

PLANETARY DEFENSE MISSION APPLICATIONS OF HEAVY-LIFT LAUNCH VEHICLES

George Vardaxis* and Bong Wie†

This paper expands the previously established capabilities of the Asteroid Mission Design Software Tool (AMiDST) to include launch vehicles currently under development by SpaceX and NASA, in addition to the Delta II, Delta IV, and Atlas V class launch vehicles, for its planetary defense mission applications. A fictional asteroid, designated 2015 PDC, is used as a reference target asteroid to further demonstrate the effectiveness and applicability of the AMiDST for planetary defense mission design and planning. During the 2015 IAA Planetary Defense Conference, the asteroid 2015 PDC was used for an exercise where participants simulated the decision-making process for developing deflection and civil defense responses to a hypothetical asteroid threat. The planetary defense missions considered in this paper are primarily focused on short-warning time scenarios (90 days, 60 days, and 30 days) where a very large (5,000 to 10,000 kg) space system would be launched using heavy-lift launch vehicles such as Delta IV Heavy, Falcon Heavy, or the SLS, to intercept and disrupt the oncoming target asteroid.

INTRODUCTION

There is a very real and ever-present threat that Earth faces every day from asteroids. Most of the impacts are from objects that are too small to do any damage on the surface due to the fact that they burn up in the atmosphere. However, there are the rare occurrences where a large enough object encounters the Earth and causes some serious damage. The number of identified potentially hazardous near-Earth asteroids (NEAs) is increasing, and so too is the likelihood of one of those asteroids posing a non-negligible threat to the planet. The human race is in a unique position to do something about those threats, to either mitigate or eliminate them.

Dealing with the threat of any near-Earth object (NEO) has three main steps: detection, characterization, and mitigation. NASA, as well as other organizations, have put a lot of effort into the detection/tracking of all known near-Earth objects, both threatening and non-threatening. Over the last 20 years there has been a lot of research conducted at various levels on the effectiveness of NEO deflection/disruption strategies. These strategies include kinetic impactors, gravity tractors, and nuclear explosive devices [1–6]. The non-nuclear technique options, for NEO deflection missions, will require lead times of more than 10 years to be effective, even for small asteroids. The amount of the imparted velocity perturbation to the target needs to be about 1-2 cm/s when the lead time is more than a decade, but due to uncertainties and constraints in the detection and tracking of the potentially hazardous asteroids, the warning time is normally much less than that. At the Asteroid Deflection Research Center (ADRC) of Iowa State University, there has been a lot of research

*Corresponding Author, Post-Doctoral Research Associate, Asteroid Deflection Research Center, Department of Aerospace Engineering, Iowa State University, Ames, IA 50011, vardaxis@iastate.edu.

†Vance Coffman Endowed Chair Professor, Iowa State University, bongwie@iastate.edu.

work done on mitigation methods regarding NEOs with short warning times (<5 years) by studying potential mission designs to disrupt hazardous asteroids.

The challenge recently has shifted to how to mitigate the threat of a NEO on an Earth-impacting trajectory in a timely and reliable manner when the warning time is short (< 5 years). For small NEOs impacting fairly unpopulated regions, the best course of action may just be to evacuate the region. However, for larger asteroids or impacts in significantly populated regions, the threat may be best mitigated by disrupting (destroying or sufficiently fragmenting) the body, or by perturbing the trajectory of the body so that it would impact a different location or missing the Earth entirely. But, when the mission lead time is short, the velocity change needed to be delivered to the asteroid becomes rather large. So, for the best mission scenarios with short warning times (less than 5 years), the use of high-energy nuclear explosives in space may be the only option [1]. To this point in time however, there is no consensus on how to safely and reliably mitigate the impact threat of hazardous NEOs with short mission lead times.

As concluded by the 2010 NRC report [1], when the warning time is short, asteroid disruption may be the only feasible strategy for threat mitigation. There is plenty of apprehension regarding the nuclear disruption approach, but it can become an effective mitigation method if most of the resulting asteroid fragments disperse at speeds greater than that of the local escape speed of the asteroid, so that only a small fraction of the fragments end up impacting the Earth. However, non-nuclear techniques would be preferred, even for disrupting hazardous asteroids.

Because nuclear energy densities are nearly $1\text{E}+6$ times higher than those possible with chemical bonds, a nuclear explosive device is the most mass-efficient means for storing energy with today's technology. So, barring the use of a 2000 kg spacecraft equipped with a 300 kg nuclear explosive device (NED), for example, to conduct a disruption mission to a potential hazardous asteroid, a large kinetic impactor spacecraft must be used in order to attempt a similar type of disruption mission. Therefore, this paper is concerned with the disruption mission trajectory designs for a multiple kinetic-energy impactor vehicle (MKIV) system with its mass range of 5,000 to 12,000 kg [7], in application to asteroid 2015 PDC [8].

An in-house design tool, named as the Asteroid Mission Design Software Tool (AMiDST), is used to conduct the mission and trajectory design for near-Earth asteroid missions [9]. The unique trajectory of asteroid 2015 PDC presents the opportunity to show how difficult it can be to design missions to these types of hazardous NEOs. Throughout this paper, the infeasibility of a rendezvous mission will be depicted, and then efforts will be focused on the necessity of HLLVs for short-duration mission designs.

ASTEROID MISSION DESIGN SOFTWARE TOOL (AMIDST)

In this section a brief discussion will be provided regarding the major portions of the AMiDST program: launch vehicle selection, mission cost estimation, and trajectory optimization.

The mission design studies conducted using this tool considered five classes of launch vehicle: i) Delta II, ii) Delta IV, iii) Atlas V, iv) Falcon, and v) SLS. In these mission designs the total mass of the MKIV system is set to be in the range of 5,000 to 12,000 kg for all disruption missions. Given the amount of mass that would be going into orbit, it is fairly obvious that only the larger launch vehicles would be capable of placing the spacecraft into any potential interplanetary orbit. Therefore, only the largest U.S. launch vehicles of the entire collection will be considered: Atlas V 551, Delta IV Heavy, Falcon Heavy, and three variations of the SLS.

Mission cost estimation to design and fabricate the missions is an important task necessary for an early assessment of mission viability and feasibility. The final total cost of each mission is given as a combination of the cost for the launch vehicle, the spacecraft, and estimated mission operations. A cost estimation algorithm was developed to determine the costs associated with constructing the spacecraft, based on a number of previous spacecraft missions with similar goals and parameters.

In regards to a reference 10,000 kg MKIV system that will be used for all the disruption missions case studies considered in this paper, the cost estimate for the mission will not be a part of the discussion due to the sheer mass of the spacecraft itself. A 10,000 kg MKIV system is far outside the range of the spacecraft used to create the polynomial fit, and would require a fair degree of extrapolation in order to assess the cost to construct such a spacecraft, and would most likely be inaccurate. Therefore, the cost of the spacecraft and the mission as a whole will not be taken into account for these conducted mission design scenarios.

Within the AMiDST program, the user is prompted with a decision between three mission types: a direct intercept, a direct intercept at a relative speed of 10 km/s, or rendezvous. Regardless of the decision, the user then enters mission parameter data related to the dates in which they would like the mission to be conducted between, as well as the upper and lower bound mission duration - being the amount of time the spacecraft can take traveling from Earth to the target. Given only this information, the AMiDST constructs a porkchop plot (graph of launch date and mission duration evaluated in terms of total mission ΔV) over the defined time period. The porkchop plot is filled by taking every possible pair of mission duration and launch duration and finding the corresponding solutions, and the entire grid space is analyzed using a customizable cost function, to find the optimal mission trajectory defined by the appropriate state variables which encompass parameters at the beginning and end of the mission trajectory. The potential design parameters are

$$\mathbf{X} = [JD, \Delta V, C3, dispersion, duration, v_{arr}, \alpha_{arr}, \alpha_{LOS}, \alpha_{Sun}] \quad (1)$$

where JD represents the Julian date at mission departure, ΔV is the total mission change in velocity required, $C3$ is the associated mission hyperbolic excess energy at departure from Earth's sphere of influence, $dispersion$ is the dispersion time after disruption, $duration$ is the mission duration, v_{arr} is the relative arrival velocity between the asteroid and the spacecraft, α_{arr} is the relative arrival angle between the asteroid and spacecraft at intercept, α_{LOS} is the line-of-sight angle between Earth and the asteroid at the time of intercept, and α_{Sun} is the approach angle of the spacecraft to the asteroid with respect to the Sun. The cost function is constructed in the following manner

$$C(\mathbf{X}) = f(\mathbf{X}) + \sum g(\mathbf{X}) \quad (2)$$

where $f(\mathbf{X})$ is a constant cost attributed to every mission trajectory and $g(\mathbf{X})$ is a variable cost that is dependent on the state variables used within the trajectory optimization [13]. The components of the overall cost function $f(\mathbf{X})$ and $g(\mathbf{X})$ take the form

$$f(\mathbf{X}) = \Delta V + \sqrt{C3} \quad (3)$$

and

$$g(\mathbf{X}) = g(JD) + g(dispersion) + g(duration) + g(v_{arr}) + g(\alpha_{arr}) + g(\alpha_{LOS}) + g(\alpha_{Sun}) \quad (4)$$

The component cost functions are defined based on the user defined upper and lower bounds for the state variables used in the cost function evaluation of the mission design space. With all missions

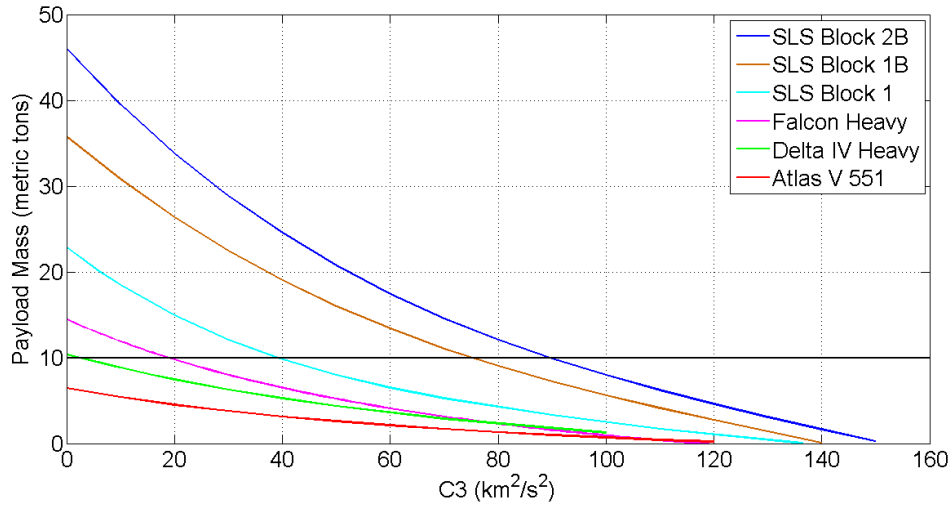


Figure 1. Interplanetary capabilities of Falcon Heavy and SLS launch vehicles.

evaluated, the mission design options are sorted based on the cost function scores, where the best missions have the lowest cost function values due to the lack of penalty accumulation because they are closer to the desired mission parameter values set by the user. Finally, the top 10 missions are presented to the user for the final decision of the desired mission design, where the AMiDST will accept the selection and create an executable m-file for the user to run and be given all the desired data for the chosen mission design.

HEAVY-LIFT LAUNCH VEHICLE (HLLV)

A large launch vehicle capable of lifting between 20,000 to 50,000 kg to low-Earth orbit is referred to as a Heavy-Lift Launch Vehicle (HLLV). Thus, Delta IV Heavy, Ariane V, and Proton-M are such HLLVs currently in service. This section provides a system-level overview of the HLLVs of the United States, including Falcon Heavy and the Space Launch System (SLS). The Falcon Heavy that is being developed by SpaceX, and the SLS Block I, SLS Block IB, and SLS Block 2B that are currently being developed by NASA. Given the information known to this point about these launch vehicles, there is a certain understanding of their predicted launch capabilities (insert citations here). In Figure 1, the interplanetary capabilities of the Atlas V 551, Delta IV Heavy, Falcon Heavy, and three variations of the SLS are shown. Looking at the figure, it can be seen that if all these launch vehicles were available for NEO missions, using a 10,000 kg MKIV system, the Atlas V 551 would be incapable of lifting the spacecraft into any interplanetary trajectory. The Delta IV Heavy would be capable of lifting the MKIV system to orbits that would have a C3 of up to about 5 km²/s². Based on the curves, the Falcon Heavy could outperform the Delta IV Heavy, lifting a 10,000 kg to C3 orbits up to about 18 km²/s². The launch vehicles with the most versatility are the SLS Block 1, Block 1B, and Block 2B configurations capable of lifting a 10,000 kg spacecraft to C3 orbits of about 40 km²/s², 75 km²/s², and 90 km²/s², respectively.

Delta IV Heavy

The Atlas V and Delta IV class launch vehicles are the most powerful classes of launch vehicles currently available (by the United States) for space missions, where the Atlas V 551 and Delta IV

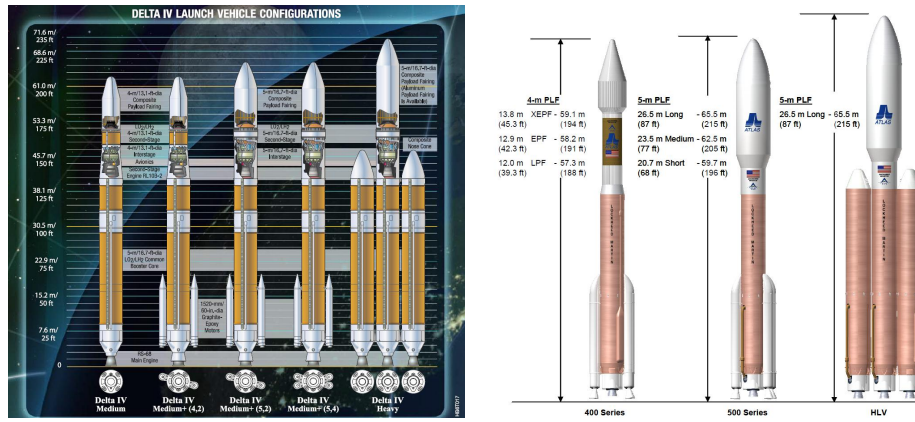


Figure 2. Delta IV (left) and Atlas V (right) launch vehicles [15,16].

Heavy are the largest launch vehicles of their respective classes. Unlike their two-stage Delta II counterparts, the Atlas V and Delta IV class launch vehicles are capable of not only taking large payloads to LEO but directly injecting them into interplanetary trajectories. The Atlas V 551 is not an HLLV; however, it is discussed in this paper as a point of reference.

Falcon Heavy

The Falcon Heavy is projected to launch later this year, and is said to be the most powerful rocket in the world at the time of its operation. The selling point for the Falcon Heavy is that it will be capable to lift over 53 metric tons into orbit, more than twice the payload of the Delta IV Heavy, at one-third the cost. Missions using the Falcon Heavy will deliver large payloads to orbit inside a composite fairing, but will be capable of carrying the Dragon spacecraft. Drawing up the proven heritage and reliability of the Falcon 9, the second-stage Merlin engine (identical to its counterpart on the Falcon 9) delivers the payload to orbit after main engine cut off and first-stage cores separate. The second stage engine is capable of restarted multiple times in order to place payloads into a variety of orbits, including low-Earth orbit (LEO), geosynchronous transfer orbit (GTO), and geosynchronous orbit (GSO). Made up of a single engine, the Falcon Heavy second stage is capable of producing 801 kilo-newtons of thrust in a vacuum, and has a burn time of 375 seconds. The Falcon Heavy's first stage is made up of three cores. The side cores (boosters) are connected at the top and base of the center core's liquid oxygen tank. Each of the Falcon Heavy's side cores (boosters) is equivalent to the first stage of a Falcon 9 rocket with 9 Merlin engines. With a total of 27 Merlin engines, the first stage is capable of generating 17,615 kilo-newtons of thrust at liftoff. Not long after liftoff, the center core engines of the first stage are throttled down, until after the side cores separate, at which time they are throttled back up to full thrust.

For missions that have exceptionally heavy payloads ($> 45,000$ kg), the Falcon Heavy offers a unique cross-feed propellant system that feeds propellant from the side cores to the center core. This enables the center core to retain a significant amount of fuel after the boosters separate [17,18]. Originally designed to carry humans into space and to fly missions with crew to the Moon or Mars, but could be used to carry a large spacecraft into orbit to meet a potentially hazardous NEO.

Space Launch System (SLS)

The design of the SLS serves to accommodate greater mass/volume to orbit, shorter transit times to destination, larger interplanetary science payloads, and enhanced reliability and safety for a variety of different missions. It is projected that the SLS Block 1 design will have the capability to carry up to five times greater mass to orbit than the Delta, Atlas, and Falcon launch vehicles. With the ability to launch such large payload masses, the SLS increases payload mass margins and offers greater propellant loads. It can also accommodate a range of fairing sizes including the existing five meter diameter size, as well as new 8.4 - 10 meter diameter fairings, and will have the capability to support up to six times greater payload volume over current launch vehicles.

Based on the currently accepted launch capabilities of the SLS, shorter mission durations are also possible to various mission destinations. Taking the Europa Clipper mission, the flight time could be reduced by 70% through the use of the SLS rather than the Atlas V 551. Launching into a C3 orbit of $15 \text{ km}^2/\text{s}^2$ and requiring three planetary flybys (Venus-Earth-Earth) before arriving at Jupiter 6.4 years later, the same mission launched with the SLS would launch directly into a C3 orbit of $82 \text{ km}^2/\text{s}^2$, would not require any planetary flybys, and would arrive at Jupiter in 1.9 years. The capabilities of the SLS would allow for longer launch windows and provide more mission margin, in addition to significantly reduced cost for each year of transit reduced. Larger interplanetary science payloads enable three to four times the mass to destination and single launch of larger payload reduces payload complexity. The SLS launch vehicle range allows for missions previously deemed very difficult or infeasible to be reconsidered, such as the Asteroid Redirect Mission, Mars Sample Return, Saturn/Titan Sample Return, Ice Giant Exploration, Outer Planet Sample Return, large telescopes, and in-space infrastructure. Additional payload volume simplifies orbital operations, requiring less orbital assembly for large spacecraft. With the amount of energy able to be generated by the launch vehicle and imparted to the payload, significantly less time can be spent in Earth orbit - reducing the amount of propellant boil-off, and would eliminate the Earth flyby nuclear safety concern [20].

The three variations of the SLS that are depicted in Figure 1 are the Block 1, Block 1B, and Block 2B configurations. The SLS Block 1 configuration is considered to be capable of taking 70 metric tons to LEO, would have an 8.4 m common core, a five segment booster that is modeled from the shuttle booster, an interim Cryogenic Propulsion Stage (iCPS) upper stage, and the payload stage can be configured to use an Orion payload or a 5 m payload fairing. The SLS Block 1B configuration is estimated to be capable of take 105 metric tons to LEO, the core and boosters would be the same as that on the Block 1 configuration, the upper stage would be a common U.S. upper stage, and the payload can accommodate an Orion payload, 5 m payload fairing, or 8.4 m payload fairing. And, the SLS Block 2B configuration is estimated to be capable of carrying 130 metric tons to LEO, would have an 8.4 m common core, some kind of an advanced booster, a common US upper stage, and would be outfitted to carry a 10 m payload fairing.

None of the heavy lift launch vehicles from either SpaceX or NASA have yet to be built or had a test flight, so their performances to this point are speculative, but when they are operational they will be the biggest and best launch vehicles available for any sort of space mission designs.

MISSION DESIGNS FOR ASTEROID 2015 PDC

Mission designs to target 2015 PDC are shown in this section to demonstrate the capabilities of AMiDST and show the feasibility of reaching this kind of asteroid if the need arose.

The scenario for asteroid 2015 PDC begins as follows: The asteroid is discovered on April 13, 2015, the first day of the conference, at magnitude 20.9, declination -39 degrees and heading south. It is assigned the designation “2015 PDC” by the Minor Planet Center, and classified as a Potentially Hazardous Asteroid (PHA) based on its orbit. The asteroid’s orbital elements are known fairly accurately even in the first few days. Its mean distance from the Sun (semi-major axis) is 1.77 AU, and the orbital eccentricity is 0.49. Its perihelion distance is 0.90 AU and aphelion distance is 2.65 AU; the orbital period is 864 days (2.37 years). The orbital inclination is fairly small: 5.35 deg. The asteroid’s orbit comes very close to the Earth’s orbit on its outbound leg, much like the Chelyabinsk impactor, but unlike Chelyabinsk, this asteroid impacts at its ascending node. Very little is known about the object’s physical properties. Its absolute magnitude is estimated to be about $H = 21.3 \pm 0.4$, which puts the asteroid’s size at roughly 100 to 500 meters. The large size uncertainty is due to uncertainties in both albedo and H value. At discovery, the asteroid is quite distant from the Earth, about 0.34 AU (51 million kilometers or 32 million miles). It is approaching our planet and slowly brightening, but it peaks at only magnitude 20.3 on May 4. It reaches a closest approach of about 0.19 AU (28 million km or 18 million miles) from Earth on May 12. It never gets within range of the Goldstone radar and it’s too far south at close approach for the Arecibo radar. The JPL Sentry system and University of Pisa’s CLOMON system both identify many potential impacts for this object at several future dates. The most likely potential impact date is September 3, 2022, but the impact probability for that date is still low in the first week after the asteroid is discovered. Nevertheless, as the object is tracked over the next few weeks, the impact probability for 2022 starts to climb, reaching 0.2% a month after discovery. Even as the asteroid fades past magnitude 22 in early June, it continues to be observed and tracked since the chance of impact just keeps rising. The first part of the scenario ends in mid-June 2015, when the probability of Earth impact in 2022 has reached 1% and continues to rise. The rest of the scenario will be played out at the conference. It is clear that the object will be observable through the rest of 2015, although it will be quite faint (22nd and 23rd magnitude) and observers will require fairly large (2-meter- class) telescopes to track it. In December 2015 and January 2016, the asteroid will fade through 24th and 25th magnitudes, requiring very large aperture telescopes such as the 4- and 8-meter class facilities of CFHT, Keck, Gemini, Subaru, VLT, etc. In the spring of 2016, the asteroid will move too close to the Sun to be observed, and it will remain unobservable for about 7 months. The asteroid’s uncertainty region at the time of the potential impact is much longer than the diameter of the Earth, but its width is much less. The intersection of the uncertainty region with the Earth creates the so-called “risk corridor” across the surface of the Earth. The corridor wraps more than halfway around the globe [8].

The fictional asteroid 2015 PDC is detected on April 13, 2015 and is believed/known to be on an Earth-impacting trajectory, with a predicted impact on September 3, 2022. That means that there is about seven years of time in which any missions can be devised and conducted for reconnaissance or deflection/disruption attempts. Since the asteroid is only estimated to have a 0.2% impact probability shortly after its discovery, a rendezvous mission to the target body is considered in order to gain more information about the asteroid’s orbit and physical parameters. Once a rendezvous mission has launched and had some time near the target, a disruption mission to the hazardous body can be launched to try and mitigate the threat to Earth.

Given the information known about the asteroid to this point, it is easy to see how difficult a rendezvous mission to asteroid 2015 PDC will be, so the choice of spacecraft size is fairly irrelevant to the rendezvous mission design process. However, the mass of the spacecraft that would be used

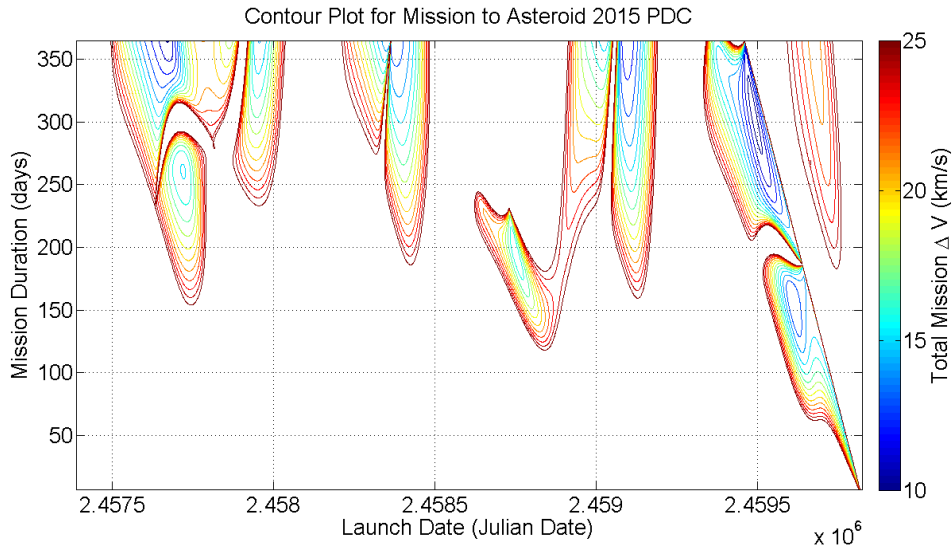


Figure 3. Contour plot of total mission ΔV for a rendezvous mission to asteroid 2015 PDC before its potential Earth impact in 2022.

in the disruption mission design analysis will be 10,000 kg, for it is assumed that a spacecraft of that size would be massive enough to disrupt the asteroid. With such a large spacecraft size, it also allows us to focus on short-term missions (ones that could be launched and encounter the target body within the last year prior to the expected impact) and the list of possible launch vehicles capable of putting the spacecraft into the desired interplanetary orbit shrinks. Therefore, the launch vehicles included in the mission design analysis will consist of the the heavy lift launch vehicles that are currently available and should be available in the future: Delta IV Heavy, Falcon Heavy, and a few varieties of the SLS.

Rendezvous Mission Designs

For a rendezvous mission, one of the most important parameters to consider, beyond the ability to place the spacecraft into the appropriate trajectory to meet the target, is the speed at which the spacecraft approaches the target. If that speed is too large, the spacecraft would require a large amount of propellant in order to make up the speed difference. Therefore, one of the quantities to be minimized through the mission optimization process is the total mission ΔV . Figure 3 shows the resulting rendezvous mission contour plot for asteroid 2015 PDC. Looking at the contour plot, it is easy to see that there are only a handful of regions where potentially feasible rendezvous missions could exist to this asteroid body. Most of the potentially mission feasible regions lie at the top of the contour plot, where the mission durations are highest. There is one region however that could have feasible missions around 150 days in duration, but the launch date is about a year prior to the expected impact date. Such a late launch date for a rendezvous, reconnaissance mission does not really allow for the potential of several deflection/disruption mission attempts, in the case that they fail or do not have the desired effect. In order to allow for a larger window in which deflection/disruption missions can be conducted, the time between target acquisition and the expected impact date, within the AMiDST program, will have a lower bound limit of 1000 days (about 3 years).

Total mission ΔV and acquisition time are only two of the important mission parameters needed

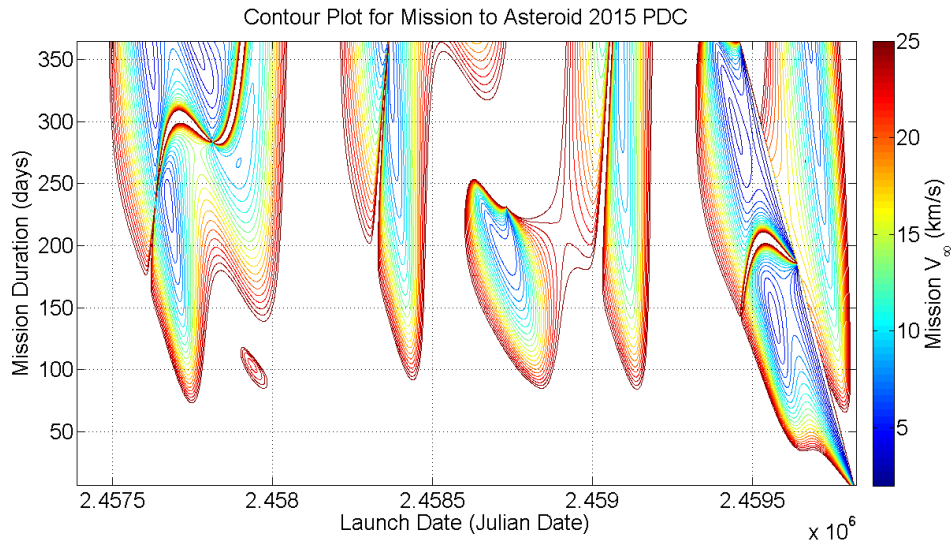


Figure 4. Contour plot of mission V_{∞} for a rendezvous mission to asteroid 2015 PDC before its potential Earth impact in 2022.

to evaluate mission logistics, the mission V_{∞} is also needed in order to assess the relative ease with which the launch vehicle can set the spacecraft in its interplanetary orbit. A lower mission V_{∞} would indicate that the interplanetary orbit would be easy to enter into, after leaving the Earth's sphere of influence. On the other hand, a higher mission V_{∞} would be either because the energy of the hyperbolic orbit needed to enter into the required interplanetary trajectory is very large or the interplanetary trajectory is out of the ecliptic plane. Figure 4 shows the rendezvous mission V_{∞} contour plot for asteroid 2015 PDC. Observing the contour plot, it can be seen that the low mission V_{∞} values exist in the same regions as the low mission ΔV values, and in a couple other regions as well. In those regions where the V_{∞} values are low and the ΔV values are not, it would imply that the relative arrival velocity between the spacecraft and the target body is the larger contributor to the overall mission ΔV . Despite the requirement and limitations placed on the value of the mission V_{∞} , in regards to the optimal rendezvous mission, the limiting factor in this particular mission design will be the total mission ΔV and the relative arrival speed.

In order to find the optimal rendezvous mission for a spacecraft mission to asteroid 2015 PDC, the following mission parameters were chosen as a part of the optimization process: launch date, total mission ΔV , mission C3, target acquisition time, mission duration, and relative arrival speed. The resulting mission trajectory is shown in Figure 5. The green line indicates the track that Earth takes during the time that it takes the spacecraft (blue line) to arrive at asteroid 2015 PDC (red line). Within the time that it would take the Earth to complete one revolution, the spacecraft launched on September 5, 2018 would arrive at asteroid 2015 PDC and begin its reconnaissance mission. The launch window surrounding this optimal mission is fairly small, about four days in length. Despite the limited length of the launch window, the remaining of the top ten missions occur within that same launch window but with varying length of mission durations. From that optimal launch window, there would be around 1094 days between the spacecraft's encounter of the asteroid body and the expected impact date for the spacecraft to observe the physical parameters of the body and track the asteroid's orbit track around the Sun.

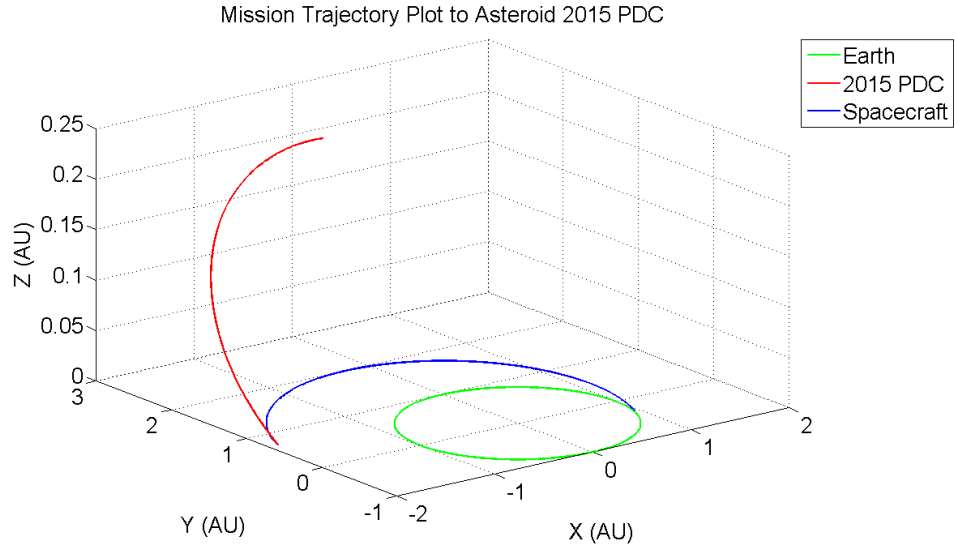


Figure 5. Trajectory plot for the optimal rendezvous mission to asteroid 2015 PDC between its discovery date and the expected impact date.

Based on the shape of the spacecraft trajectory for the optimal rendezvous mission depicted in Figure 5, the spacecraft has to be launched from Earth's sphere of influence on a fairly high energy hyperbolic orbit ($C3 \approx 26.5 \text{ km}^2/\text{s}^2$) because while the interplanetary orbit stays near the Earth's ecliptic plane, the trajectory reaches beyond Mars orbit and encounters the asteroid body nearly 2 AU from the Sun (almost 3 AU from Earth). The total mission ΔV is about 12.38 km/s, about 8 km/s of which are needed by the spacecraft to match the asteroid's speed. Based on the resulting optimal rendezvous mission parameter values, the necessary ΔV that the spacecraft needs to match the asteroid's speed is relatively large - meaning that the spacecraft needs to be large enough to hold the necessary fuel needed to make that change in speed. In fact, about 85% of the total spacecraft mass would need to be fuel in order to successfully achieve that kind of velocity change. So, if a very large spacecraft (mass of 5,720 kg) is chosen to carry out this mission, then only about 950 kg of the spacecraft would be available to be used for the payload, structure, and instrument suites.

Disruption Mission Designs

The difference between a disruption mission and a rendezvous mission is the approach to the target body. The optimization process would be conducted in much the same way as that for the rendezvous mission, but would ignore the relative arrival speed of the spacecraft to the target asteroid. The use of a large spacecraft as a pure kinetic impactor implies that the relative speed between the body is fairly irrelevant to the AMiDST mission design process, but would be useful to know for the momentum and energy transfer from the spacecraft to the target body. Before looking at the different mission design types, the total mission ΔV (Figure 6) and V_∞ (Figure 7) contours can be analyzed to understand how accessible asteroid 2015 PDC is from Earth between 2015 and 2022 for a deflection/disruption mission. Examining Figure 6, it can be seen that short-durations missions (less than 100 days) would be very difficult to be feasibly constructed for a spacecraft, outside a few months before the expected impact date. When looking at mission durations of more than 100 days, it appears that there are periodic regions of the contour plot where a an intercept mission can be

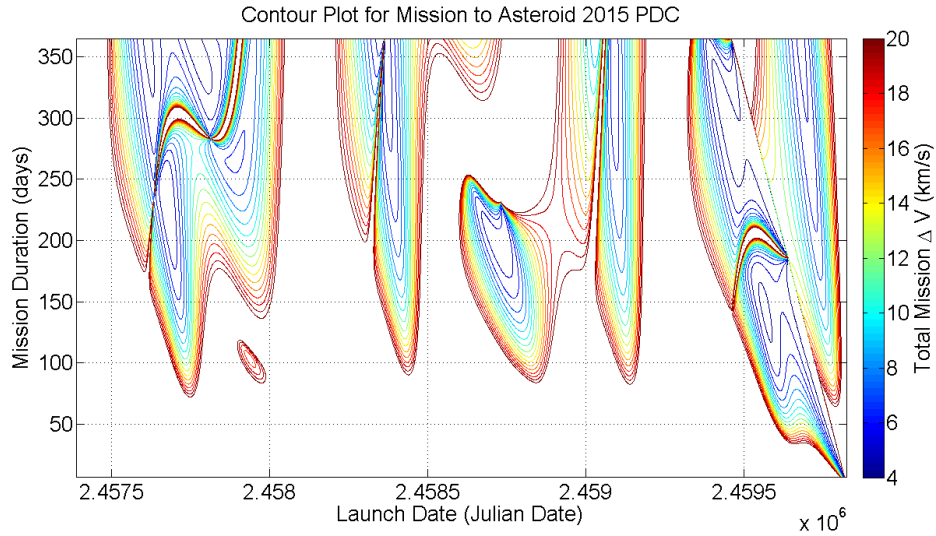


Figure 6. Contour plot of total mission ΔV for an intercept mission to asteroid 2015 PDC before its potential Earth impact in 2022.

launched to asteroid 2015 PDC. The V_∞ contour in Figure 7 validates the notion that short-duration missions would be hard to come by before the few months prior to the expected impact. Observing the remaining portion of Figure 7, it can be seen that the accessible regions of the contour plot are smaller than what they appeared to be in ΔV contour plot. Based on the V_∞ contour plot. In fact, the easiest times to launch a mission would be soon after its discovery (around the 2015/2016 time frame) and within a year of the anticipated impact date. If the rendezvous mission was conducted, the spacecraft would arrive at the target body nearly three years prior to the expected impact date, but to give time for the spacecraft to gather enough information about the orbit and the physical characteristics of the asteroid, the mission designs discussed here will focus on short-warning times - 90 day, 60 day, and 30 day warning time mission designs.

Short-duration, Long-dispersion Mission The short-duration, long-dispersion mission type is the most desirable disruption mission type given its tendency to maximize the amount of time from launch to the anticipated impact date. For the case of asteroid 2015 PDC, the optimal short-duration, long-dispersion mission results are discussed here briefly to show its impracticality.

The required mission ΔV and C3 orbit for a short-duration, long-dispersion mission scenario would be too large for any launch vehicle currently in operation to place even a small spacecraft into the necessary orbit to intercept the target body. None of the top 10 optimal missions are feasible: the mission ΔV is nearly 14 km/s from LEO and the required orbit C3 is about $350 \text{ km}^2/\text{s}^2$ - not even the most powerful SLS launch vehicle configuration is capable of lifting a spacecraft into this kind of orbit. The trajectory depicted in Figure 8, gives justification for the highly energetic orbit required by the spacecraft to leave Earth and meet 2015 PDC. The algorithm tried to push the mission duration as high as possible, in an attempt to reduce the required ΔV and C3, but the restriction of a maximum mission duration of 90 days did not allow for more feasible missions to be considered in this analysis. The date at which this short-duration, long-dispersion mission would launch would be in the middle of a launch window centered on December 18, 2016, a little more than a year and a half after discovery of asteroid 2015 PDC. To compound the difficulty of this mission design, if there was some way of getting the spacecraft into the required interplanetary orbit, there

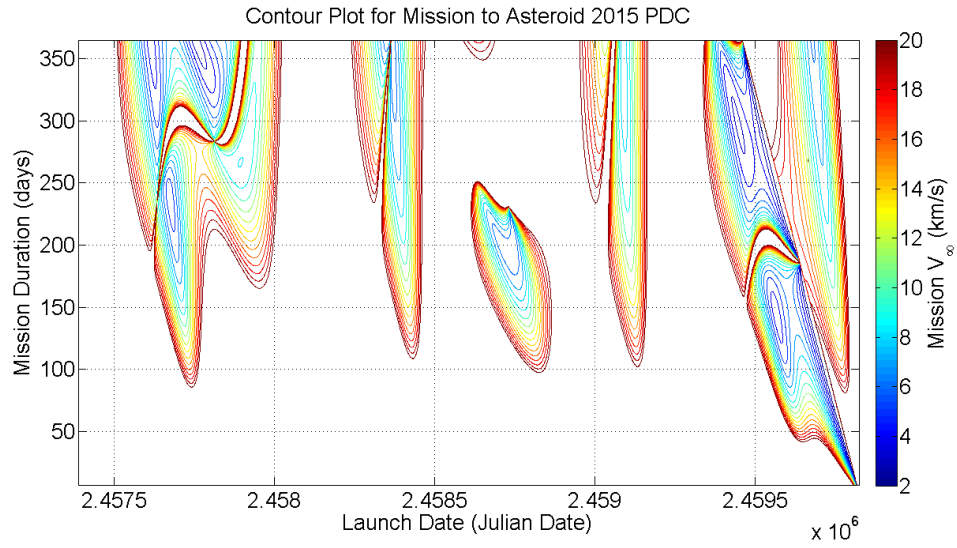


Figure 7. Contour plot of mission V_∞ for an intercept mission to asteroid 2015 PDC before its potential Earth impact in 2022.

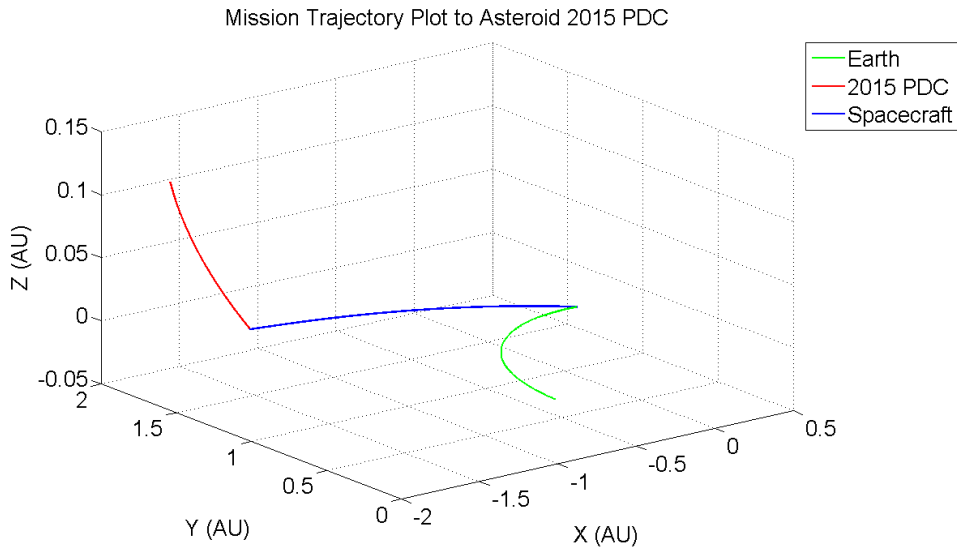


Figure 8. Trajectory plot for a short-duration, long-dispersion intercept mission to asteroid 2015 PDC.

Table 1. Design parameters for a 90-day-warning intercept mission to asteroid 2015 PDC.

Mission Parameter	Value
Asteroid	2015 PDC
Spacecraft Designation	Kinetic Impactor
Spacecraft Mass (kg)	10,000
Departure ΔV (km/s)	3.374
C3 (km^2/s^2)	3.244
Departure Date	July 13, 2022
Mission Duration (days)	45
Dispersion Time (days)	7
Arrival Angle (deg)	19.079
Impact Velocity (km/s)	13.073
Arrival Date	August 27, 2022
Launch Vehicle	Falcon Heavy, SLS

would be the added difficulty of the hypervelocity encounter between the spacecraft and asteroid. In the case of this mission design, the relative arrival speed between the two bodies comes to be about 35 km/s, much faster than any relative arrival speed found in any other mission designs. It is possible that a more feasible optimal mission design could be obtained with an alternate definition of short-duration, but for the sake of consistency between this example asteroid, and the other asteroids studied [9], the definition is unaltered.

Assuming that it is too late for all other options, or even worse they have all failed, to diminish the threat posed by asteroid 2015 PDC, short-duration, short-dispersion mission designs are constructed to deal with the imminent threat from the target asteroid. A short mission duration and short dispersion time implies that the asteroid has now entered the terminal phase of its orbit with respect to the Earth. Thus, with such a late launch window, the task of reaching the asteroid from Earth should be simpler to accomplish.

90-Day-Warning Mission Design For this 90-day warning time mission design, it is designated that half that warning time is set for the maximum allowable mission duration and the other half is the maximum allowable dispersion time - up to 45 day mission duration and up to 45 day dispersion time. With the bounds set on the mission duration and dispersion time, along with the spacecraft size set to 10,000 kg, the best mission to intercept 2015 PDC is described in Table 1. The hyperbolic orbit that the spacecraft would be set on to leave Earth's vicinity has a C3 value of about $3.25 \text{ km}^2/\text{s}^2$. The top 10 missions had mission durations that were running up against the upper bound of 45 days, and had dispersion times near the lower bound of seven days. The top mission, described here, has a mission duration of 45 days and a dispersion time of 7 days. With the anticipated impact date of the asteroid on September 3, 2022, this mission would launch on July 13, 2022 and would arrive at the target body with a relative impact speed of just over 13 km/s, impacting the body at an angle of about 19 degrees. Looking at the interplanetary trajectory of the spacecraft, seen in Figure 9, it is easy to see that the C3 orbit is actually quite easy for most any currently available launch vehicle to place a reasonably sized spacecraft into it. However, given the sheer size of the spacecraft designated for these last-minute missions none of the currently available launch vehicles are capable of entering into the required orbit, not even the Delta IV Heavy. Based on the assumed

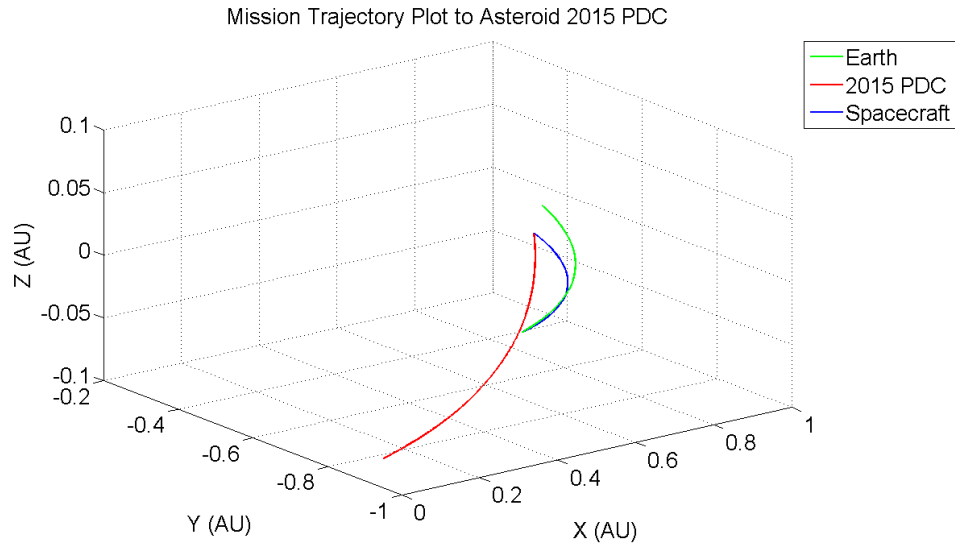


Figure 9. Spacecraft mission trajectory to 2015 PDC with 90-day warning time.

capabilities of the Falcon Heavy and SLS launch vehicles, they would be capable of lifting the spacecraft and enabling the possibility of mission success.

Allowing for more time for the mission flight time, while still maintaining the 90-day mission warning time, the mission parameters stay pretty similar to the ones given in Table 1. The mission durations still ran to upper bound and the dispersion times stayed near its lower bound. The missions got easier to launch, lower C3 values, with the more flight time that was allowed, but the launch vehicles needed for mission feasibility did not change, still requiring the Falcon Heavy or SLS to place the spacecraft in the necessary interplanetary trajectory. What this analysis shows is that with 90 days of warning time, the mission duration and launch vehicle selection are not limiting factors to the design of the asteroid intercept missions. In fact, the dispersion time is the primary parameter of interest and concern, regardless of the amount of time designated to mission duration and dispersion time the missions were easier to conduct with less dispersion time.

60-Day-Warning Mission Design Following the precedent set by the previous mission design example, the 60-day warning time design is split to have half that time designated for the mission duration and the other half for the dispersion time - up to 30-day mission duration and up to 30-day dispersion time. Based on the results of the 90-day warning time mission design study, the results of this 60-day warning time mission design study, shown in Table 2, are similar to those of the previous mission design study. With the limitation of up to 30 days for the mission duration means that the orbit necessary to place the spacecraft into the intercept trajectory to meet 2015 PDC is more energetic. The top evaluated mission design requires a mission duration of 30 days, the upper limit for the spacecraft mission duration, leaves seven days for the impact to take effect before the anticipated Earth impact date, and the C3 of the interplanetary trajectory is a bit more than $7 \text{ km}^2/\text{s}^2$. The spacecraft trajectory relative to the Earth and 2015 PDC during that 30-day mission duration is shown in Figure 10. The additional launch energy required to place the spacecraft into the mission orbit results in a more energetic impact between the spacecraft and the asteroid and a larger relative impact angle. Like the 90 day warning time mission design, the launch vehicles capable of lifting the 10,000 kg spacecraft into the interplanetary trajectory to intercept asteroid 2015 PDC are the

Table 2. Design parameters for a 60-day-warning intercept mission to asteroid 2015 PDC.

Mission Parameter	Value
Asteroid	2015 PDC
Spacecraft Designation	Kinetic Impactor
Spacecraft Mass (kg)	10,000
Departure ΔV (km/s)	3.543
C3 (km^2/s^2)	7.035
Departure Date	July 28, 2022
Mission Duration (days)	30
Dispersion Time (days)	7
Arrival Angle (deg)	20.569
Impact Velocity (km/s)	13.961
Arrival Date	August 27, 2022
Launch Vehicle	Falcon Heavy, SLS

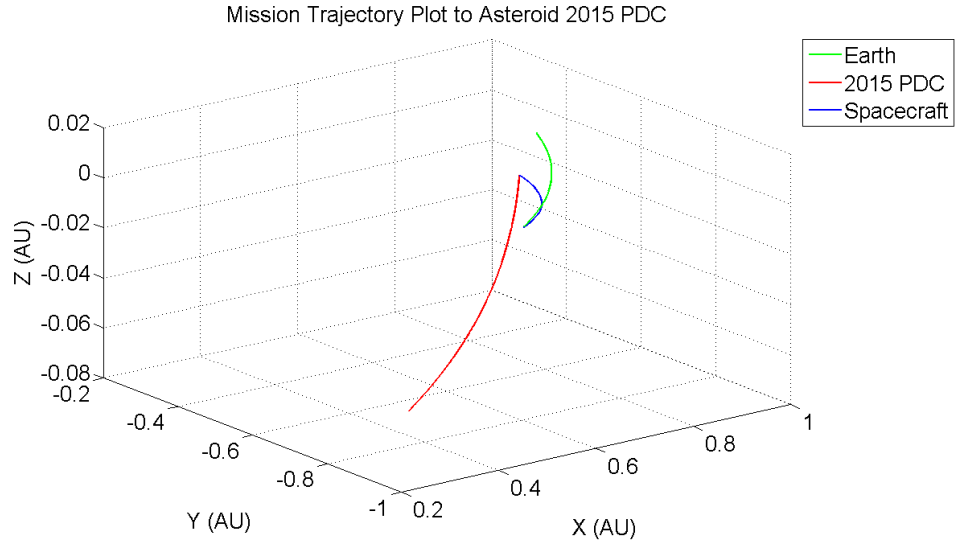


Figure 10. Spacecraft mission trajectory to 2015 PDC with 60-day warning time.

Table 3. Design parameters for a 30-day-warning intercept mission to asteroid 2015 PDC.

Mission Parameter	Value
Asteroid	2015 PDC
Spacecraft Designation	Kinetic Impactor
Spacecraft Mass (kg)	10,000
Departure ΔV (km/s)	4.425
C3 (km^2/s^2)	27.806
Departure Date	August 12, 2022
Mission Duration (days)	15
Dispersion Time (days)	7
Arrival Angle (deg)	25.411
Impact Velocity (km/s)	16.571
Arrival Date	August 27, 2022
Launch Vehicle	SLS

Falcon Heavy and the SLS variations.

30-Day-Warning Mission Design The shortest warning time mission design study conducted was 30 days - up to 15 day mission duration and up to 15 day dispersion time. With such a short timespan for the spacecraft to get from the Earth to the asteroid, the mission trajectory should be fairly simple in terms of the ease of getting into the orbit and getting to the threatening asteroid. However, looking at Table 3 it can be seen that a mission at this late stage of the asteroid's approach is anything but easy. Needing over 4.5 km/s from low-Earth orbit to enter into the hyperbolic escape orbit of almost $28 \text{ km}^2/\text{s}^2$ to intercept 2015 PDC, the difficulty of this interplanetary trajectory can be seen in Figure 11.

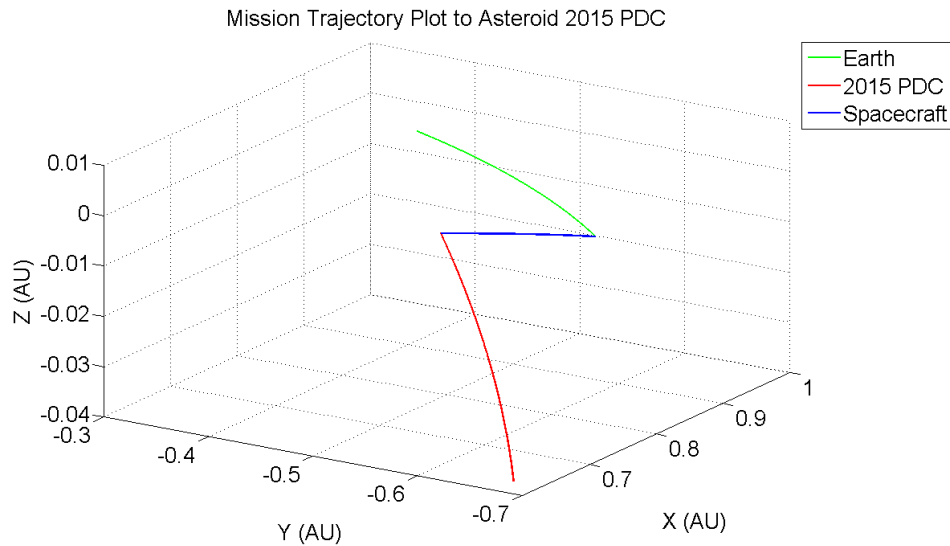


Figure 11. Spacecraft mission trajectory to 2015 PDC with 30-day warning time.

It is easy to see that the spacecraft needs to leave the ecliptic plane of the Earth to intercept the asteroid in 15 days. The combination between the launch energy given to the spacecraft to meet 2015 PDC and the position of the asteroid in its orbit (going away from periapsis) explains the large relative impact velocity and relative arrival angle between the spacecraft and 2015 PDC. Unlike the 90 day and 60 day warning time missions shown previously, the 30 day warning time mission design can only be accomplished by the use of an SLS launch vehicle. Taking a closer look at the top 10 mission designs for this type of mission design only the top four missions would be able of being completed using the SLS Block 1 launch vehicle or larger - the rest of the top missions would require at least the SLS Block 1B configuration to be feasible.

CONCLUSIONS

The trajectory of asteroid 2015 PDC does not make the design of late, last-minute mission designs trivial. Asteroid 2015 PDC has an orbit that has a periapsis inside Earth's orbit and an apoapsis that is beyond Mars's orbit, meaning that it spends a relatively small amount of time near Earth and has a fairly large speed near the periapsis of its orbit. That implies that these short-duration mission designs would require a lot of launch energy, which only complicates a planetary defense mission requiring a 10,000-kg kinetic-energy impactor system. Because not only does a launch vehicle need to lift such a heavy spacecraft off the ground, but it needs to provide the spacecraft with enough velocity to cover a large distance in a relatively short amount of time. As the example mission designs presented in this paper show, short duration missions early in the available launch window between discovery and the anticipated Earth impact are infeasible, but last-minute mission designs are possible to be conducted through the use of launch vehicles that are currently being developed. Despite the relative ease of these late-launch and short-dispersion time missions, it should be noted that they should be considered as a last resort when all other attempts have failed, and not considered only when there are no other options able to be taken.

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