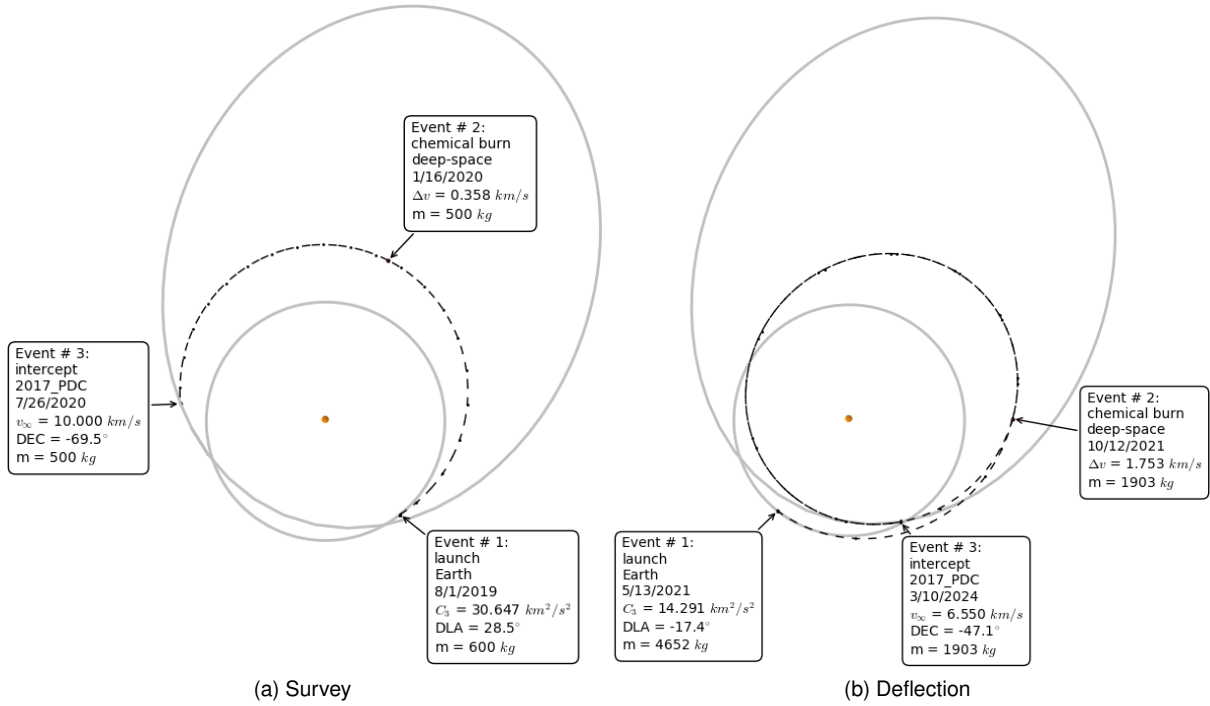


Figure 6: DSM double spacecraft mission with Atlas V

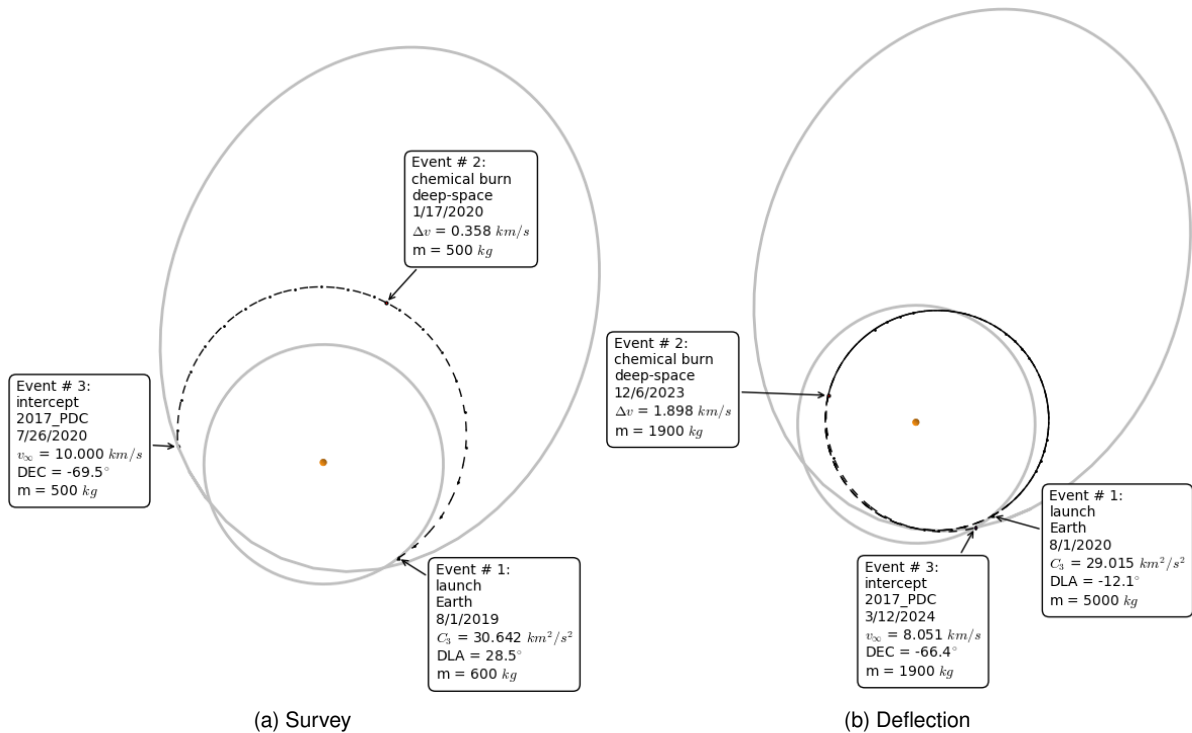


Figure 7: DSM double spacecraft mission with Delta IV

With a double spacecraft concept, the first vehicle needs to survey the asteroid before the second spacecraft is launched. This option requires only flyby cases to be explored since rendezvous solutions only require one vehicle. Figures 10 and 11 show, respectively, solutions for the survey and flyby deflection. Both solutions show a valid mission profile with the deflection happening at the second peak.

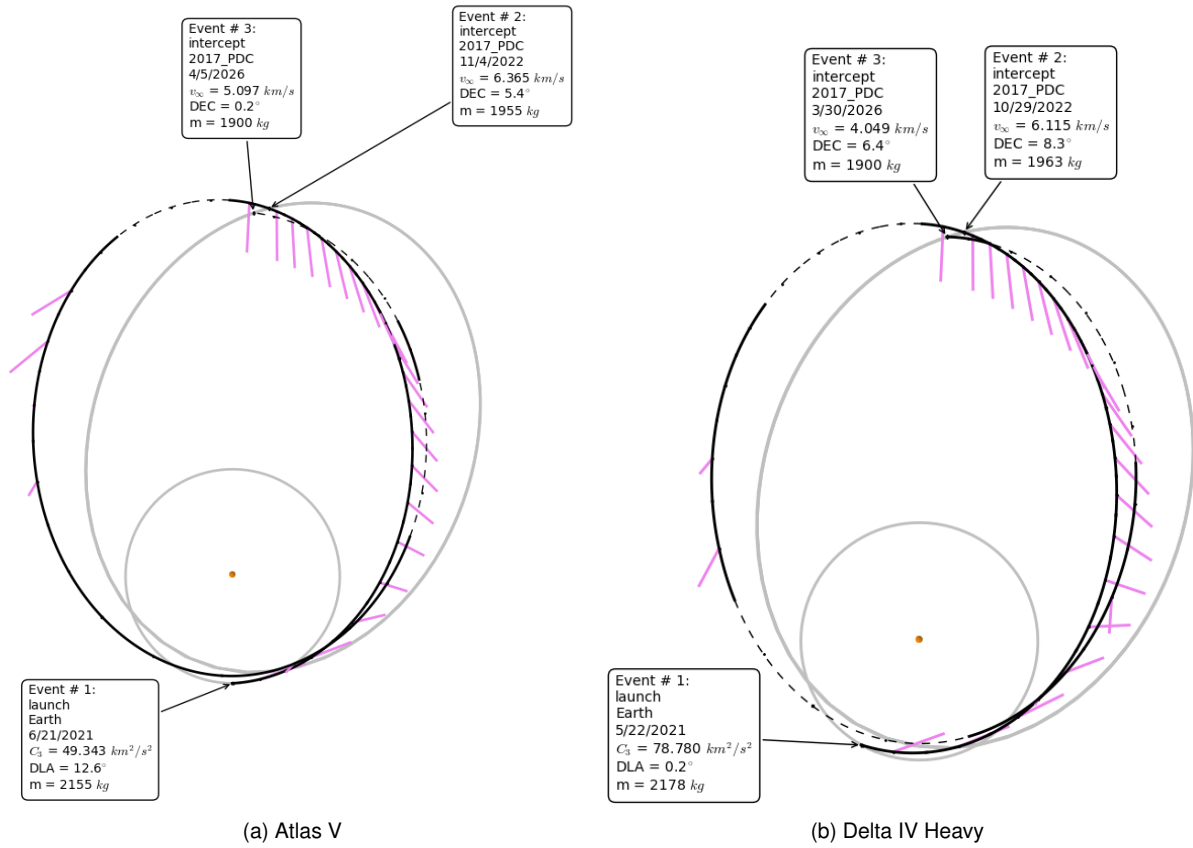


Figure 8: Low-thrust single spacecraft with two flybys

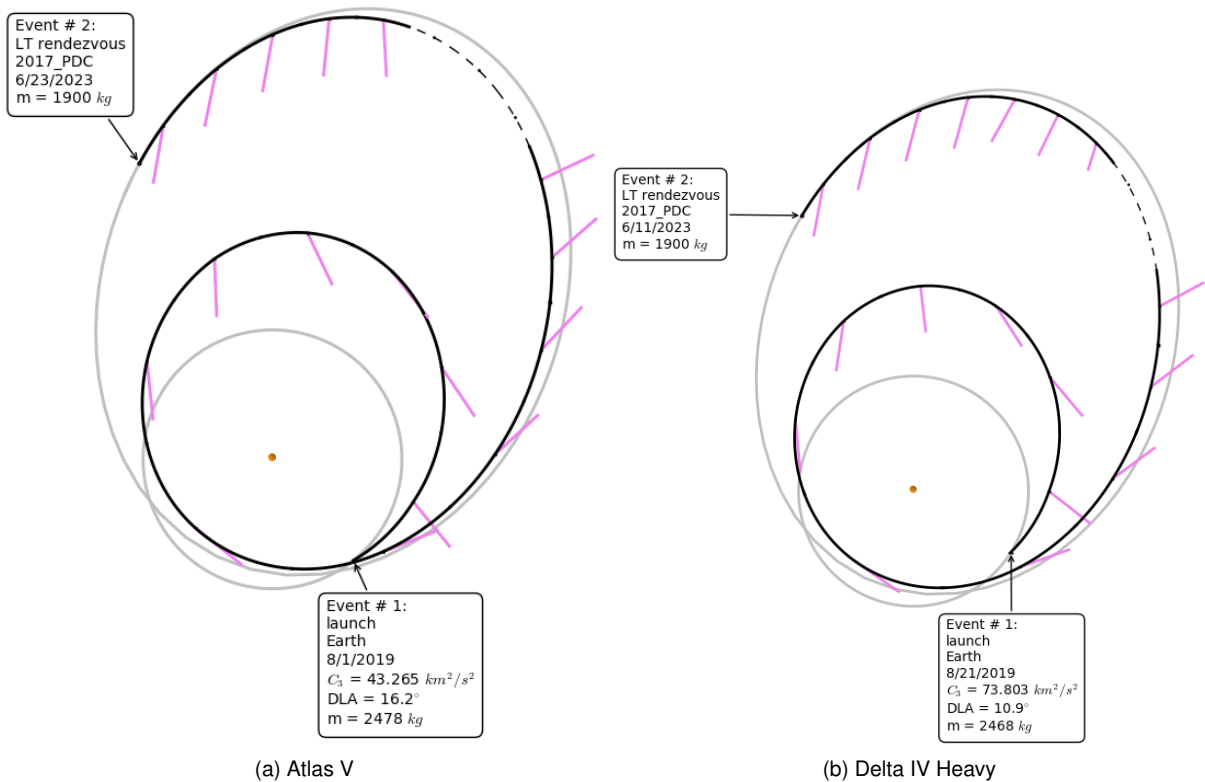


Figure 9: Low-thrust single spacecraft with rendezvous

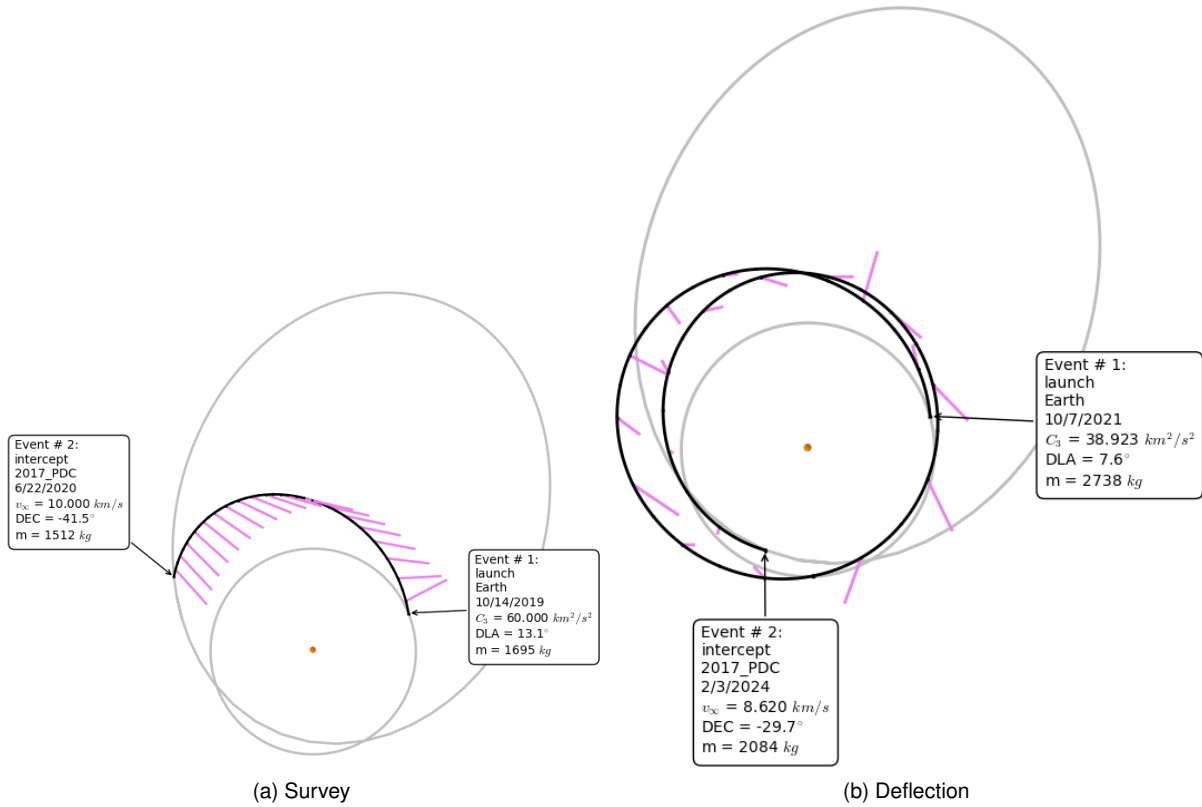


Figure 10: Low-thrust double spacecraft mission with Atlas V

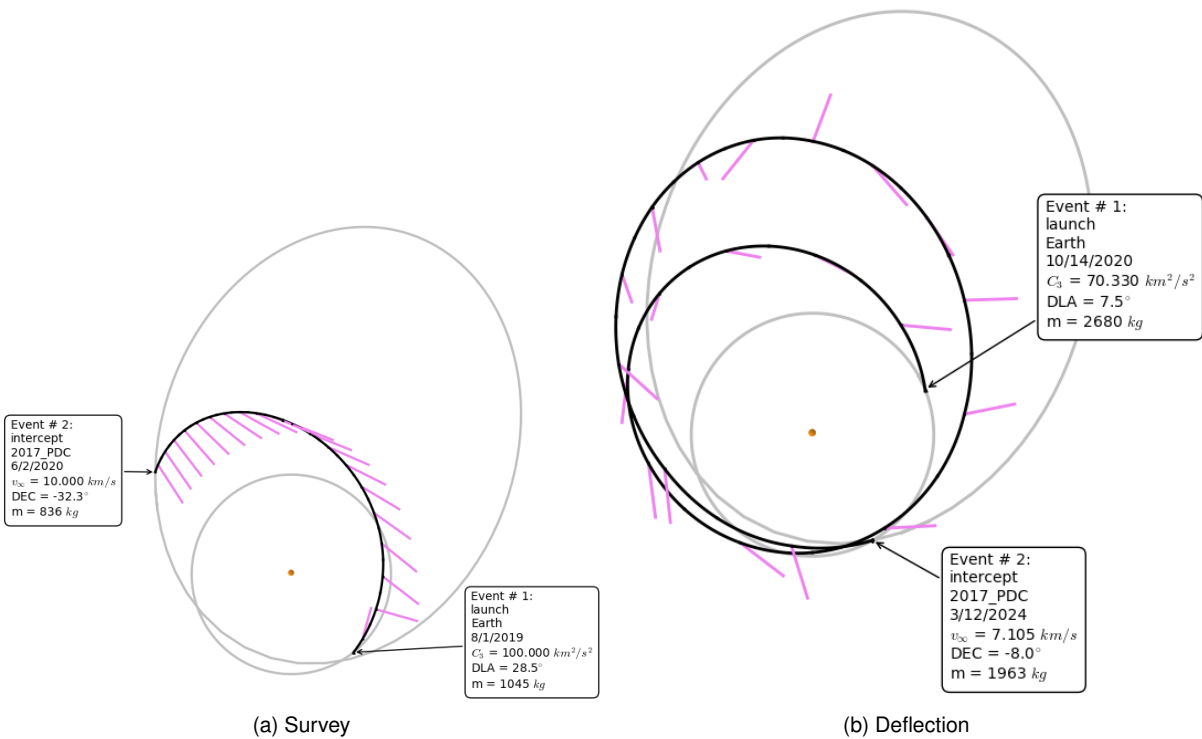


Figure 11: Low-thrust double spacecraft mission with Delta IV

4.4. Disruption

As stated previously, smaller asteroids cannot be effectively deflected, because the energy imparted would result in a deflection velocity that exceeds the deterministic escape velocity. For such cases,

disruption is the only remaining option. Disruption is the attempt to vaporize the asteroid or part of it, such that any remaining material would be vigorously scatter and not impact Earth. If it does, the pieces are small enough to burn during the atmospheric entry. Decision to use this option will, most likely, have political consequences. Therefore, authorization for the mission execution will take time. The objective function for such missions were set as the latest possible launch date with the latest arrival date bounded by the final day that it was possible to noticed a small deflection for case 3 as can be seen in Figure 2 (latest value of 290 days before nominal asteroid-Earth impact). This value guarantees that even if small pieces remain, the imparted Δv will be sufficient to avoid collision with the planet. A rendezvous encounter is selected as the most conservative since disruption cases need a highly precise guidance that can only be achieved with this type of encounter.

Figure 12 shows the low-thrust rendezvous solutions. As discussed previously, DSM trajectories cannot reach rendezvous solutions and therefore are not considered here. Note, both launchers combined with the propulsion system are capable to deliver the spacecraft on the same dates.

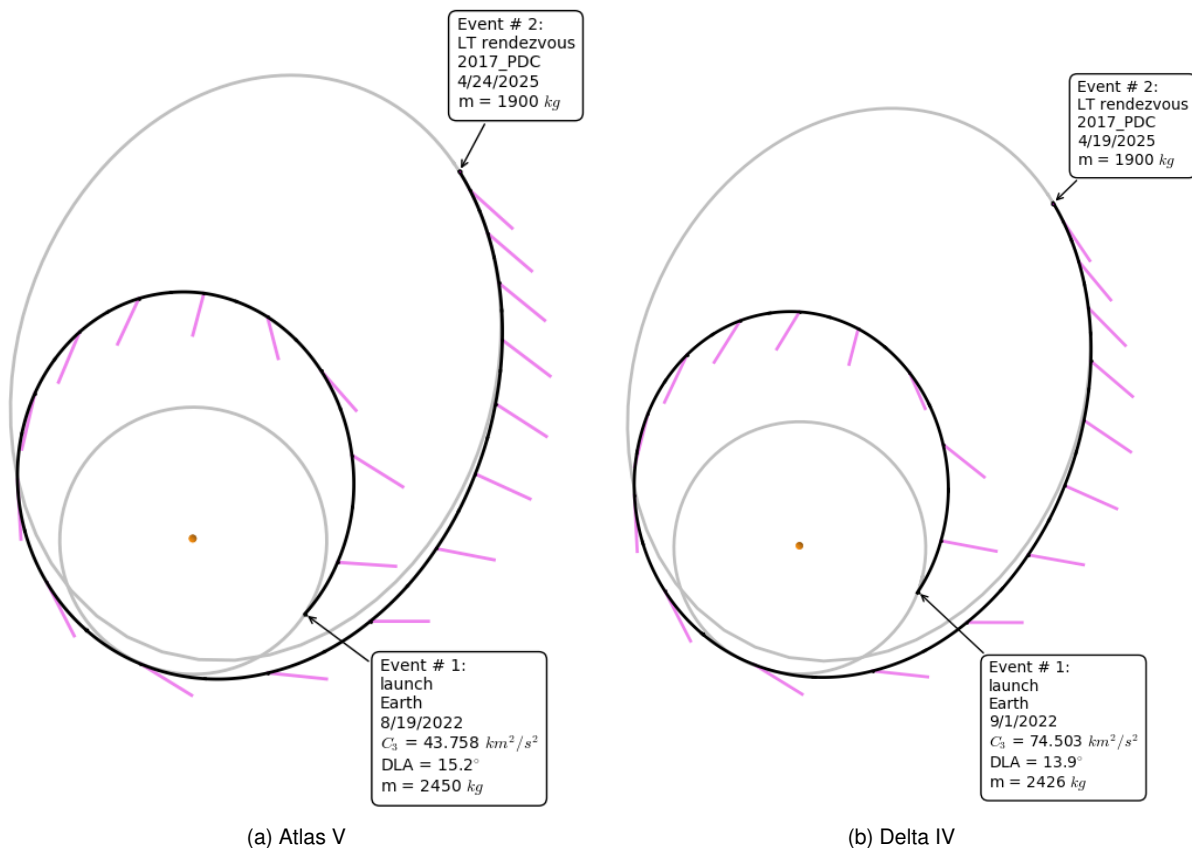


Figure 12: Low-thrust double spacecraft mission with Atlas V

It is important to note, by utilizing the disruption option, it is not possible to guarantee that a sizable piece of the asteroid will remain or even if the asteroid is going to be fragmented. It is not known how the energy will be transferred through the body since its exact internal composition is unknown. Arriving at hyper velocity speeds, the spacecraft will be vaporized on impact and the fragmentation cannot be determined without assuming a hydrodynamic-like behavior of the system with a fragmentation model [1].

5. Conclusion

Hypothetical asteroid 2017 PDC was designed as a realistic problem for planetary defense studies. The core objective is to mitigate the 2017 PDC impact with Earth from its nominal date of July 21, 2027. This paper utilized a trade-off study of different mission concepts focusing in real available hardware. Deflection strategies focused on nuclear explosive device and kinetic impactor, applied in both deflection and disruption strategies. Ideally, the deflection would be first choice. In some cases, however, the

combination of orbit, timing and asteroid size are incapable of generating a significant change in the impact or risk breaking the body. In these scenarios disruption was selected as the only alternative.

This paper further utilized a novel approximation method to find peak deflection points over the asteroid lifetime. It also directly optimizes the trajectory that generates the best deflection from Earth as an objective function.

The trade study showed that different mission scenarios can be implemented to successfully deflect the biggest and most massive asteroid using different launchers, combined with a spacecraft with reasonable mass. All three trajectory methods explored (ballistic, deep space maneuver and low-thrust) can reach successful missions. Solutions for these cases range from a deflection of 1.5 to 3.6 Earth radii. Smaller asteroids can be deflected with less energy, however, this does not guarantee that the deflection will not disrupt the body nor insure a deflection from Earth. In such cases, disruption would be the most secure way to neutralize the threat.

Acknowledgments

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- [1] Zimmerman, B., and Wie, B., A GPU-Accelerated Multiphase Computational Tool for Asteroid Fragmentation/Pulverization Simulation, AAS 16-242, AAS/AIAA Space Flight Mechanics Meeting, Napa, CA, February 14-18, 2016.