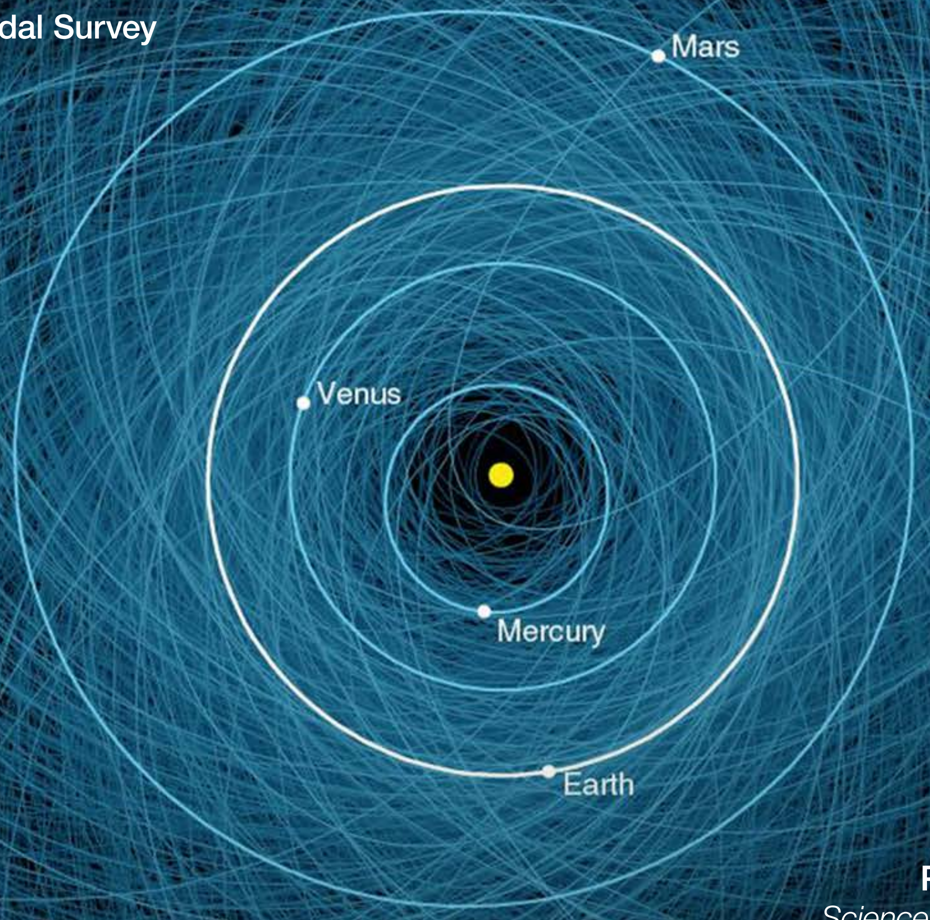
National Aeronautics and Space Administration

Planetary Defense Missions

Rapid Mission Architecture Study

Small Solar System Bodies
Planetary Science Decadal Survey

August 13, 2021



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Disclaimers/Acknowledgements

Pre-Decisional Information – For Planning and Discussion Purposes Only

The research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration (80NM0018D0004).

The cost information contained in this document is of a budgetary and planning nature and is intended for informational purposes only. It does not constitute a commitment on the part of JPL and/or Caltech.

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PLANETARY SCIENCE DECADAL SURVEY

Mission Concept Study Final Report

Table of Contents

List of Figures

List of Tables

EXECUTIVE SUMMARY

The urgency to develop mature, validated Planetary Defense capabilities has recently become clearer as knowledge of Earth's impact history has improved, and as illustrated by the dramatic explosion over Chelyabinsk, Russia in 2013, which was captured in many audio-video recordings. Vast improvement of Near-Earth Object (NEO) survey sensitivity and discovery rate by ground and space-based observatories beginning in the mid-2020s, followed by the close flyby of the 340-m asteroid Apophis in 2029 within the Geosynchronous satellite belt, will substantially increase public awareness and scientific understanding of this existential threat to our planet.

This report responds to a request from the NASA Planetary Science Decadal Survey Small Solar System Bodies panel to "investigate a range of planetary defense mission concepts in the areas of both Near-Earth Object (NEO) characterization and mitigation in support of upcoming Decadal Survey discussions. To that end, we review existing study materials and results of previous flight projects, as well as the current national policy for threat emergency protocols. We consider operational scenarios and present new results comparing and contrasting the efficacy of available mitigation (asteroid deflection) technologies. We consider over 30 potential Planetary Defense (PD) demonstration missions for their potential costs and benefits, weighed against their operational risks.

Without knowing future budget allocations, we recommend a variety of projects and budget timelines for consideration. Establishing a regular cadence of flight demonstrations is needed to ensure we are prepared to respond should a global or regional threat emerge. The highest priority recommendation is a rapid-response, flyby reconnaissance mission targeted to a challenging NEO, representative of the population of highest-probability hazardous objects. Such a mission should test flyby characterization methods to assess their capabilities and limitations, to better prepare for a short-warning-time NEO threat. Other priority mitigation and characterization mission objectives identified include, in no particular order: 1) a characterization tour mission to gain characterization information required for future deflection/disruption missions, and to exercise characterization capabilities for a range of NEO targets; 2) a kinetic impact mission on a small NEO (~50–100 m in diameter) and at a higher closing speed than the Double Asteroid Redirection Test (DART) mission to acquire the needed experience of kinetic impact mitigation missions; and, 3) a slow-push mitigation mission demonstration, such as ion beam deflection, to develop several different technologies that can be available and optimized for specific mitigation situations that may arise. In addition to key technology and capability developments, we advocate the definition and award of two new flight projects in the next decade. The PD missions considered and described divide into two fundamental categories: flyby/ intercept and rendezvous missions. If the PD budget profile allows, a combination of these mission types would cost-effectively accomplish the highest number of PD priority mission objectives in the next decade. The costs for such hybrid missions are estimated at ~$500 M or less including launch services and operations, and emphasize PD technology demonstrations as opposed to competed science-driven projects like Discovery. Hybrid missions could combine characterization and mitigation objectives from otherwise distinct flights, offering a high return on investment. Other lower-cost, more focused demonstrations have also been identified and are described in detail here. These PD demonstrations could provide the basis for a sustained PD mission line into the next decade.

Effective Planetary Defense requires a long-term commitment to technology development, demonstrations, situational awareness, and operational readiness. Although it may not happen for some time, it is certain that the Earth will be impacted by an asteroid large enough (~50 m diameter or larger) to cause significant damage/loss of life. There exist >100,000 NEOs of this size or larger, which could lead to an impact like the 1908 airburst over Russia when a 40–60 m diameter asteroid flattened 2000 km^2 of forest in Tunguska. If this had instead occurred over a modern city there clearly would have been mass casualties. With a moderate investment and long-term planning, we have the ability to protect the Earth from the majority of such threats. This report proposes effective steps in that direction.

1 INTRODUCTION

1.1 EXISTENTIAL THREAT

While geological records and modern astrometry have clearly established that Earth impacts by external objects are a low-likelihood, the consequence is high, as exemplified by the 2013 videos of an estimated 17 m asteroid exploding 30 km above Chelyabinsk, Russia. This relatively minor event on a cosmic scale unleashed approximately 0.5 Mt of energy, resulted in significant structural damage, and caused over 1600 injuries (Figure 1-1), but fortuitously no deaths. Nonetheless, current threat protocols establish 50 m diameter as the threshold for mitigation (Interagency Working Group on NEO Impact Threat Emergency Protocols, 2021).

The high end of the risk spectrum may hold *eight orders of magnitude more energy* (10^8 Mt), represented by events such as the Chicxulub impact near the Yucatan peninsula that is believed to have caused mass extinctions approximately 66 million years ago (Wei-Haas, 2019). Fortunately, Near-Earth Objects (NEOs) of this size class (> 1 km) are readily observed and it is estimated that there are at most only a few tens of these yet undiscovered (Harris & Chodas, 2021). The known population of ~890 large (>1 km diameter) NEOs have been confirmed to not constitute near-term impact threats (Figure 1-2). The George E. Brown Jr. Near-Earth Object Survey Act passed by Congress in 2005 mandates that NASA carry out programs to discover at least 90% of NEOs greater than 140 m in diameter. This is the objective of the NEO Surveyor Mission (NEOSM) currently in Phase B, and other ground-based searches planned and already in progress (§2). However, these searches address only a small portion of the gap between NEOs already discovered (red curve in Figure 1-2) and the estimated actual population (blue circles, in Figure 1-2). Therefore, the primary source of risk is the large number of intermediate-sized objects approximately 50 m in diameter and larger that have not yet been discovered, shown by the red shaded region.

The threat is serious enough that hypothetical impact exercises have been done bi-annually since 2013 (JPL CNEOS, 2021a) to improve operational readiness and to better understand weak links in our PD capabilities.

Figure 1-1. Airburst over Chelyabinsk Feb 15 2013 (Tuvix72, 2013)

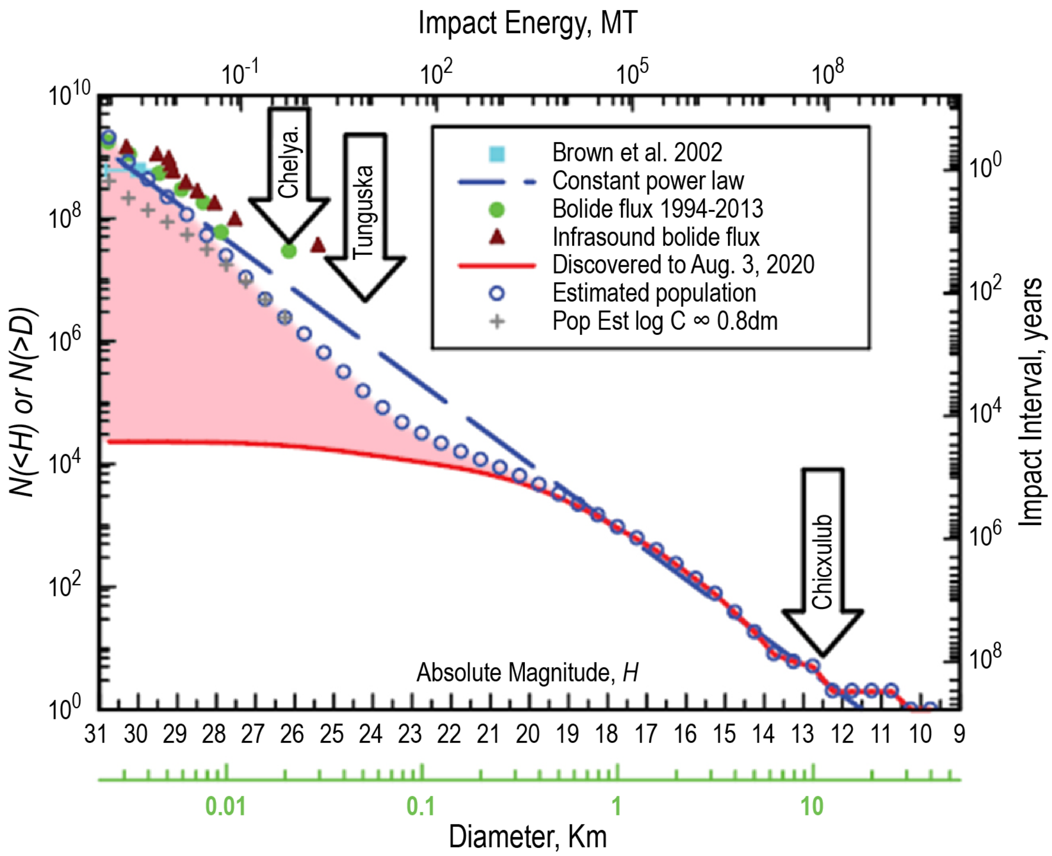


Figure 1-2. Near-Earth asteroid cumulative population (N) in terms of estimated diameter *D* and absolute magnitude *H.* The shaded area represents the estimated population that remains to be discovered. The absolute numbers are difficult to see on this log plot: the number of undiscovered NEAs larger than 140 m (shaded sliver to the right of H = 22) is on the order of 10,000 (Adapted from Harris & Chodas (2021)).

1.2 STUDY REQUEST

This study report responds to a Dec 4, 2020 request for a Rapid Mission Architecture (RMA) trade study to investigate a range of planetary defense mission concepts in the areas of both NEO characterization and mitigation. This RMA study is needed to inform Decadal Survey discussions that will identify and prioritize planetary defense missions for the next decade, taking into account planetary defense objectives as well as the technological readiness and the estimated mission costs that result from this study. It takes as its starting position that planetary defense missions already supported by NASA and ESA, namely DART, NEOSM, and Hera, are successfully implemented.

1.3 SCOPE

This study includes a broad survey of representative *demonstration* missions that could be considered affordable in the next decade (2023–2032), and provides an assessment of their system architecture, sizing, cost, complexity, and other considerations. These missions include a wide variety of characterization and mitigation objectives that are unique to development of operational Planetary Defense capabilities (Appendix C). This study does not consider operational missions or the programmatic issues associated with rapid response to a newly-discovered threat, but rather demonstrations of critical techniques for risk reduction, operational readiness, and expanding the NEO knowledge base (§4). However, operational considerations are used to guide selection of the most useful demonstrations (Appendix C).

1.4 STUDY ASSUMPTIONS AND GUIDELINES

This study is developed in support of the Planetary Science and Astrobiology Decadal Survey, and assumes that the desired mission cost to be <$500 M (including LV procurement but excluding foreign contributions).While some of the results in §4 exceed this, we believe they provide useful information to help understand the large-scale options and trade space. Per para 1.1 above we focus on appropriately sized NEOs (> 50 m in diameter). We especially assume successful launch of the NEO Surveyor Mission (NEOSM), which is considered vitally important to closing the population survey gap evident in Figure 1-2. We also assume successful launch and operation of the DART (2021) and Hera (2024) missions, and continued operation of ground-based NEO discovery assets (e.g., Catalina Sky Survey, PANSTARRS, etc.) and completion of the Vera C. Rubin Telescope/Large Synoptic Survey Telescope (LSST) (§2).

Like DART, our baseline assumption is a Class C, single string spacecraft design using mostly existing technologies, and a moderately-aggressive project schedule of 54 months from start to launch (§4.1.5). To expand the trade space into more affordable options, we have also considered the savings possible from lower-cost launch vehicles, shorter schedules and Class D implementation, and a possible future generation of lower mass and power instruments.

1.5 STUDY OBJECTIVES AND APPROACH

This study performs first-order sizing and costing of a variety of system architectures responding to the list of missions in the trade study matrix provided in the study request. These include demonstration of "mitigation" missions to prove existing technologies and to improve operational readiness to prevent a NEO impact with Earth. We also study "characterization" missions which demonstrate and exercise key capabilities to obtain information about the dynamical and physical characteristics of NEOs, crucial information that would be needed into inform any mitigation approach. Both mission types would collect information that will be useful for planetary defense objectives, particularly to inform the development and implementation of future mitigation strategies and techniques. Better detection and characterization of the NEO population as a whole is also critical to reduce risks and ensure successful mitigation.

Understanding the average density and structural integrity of smaller NEOs (e.g., <~140 m diameter) is critical to understanding the threat adequately to mount a proper defense, but such information is currently inadequate. Detailed data from ground-based planetary radars (e.g., Goldstone and Arecibo) and the few in-situ missions to date suggests a very heterogeneous population varying from "rubble piles" to solid rocks (§2.1). This is problematic because the efficacy of the available deflection methods depends on the mass, cohesiveness, and other physical properties of the object. It is also of great concern that an intended deflection may disrupt a loosely-bound object into multiple objects, and inadvertently increase the probability of impact (albeit with smaller pieces). Thus, without a valid data base of reliable NEO characteristics, we cannot confidently predict the properties of a newly-discovered threat without actually observing it in situ. For example, approximately 15% of NEOs are actually binary systems of two gravitationally bound objects; launching a deflection mission to one of these binaries without that knowledge could jeopardize mission success. In operational scenarios, we foresee a two-step process of characterization, followed by mitigation activities appropriate for that particular object. In stressing cases this in-situ characterization may need to be performed more rapidly than currently feasible (§3, §5, Appendix C).

There are many cases where characterization and mitigation activities may be naturally combined into a single mission, as seen in §4. So, while the characterization and mitigation objectives may differ operationally, the reader should keep in mind that for present purposes they may be combined in various ways as a vastly reduced set of possible demonstration flight projects.

The majority of this report involves implementation of JPL's Team X concurrent design process, operated in a high-level architecture mode (Nash, 2020). This produces spacecraft sizing (mass and power) and other information sufficient to perform first-order costing (Hogstrom et al., 2019). Instrument sizing (mass and power) is conservatively allocated using existing hardware designs of analogous instruments. To cover the large space of representative concepts within study time resources, we limit the design detail to parametric descriptions of key spacecraft subsystems but do not establish actual design details. This is sufficient for the desired cost estimations. We also have provided as parameterized inputs to the Team X sessions some attributes that are typically provided as outputs (e.g., ΔV and data volume), including representative instrument descriptions (mass and power). This is presented more fully in §4.

1.6 SUMMARY OF PREVIOUS WORKS AND DESIGN CONCEPTS

Research papers, white papers, technical reports, and slide packages were pulled from various sources including available online repositories, and JPL-internal libraries. The literature study assembles these papers, categorizes them, and includes statistics regarding age, target body, maturity, and other data (Appendix B). The materials collected were considered and informed the architecture trade space presented in §4.

2 PROGRAMS AND UPCOMING EVENTS THAT INFORMED THE STUDY

As the current rate of detection is insufficient, NEOSM (§2.3) is dedicated to achieving the detection of 90% of 140 m diameter and larger objects goal set forth by Congress, supported by existing and upcoming Ground-Based Observatories (GBOs) (e.g., Vera C. Rubin Telescope/LSST). In addition, the synergy between GBOs and infrared orbiting assets (NEOSM and NEOWISE) will add significant value to the detection and characterization of this diverse group of objects which are poorly understood. The upcoming flight missions of DART and Hera (§2.2) will make contributions to PD characterization and mitigation: These missions will provide an important demonstration of the effectiveness of the kinetic impact technique, as well as collect valuable information regarding the physical nature of a small secondary asteroid within a NEO binary system.

2.1 PREVIOUS FLIGHT MISSIONS

The first spacecraft observations of asteroids or comets occurred in the 1980s with the International Cometary Explorer's 1985 comet flyby, five spacecraft which encountered Halley's Comet in 1986 from JAXA, Russia, and ESA, and Galileo's asteroid flybys in 1991 and 1993. These missions laid groundwork for future dedicated asteroid and comet exploration, which has been the objective of various high-profile NASA, JAXA, and ESA missions.

Each of the previous small body missions completed science campaigns which were aimed at diverse science objectives. They brought about new technologies, navigation methods, and targeted instrumentation. A striking conclusion is that most missions returned unexpected results which generated further questions about the properties of such objects (Table 2-1). This highlights the importance of characterization missions to help develop planetary defense mitigation techniques given the varied physical and dynamical attributes among the small body population.

Table 2-1. Previous missions to asteroids and comets have consistently yielded surprising results

Mission	Target	Key Milestones / Technologies	Unexpected Results
Giotto	1P/Halley	First comet intercept; Debris protection and attitude mitigation	Nucleus much darker than expected and only certain areas were active with jets (ESA, 2021).
NEAR	433 Eros	First asteroid rendezvous, orbit, and soft landing	Discovered ongoing surface chemical and spectral processes. Ponded craters with floors seemingly filled with fine dust and mass wasting. Asteroids can have immense structural complexity (McCoy et al., 2002).
Stardust	81P/Wild	First small body sample return (from coma)	Dramatically different surface than similar observed objects (Brownlee et al., 2004).

Mission	Target	Key Milestones / Technologies	Unexpected Results
Deep Impact	9P/Tempel	First high energy kinetic impact; First observation of subsurface structure of comet nucleus	Crater ejecta obscured desired imaging; hypervelocity KI is feasible but unique environmental situations may confound PD objectives (Henderson & Blume, 2015).
Hayabusa	Itokawa	First small body surface sample return; First multiple take-off and landing from a small body	Significantly out of family properties for body of this type. Various hypotheses regarding shape agglomeration and re-agglomeration (Fujiwara et al., 2006). Evidence towards rubble piles for small asteroids (huge implications for PD of small asteroids).
Hayabusa2	Ryugu	C-Complex asteroid type; First artificially generated crater on an asteroid and sampling of sub-surface material; Surface sample return; Surface rovers	Boulders everywhere when remote thermophysical modelling from the ground of the object suggested only fines on the surface. (Boulders may have implications for PD)
OSIRIS-REx	Bennu	C-Complex asteroid type; Surface sample return	Lack of fine-grained material on surface and boulders everywhere; surface sampling suggested extremely low strength surface; high amount of sample collected suggests fine grained material exists; detected small grains being naturally ejected from the surface.
Rosetta	67P/ Churyumov-Gerasimenko	First long term prox-ops around a comet; First landing on a comet and deployment of Philae lander	Odd duck-shape suggestive of two objects in the past that are now joined; Different isotopic water on/in body. Philae lander bounced and wedged under shadowed cliff limited operations; Detailed surface features seen in last moments of mission (Gibney, 2016)

2.2 DART AND HERA

DART is currently slated for launch in November 2021. The mission is motivated as a demonstration of asteroid deflection by kinetic impact, intending to strike the Didymos secondary (Dimorphos) with 500 kg of mass at 6.7 km/s in fall of 2022 (Rainey et al., 2020). The primary advantage of the DART mission over the similar and concluded Deep Impact mission is the smaller and more challenging target body, which makes it a valuable contribution to asteroid characterization as well. Models predict that Dimorphos is, a rubble-pile style body, with unknown cohesion and internal structure, though there are no direct observations or measurements of the object. The body's response to impact will be monitored at first by a pre-deployed CubeSat and by ground-based observers, and a follow-up mission (Hera) will inspect the results in 2026. Subsequent to DART, the next performance step that is important to demonstrate is successful intercept at much higher closing velocities that are more effective for KI mitigation (e.g., 10–20 km/s).

The fact that Didymos is a well-observed binary system means that the momentum exchange due to the impact will be more precisely quantifiable; this will help measure the enhancement factor "beta" that may result from the ejecta leaving the surface. The characterization of Dimorphos is novel as it will be the smallest asteroid visited to date and may shed light on otherwise unknown internal properties of rubble-pile bodies (Michel et al., 2018). Such properties are major assumptions which go into modeling NEOs and planetary defense mitigation techniques, and there is high uncertainty associated with them due to the continually growing diversity of observed objects (Naidu et al., 2020).

2.3 NEOSM

Properly-stationed spaced-based infrared observatories offer unique capability critical to PD objectives (NASEM, 2019). The NEO Surveillance Mission (NEOSM) is a space-based telescope mission currently in development which will serve as an infrared asteroid survey residing in the Earth-Sun L1 Lagrange point. Compared to the repurposed NEOWISE, which is over 10 years past its planned lifetime and only has two of four observation channels left functioning, NEOSM is a *dedicated* NEO planetary defense mission (Mainzer, 2019). NEOSM's position at the L1 Earth-Sun Lagrange point provides a marked increase in visibility as well as a cooler thermal environment when compared to NEOWISE's low Earth orbit (LEO) orbit. NEOSM's primary objectives are to detect, track, and characterize NEOs, to help meet the goal of finding 90% of the 140 m and larger NEOs.

By observing in the infrared, NEOSM can produce direct estimates of asteroid diameters, far superior to having to infer diameters based on absolute magnitude as must be done by GBOs. It will be targeted to obtain high quality orbits and physically characterize bodies of interest which are either known risks or newly detected (Sonnett et al., 2020).

NEOSM will operate synergistically with the Vera C. Rubin Telescope/LSST and the existing GBOs to achieve better detection and characterization performance through contribution of photometric and astrometric data which enables improved orbit and albedo determination, and mass estimates. Although the main focus of NEOSM is on discovering 140 m and larger NEOs, the mission is expected to detect smaller targets as well. This is important because even relatively small asteroids (e.g., 50 m diameter) pose a threat. There are more than 100,000 NEOs of this size or larger; we expect to detect many of these in the coming decade. It is possible that some of these will be on Earth-impact trajectories, making the operational decision-making process a real situation rather than hypothetical (Appendix C).

2.4 2029 APOPHIS ENCOUNTER

Apophis is a recently-discovered potentially hazardous asteroid (PHA) with a roughly 340 m mean diameter that is predicted to miss the Earth by just 32,000 km (inside the Geosynchronous belt at 35,786 km) on April 13, 2029 (Chesley, 2005). This makes it appear to be a compelling special target (Binzel et al., 2020; Cheng et al., 2020). However, the close physical approach to Earth does not equate to a low-ΔV (low-cost) rendezvous; it would require ~1.8 km/s, as is typical for other NEO rendezvous cases (§4.1.1). The close approach with the Earth does however suggest an opportunity for coordinated Earth-based observation campaigns (out of scope of this study). In addition, recent radar data has eliminated the probability of an Earth-impact for more than 100 years. The size of Apophis is larger than the objects that are the primary threats of interest (§1.5), so close examination of its characteristics may be less relevant than other, smaller PHAs. Due to its very close Earth flyby (and eventual returns), there is a great deal of concern about any activity that may alter its orbit and increase the hazard, so any direct interactions by spacecraft are not recommended. Finally, the OSIRIS-REx extended mission is currently proposing to rendezvous with Apophis subsequent to closest approach, and utilize its suite of remote sensing instruments for detailed observations including high-resolution imagery, spectroscopy, and altimetry (Lauretta et al., 2017; Bartels, 2021).

So while Apophis will be visible to the naked eye from Western Europe and Northern Africa, and hence create a great deal of interest in PD, Apophis is not the intended or highest priority target for the PD demo missions that are the subject of this report. However, given the parametric nature of the study cases herein, any of the rendezvous cases from §4 could envelope missions to Apophis if that were determined to be desirable.

3 PERTINENT OPERATIONAL SCENARIOS

New considerations of operational scenarios provide context for the Planetary Defense system architectures studied for risk reduction in §4, and for the recommendations in §7. While not explicitly requested for this study, accurate understanding of operational PD missions is critical to defining useful demonstrations thereof. This section is a summary of more in-depth considerations in Appendix C.

Response to new threats depends on their sizes and warning times, according to the logic flow from the Interagency Working Group on NEO Impact Threat Emergency Protocols (2021) (Figure 3-1).

3.1 EFFICACY OF DEFLECTION TECHNOLOGIES

Selection of appropriate deflection technologies for a given scenario is a delicate balance between generating an adequate amount of deflection, given the asteroid mass and available warning time, without causing unwanted disruption. For purposes of comparing deflection techniques, we impose a practical limitation of assessing deflection capability using a single high-performance launch (although in practice at least one backup would likely be launched). Previous case studies have suggested the hypothetical situation of a large number of launches and deflections in a short period of time, but this option is rejected as infeasible with today's infrastructure or for any likely future scenario (Barbee et al., 2018) and can lead to clearly unrealistic scenarios (Woo & Gao, 2021). While multiple launches over a longer time scale (e.g., years or decades) are certainly feasible, they are not considered here. Whatever the chosen deflection technique, knowledge of the mass, binarity, and precise orbit of the asteroid would be extremely valuable in designing and executing an effective deflection campaign (Appendix C). A prior reconnaissance ("recon") mission would therefore be highly desirable. A rendezvous would be preferred for reconnaissance, since it could provide this critical knowledge, but if there is inadequate time, a "fast flyby" recon (§4.2.2) would be still be highly valuable to provide at least approximate information of the object's key parameters.

3.2 KINETIC IMPACT

The Kinetic Impact (KI) technique is probably the most obvious and simplest approach to asteroid deflection (§4.2.5), although it is limited by a number of factors, and its efficacy may be very unpredictable. The KI technique achieves deflection by transferring momentum to the asteroid via a single impact, but the precise ΔV imparted to the asteroid is uncertain due to uncertainty in the momentum enhancement "beta" factor, and there is the chance of disrupting the asteroid into multiple pieces making the outcome highly unpredictable (Barbee et al., 2018).

Given the uncertainties with the NEO's characteristics, it would be highly desirable to precede it with a rapid-response reconnaissance mission. Preferably this would be a rendezvous mission in order to provide a good estimate of target mass, its precise trajectory (if not already known), and to assess complicating factors such as asteroid shape and number of satellites (referred to henceforth as "binarity"). A rendezvous mission could also remain on station as an observer, and measure the achieved velocity change, confirming the deflection, and post-verifying integrity.

A fundamental limitation of KI deflection is that its direction is determined by the intercept geometry and may be far from optimal. This handicap cannot be easily overcome by simply hitting the asteroid harder because of launch vehicle (LV) limitations and the risk of disruption. Furthermore, as a consequence of orbital mechanics, KI deflection suffers from a "handedness handicap" that makes deflection in one direction much harder to produce than deflection in the opposite direction. As a result, if the predicted impact is at an unfavorable location on the Earth disc, a KI deflection may be forced to move the asteroid trajectory across a long chord of the Earth disc, while another technique could choose to move the trajectory across a much shorter path. As a result, KI deflection may have to be sized to provide a larger deflection than other methods.

3.3 NUCLEAR

Although there is not a great deal in the public literature, the possibility of using the radiation from a nuclear blast in close proximity to an asteroid to deflect its orbit has been known for at least thirty years (Ahrens & Harris, 1992). This technique has the advantage of allowing a much higher amount of momentum to be transferred to the asteroid than is feasible with a single KI (Miller & Dearborn, 2015). Due to the importance of precisely controlling the trigger stand-off distance, it would be strongly preferable to deliver the device as part of a rendezvous mission as opposed to a hypervelocity intercept, which may appear excessively risky. Also, a rendezvous nuclear deflection could push the asteroid in the optimal direction, which may be not be feasible with a flyby nuclear deflection.

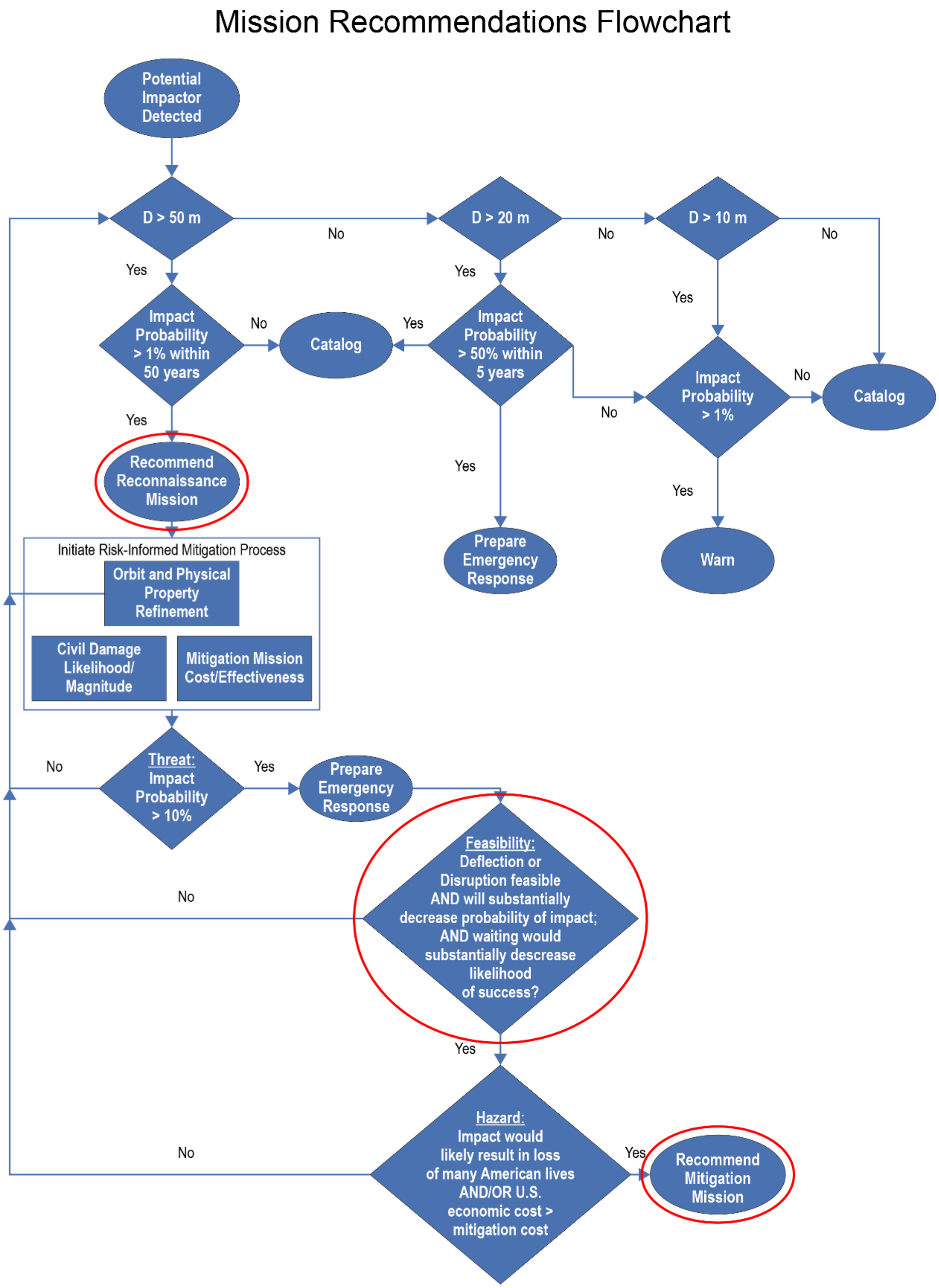


Figure 3-1. Proposed decision tree for consideration of characterization / mitigation missions (Interagency Working Group on NEO Impact Threat Emergency Protocols, 2021). This study report is intended to support risk reduction at the key decision steps shown in red circles.

3.4 ION BEAM DEFLECTION

Although the available forces are small, Ion Beam Deflection (IBD) has been proposed as a natural consequence of using electric propulsion thrusters to provide a slow, controlled deflection (Brophy et al., 2018; Bombardelli et al., 2019). This has not been examined extensively by the PD community so we provide simulations of its effectiveness in §3.5. IBD inherently requires rendezvous and extended proximity operations; this carries the benefits of detailed characterization and a very controllable deflection not available with the other techniques. Gravity tractor (GT) performance is enveloped by the IBD performance shown in Figure 3-2, although it is much less efficient and more complex than IBD (§4.2.6) (Brophy et al., 2018), so is not shown separately.

3.5 COMPARISON OF DEFLECTION TECHNIQUES

We combine modeling of KI and IBD, including disruption limitations similar to previous work by Miller & Dearborn (2015), but now using realistic asteroid trajectories and full-up orbital simulations of deflected trajectories to yield a stochastic data set of discrete realizations (Figure 3-2). All cases assume a single Falcon Heavy launch and accurately compute the deflection capability at each time in the simulation (Appendix C). The results, across a sampling of 15 impacting asteroid trajectories, indicate not only that KI and IBD have similar overall performance envelopes, but also that the risk of asteroid disruption is a significant limitation for successful KI deflections over a large region of the parameter space. These KI simulations assume no momentum enhancement by ejecta (beta=1); the region of KI disruption would only grow for larger values of beta. Less obvious in Figure 3-2 is that depending on the precise threshold assumed for disruption, for many asteroid orbits the required deflections are so large that a KI mission cannot successfully deflect without also disrupting (a “pure” deflection), regardless of asteroid size, even with 3 decades of warning. For example, assuming a 10% disruption threshold (deflection $\Delta V<10\%$ of escape velocity), only half of the cases simulated provided any KI pure-deflection capability over the 35-year simulation (of course, multiple smaller KI deflections might achieve success, but this much more complicated approach violates the single-mission assumption). An important caveat on our analyses of KI disruption is that we assume a simple disruption criterion in terms of an asteroid’s surface escape velocity, and apply this across all asteroid sizes. In reality, the disruption threshold for small (< 100 m diameter) asteroids is unknown and may differ significantly from that of larger asteroids, which are more likely to be rubble piles. In a real mitigation scenario, even if the threatening asteroid has been fully characterized by a prior recon mission, the asteroid’s disruption threshold may be unknowable ahead of an attempted impulsive deflection. As asteroid diameter grows in Figure 3-2, the mass grows with the cube of the diameter so both IBD and KI become performance limited in the range of 100–300 m diameter targets with 10–30 years of warning time, the precise maximum depending on the asteroid orbit.

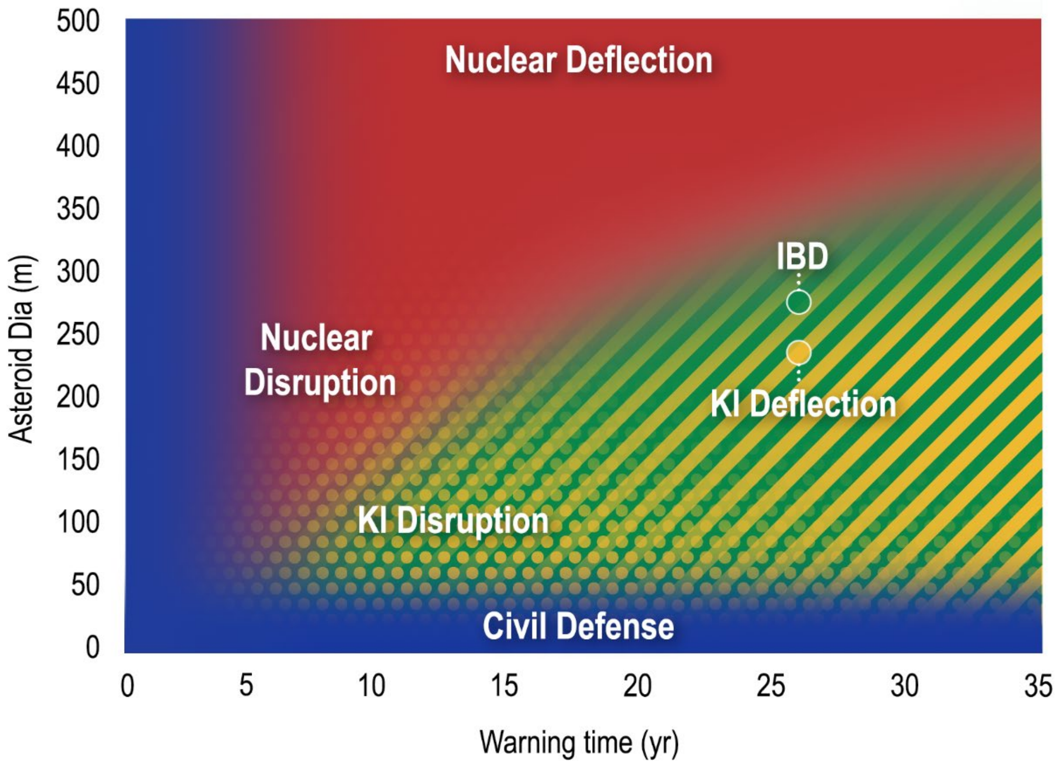


Figure 3-2. Numerical results from simulating deflection capabilities of various deflection techniques across a variety of asteroid sizes following fifteen different Earth-impacting orbits. KI techniques (yellow) largely overlap the region where IBD is effective (green). Assuming a 10% disruption threshold, only half of the KI scenarios offered any pure-deflection capability within 35 years of impact (area of solid yellow stripes). If the warning time is very short and/or the asteroid relatively small, deliberate KI disruption may be the only viable non-nuclear technique (yellow dots). See Appendix C for details.

KI and IBD deflection technologies present different reliability considerations. While less mature, IBD may eventually offer more robust defensive capabilities than KI. For example, KI mission risk is compounded if multiple launches and impacts are required, leading to changes in target physical properties (integrity) and surrounding environment (dust and debris), and the modified trajectory would become increasingly unpredictable with each successive impact. For this reason, we limited deflection modeling (Appendix C) to what is feasible with a single high-performance LV (Falcon Heavy Expendable) and assumed no ejecta enhancement of momentum transfer. Time permitting, KI deflections on successive asteroid orbits would be feasible however (Appendix C).

For short warning times, KI may achieve adequate deflection several years sooner than IBD, but KI would likely disrupt the asteroid, while IBD would not. The net effect of this is roughly a wash between the two technologies (Figure 3-2).

For long warning time cases, deflection via a rendezvous and "slow-push" method is more robust and tolerant of flight system faults and problems, because in that case the deflection occurs over much longer time scales and allows substantial time for fault diagnosis and recovery that otherwise is impossible with a KI. Conversely, "slow push" missions must operate successfully for many years to achieve success, in comparison to single-event KI missions (IBD missions were limited to 15-year lifetimes in the simulations). Slow-push deflection may also be more robust to unexpected target characteristics (e.g., rubble pile) than KI, which may yield unexpected/undesirable results, but may be limited for certain situations (e.g., binary objects). Further, KI deflection allows little choice of the intercept geometry, and is constrained by the approach phase angle, so may or may not be capable of producing deflection in the optimal direction. In comparison, slow-push deflection can be applied in almost any direction and is more robust in that sense.

If delivered via rendezvous, nuclear deflection also offers a high confidence of mission success and as with the slow push, may be applied in any direction and is relatively robust to target characteristics (Bruck Syal et al., 2013). It also offers the option of relatively safe disruption if necessary (Barbee et al., 2018). For the most-stressing short warning time cases and larger targets, a Nuclear Explosive Device (NED) is the only viable option (top portion of Figure 3-2). In this case, delivery via rendezvous would be strongly preferred for reliability reasons, assuming adequate time is available.

For these reasons, long-term development/demonstration of slow-push IBD (and nuclear deflection) technologies are an important path towards an optimum mitigation strategy that ultimately would provide the kind of confidence appropriate for real-life threats to the Earth given adequate warning time. Since these strategies inherently require rendezvous, a high ΔV capability most likely using Solar Electric Propulsion (SEP) is also implied as used by the Dawn mission to rendezvous with both Vesta and Ceres (Rayman et al., 2007). This same type of propulsion system can also be used for the deflection (Brophy et al., 2018) and is considered in the last family of cases presented in §4.2 and our recommendations in §7.

4 ARCHITECTURE TRADE STUDY

Our trade study covers all requested cases of NEO characterization missions (flyby reconnaissance, rendezvous, and tours) and multiple instances of all requested mitigation technologies: Nuclear Explosive Device (NED) simulator, kinetic impactor, and "slow push" techniques (e.g., Ion Beam Deflection, Gravity Tractor). To maintain generality, study inputs are defined parametrically; for example, we do not model specific trajectories to specific targets but rather envelope large families of such. The depth of these concepts is in-between concept maturity level (CML) 3 and 4 (Wessen et al., 2013), which is appropriate for this study. This implies some risk in the scope of the system design and consequently some cost risk which is mitigated by the assumption of 30% reserves in the costing.

4.1 PARAMETRIC INPUTS

4.1.1 MISSION DESIGN

In light of the parametric nature of this study, we do not develop specific trajectory designs with specific launch dates and specific targets. Rather, we have reviewed previous studies and performed additional analyses to identify general families of mission design drivers (C3 and ΔV) that envelope the majority of cases within each mission type (Table 4-1). Since the NEO population is quite large, we are allowed to select targets that are readily accessible in terms of C3, ΔV, and Time of Flight (TOF) to reduce the cost of these demo missions. It is also notable that although the trajectories considered herein were applied to a NEO population up to ~300 m in diameter, smaller targets (e.g., < 100 m in diameter) would be most valuable for characterization because very little is known about them.

4.1.1.1 Candidate Tours

Following the approach in Papais et al. (2020) we used the JPL Star tool to assess the opportunity for tours of four NEOs. In particular, we reduced the large NEO population to the size range of interest (<300 m in diameter inferred from the respective absolute magnitudes). This yields a pool of approximately one thousand targets. Our simulation then found 557 converged trajectories, with a launch energy (C3) ranging from 0 to 7.9 km^2/s^2, velocity change (ΔV) from near-zero to 1.28 km/s, and TOF from 2.1 to 4.5 years. If we filter these results to be compatible with a low-cost system design (low C3 and ΔV and shorter TOF) then we see the results in Figure 4-1. This demonstrates that a C3 <2 km^2/s^2 and ΔV of <250 m/s offers between 15 and 20 potential tours each year. This provides an adequate menu to select combinations of NEOs that would provide measurements of interest to PD demonstration objectives.

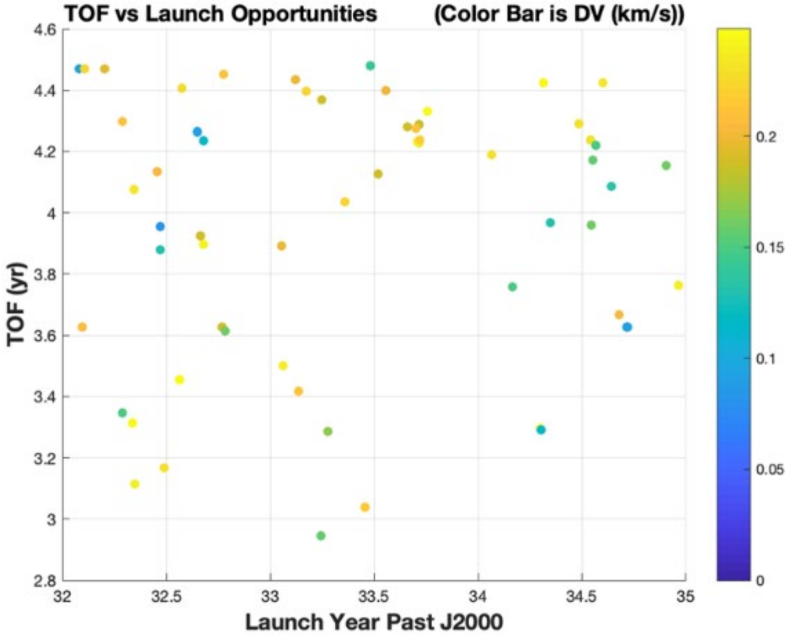
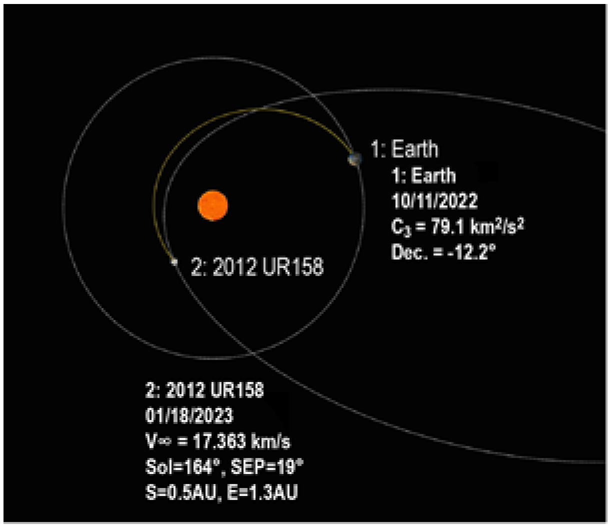


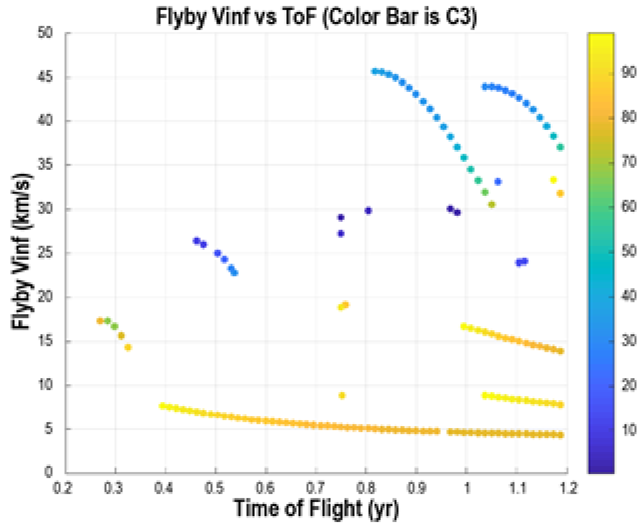
Figure 4-1. Candidate tours with flybys of four NEOs and C3 < 2 km^2/s^2

4.1.1.2 Candidate Recon/Intercept Trajectories

In "short warning" scenarios (Appendix C) the ability to (1) improve the orbit solution (2) estimate the size/mass of the asteroid, and (3) verify the number of bodies is critical to developing an effective response (JPL CNEOS, 2019a). This creates an unusual trade space where TOF is paramount, even at the expense of high C3 (and launch costs) and high flyby velocity (Vinf) (Figure 4-2). For operational missions, this means that a wide range of LVs should be available, and that the spacecraft and instruments should be capable of acquiring the needed information at high relative velocities. While high-performance LVs are mature, successful collection of high-resolution imagery at these relative velocities has not been demonstrated and will stress narrow angle camera (NAC) pointing, and Guidance, Navigation, and Control (GN&C) capabilities. Therefore, it is important for the Flyby/Recon demo to include mission design to create an encounter at high velocities (e.g., >15 km/s). This can be accomplished via careful selection of a sample target asteroid, without requiring a high C3 (e.g., the purple solutions near the center of Figure 4-2B). The selected target should also be in the 100 m diameter size class, have an orbit that is known well enough to allow a near-ballistic trajectory, and offer an intercept geometry that has a solar phase angle adequate for optical navigation and imaging.



(A) Example of minimum TOF solution (left-most side of Figure 4-2 (B).



(B) Intercept trajectory trade space

Figure 4-2. Example of intercept trajectory design for fast reconnaissance

4.1.1.3 Candidate Rendezvous Trajectories

From Papais et al. (2020), we see that a ΔV of ~ 2 km/s may be minimally-sufficient to provide rendezvous capability with some potential candidate NEOs (Figure 4-3). So, we use this number as the lowest-cost demonstration option and also consider higher ΔVs to provide a larger range of options at low risk (Table 4-1).

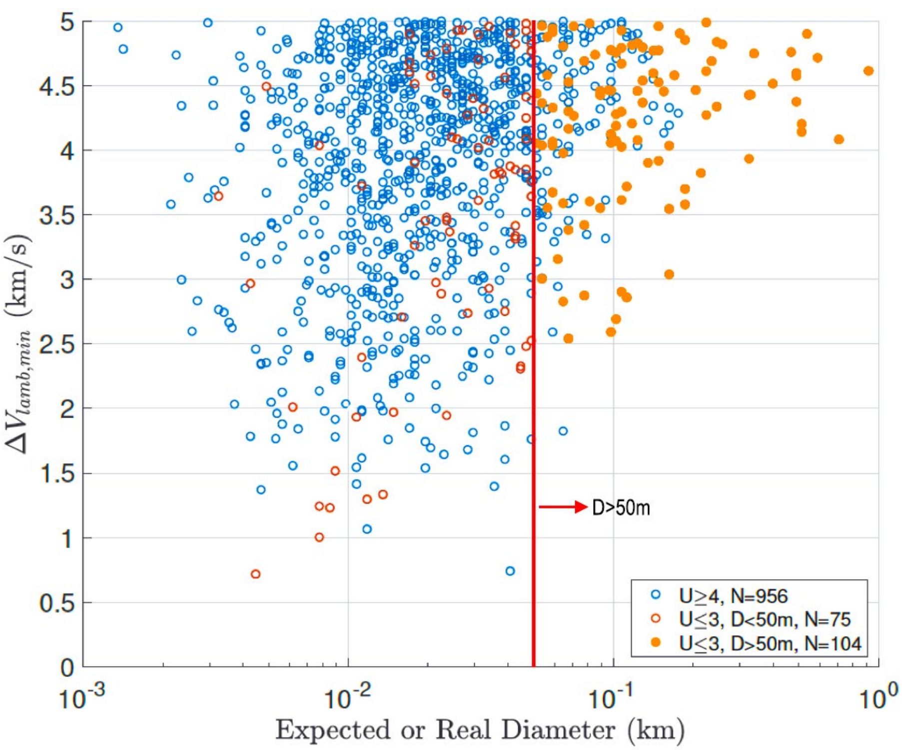


Figure 4-3. Asteroid rendezvous ΔV statistics from Papais et al. (2020). U is current orbit uncertainty code; if U<4 then there is higher confidence in predicted ΔV requirements for each case.

Table 4-1. Parametric ΔV envelopes used based on above trajectories and statistics. C3=2km^2/s^2 in all cases.

Mission Type	ΔV cases		Notes
	Minimum ΔV	Nominal ΔV	
Flyby Recon	250 m/s	---	May be designed as ballistic trajectory with minor clean-up burns
Rendezvous	2 km/s	4 km/s	Higher ΔV case allows rendezvous with large fraction of NEO population
Flyby Tour	250 m/s	---	May be designed as ballistic trajectory with minor clean-up burns
Intercept	1 km/s	---	Provides flexibility over minimum requirement of ~250 m/s
Kinetic Impact	500 m/s	6 km/s	Higher ΔV case allows SEP-based observer
Ion Beam / Gravity Tractor	2 km/s	---	These cases include additional fuel to perform deflection

4.1.2 LAUNCH VEHICLE ASSUMPTIONS

The dedicated missions studied typically require direct launches due to the specific timing and targeting inherent in PD. In rare cases it may be possible to find a rideshare with a compatible primary mission, but this requires loss of control of launch date and targeting that may become unacceptable, and also imposes strict limitations on spacecraft mass and volume. For the broad range of missions considered in this study, we exclude secondary launches, but note that in some very specific cases this may be a valid alternative.

The baseline LV assumed for the study is the Falcon 9 Return to Launch Site (RTLS), which currently offers 1593 kg launch capability to the baseline C3 of 2 km^2/s^2. This offers more than enough performance for most cases of interest in this study. For the few that require more launch mass the ASDS (Automated Spaceport Drone Ship) option almost doubles the baseline performance with a negligible cost increase. We assume that the Falcon 9 cost of $75M (slightly more than for DART) can be made possible with use of previously flown LV elements, shorter schedules, and reduction of standard services from the NASA Launch Services Program (LSP).

Since the assumed C3 is low and many of the missions studied require much less than 1000 kg launch capability, we have also surveyed the potential for lower-cost/performance LVs that may becoming available in the appropriate time frame. There are presently at least three such vehicles in development (Firefly Beta, Rocket Lab Neutron, and Relativity Space Terran 2) that could provide ~1000 kg launch mass with the addition of an upper stage. Also extrapolating from their current costs, we budget $44M for these cases.

4.1.3 TELECOM SIZING

Flyby options were assumed to produce 30 Gb of total data volume required (per target, in the case of the 4-NEO tour). This led to a 1m deployable high-gain antenna (HGA) plus an Iris radio (Shihabi et al., 2019) which would trickle back the 30 Gb of flyby data (plus an assumed 30% overhead) at 20 kbps at an average Earth range of 1AU over 3 months per target.

All rendezvous options were assumed to produce 400 Gb of total data. This led to a telecom subsystem upgrade with a 25W Solid State Power Amplifier (SSPA) rather than using the Iris radio's default 4W SSPA. With the same 1m deployable HGA, it would achieve a data rate of 100 kbps at an average Earth range of 1AU, enabling it to return the 400 Gb of data (plus 30% overhead) over about 7 months.

For options which required spacecraft-to-spacecraft relay, the mother spacecraft was assumed to have an Electra-Lite UHF relay radio; the daughter spacecraft (if modeled explicitly) was assumed to have a CubeSat UHF radio. Where the daughters are also flyby spacecraft, the mother spacecraft's data return requirements are assumed to double to 60 Gb per target. A 10 W SSPA is assumed, to double the return rate to 40 kb/s, and the data return time is kept the same at 3 months per target × 4 targets = 12 months.

4.1.4 REPRESENTATIVE INSTRUMENTS

For each study case, we have reviewed previous and current instrument designs and selected representative concepts (Table 4-2) that are believed to offer performance in accordance with the directions in the study request and PD characterization and mitigation objectives. Cost estimates of these were subsequently run using NASA Instrument Cost Model (NICM) (Appendix E).

Table 4-2. Representative instrument sizing conservatively based on previous/current missions

	Mature Mass* (kg)	Average Power (W)	Applicable cases	Basis
Vis NAC	12	17	1, 7–10, 13, 14	Average of three similar from MSSS
Vis/NIR Spec	7	12	1–9, 11, 12, 16	JPL MLPS + optics (JPL point spec. development)
Vis WAC	5	10	2–6, 10–16	Average of two similar designs
Radar 1 (HFR)	7	137	4	Hera heavy
Radar 2 (LFR)	5	50	6, 10, 12, 16	Hera bistatic (light)
LIDAR	16	31	4, 6	OSIRIS-REx LOLA
CubeSat cam	4	14	9, 13 DART	MSSS Jcam

**Additional growth factor allowance was added in addition to using mature masses from referenced missions*

4.1.5 PROJECT SCHEDULE, RESERVES, AND MISSION ASSURANCE ASSUMED FOR COSTING

We assume that flight projects less than $500 M are preferred (including launch and Phase E/F). Using DART as a representative case study, we have assumed a NASA risk classification of C (moderate risk tolerance and medium priority) in accordance with the demonstration nature of the missions addressed (NPR 8705.4). This implies that the spacecraft designs discussed in this section are single-string to minimize cost and mass. As a practical matter of implementation, there are many shades of grey between risk classifications, and projects have freedom to customize according their particular risk posture (e.g., up screening of electrical, electronic, and electromechanical [EEE] parts). Similarly, some selective redundancy would probably be implemented on a case-by-case basis (e.g., 3-of-4 reaction wheels), but this is below the level of detail studied.

We assume an aggressive 54 month schedule from project start to launch for the flyby missions. Rendezvous missions are generally more complex so we add an additional 3 months to phase B and 12 months to phase E for those cases to be more in family with typical schedules (Table 4-3). Rendezvous cases were incremented by $22 M (the average annual Phase E costs) to cover this additional year of operations. Options for much shorter schedules for operational situations are discussed in §6.

Project cost reserves are assumed to be 30% for Phase A-D, and 15% for Phase E/F.

Table 4-3. Representative project schedules

Mission Type	Phase A	Phase B	Phase C	Phase D	Phase E/F
Flyby	9 mo.	9 mo.	20 mo.	16 mo.	36 mo.
Rendezvous	9 mo.	12 mo.	20 mo.	16 mo.	48 mo.

4.2 STUDY RESULTS

Trade study cases considered are responsive to the panel's study request, including numerous second-order variations. The Flyby/Reconnaissance mission was moved to the top of the list because it is the simplest and considered a good place to start the analyses. The cases fall into the families described in the following subsections, as defined in the study request and listed in Table 4-4. These general descriptions lead into the design trade space documented in Appendix E, with the costs and other features as summarized in below.

The concepts described in §4.2 were all modeled by JPL's Team X, using the assumptions and constraints shown in §4.1. This provides realistic system sizing and cost estimations, without getting into excessive design details. The Team X data base is built on existing hardware and flight project actual costs, so inherently assumes high maturity subsystem design and conservative costing (this is revisited in §4.5).

The cost modeling is performed using standard, calibrated tools (Hogstrom et al., 2019), seeded by first-order spacecraft sizing (Nash, 2020) and baseline instrumentation costing using the NASA Instrument Costing model (NICM) (NASA OCFO, 2020) at the 50% confidence level.

As a special calibration for this PD study, we tested the costing using relevant, known past projects (including but not limited to Deep Impact, DART, Janus, and OSIRIS-REx) and demonstrated matches to within 10–20% of the known/expected project costs. The cost estimates shown in this section are therefore considered to be reasonably accurate on an absolute scale, and certainly accurate in a relative sense.

4.2.1 SYSTEM SIZING AND COST ESTIMATES

Table 4-4 shows the inputs and some key intermediate variables derived for each of the cases, and the estimated project cost. Additional detail is available in Appendix E. Not shown in the table but included in the study are power, propulsion, and communications subsystem sizing; these all feed into the mass estimates shown in the table and hence into the cost modeling. A brief description of each family, and specific option, follows.

Table 4-4. Summary of Sizing and Costing Inputs and Outputs

Option #	Name	Instruments	ΔV	Spacecraft Bus Mass (Margined)			Stack Mass	Mission Cost
				Dry	Propellant	Bus Total		
			m/s	kg	kg	kg	kg	$M FY2025
1	Flyby Recon	NAC, Point IR Spec	250	155.4	25.9	181.3	201.0	199
2	Rendezvous	WAC, Point IR Spec	2000	347.7	378.6	726.3	738.6	352
2-E	Rendezvous SEP	WAC, Point IR Spec	2000	284.3	51.3	335.6	347.9	310
3	Rendezvous high ΔV	WAC, Point IR Spec	4000	333.9	119.7	453.6	465.9	349
4	Rendezvous Radar-Lidar	WAC, Point IR Spec, HFR, LIDAR	2000	457.1	517.7	974.8	1010.5	597
4-E	Rendezvous SEP Radar-Lidar	WAC, Point IR Spec, HFR, LIDAR	2000	304.1	58.7	362.8	398.5	465
5	Rendezvous high ΔV BigSat	WAC, Point IR Spec	4000	446.0	136.0	582.0	594.3	434
6	Rendezvous 2FE	WAC, Point IR Spec, LFR, LIDAR	2000	481.1	559.4	1040.6	1092.2	626
6-E	Rendezvous SEP 2FE	WAC, Point IR Spec, LFR, LIDAR	2000	295.2	59.9	355.1	406.8	472
7	4-NEO Tour	NAC, Point IR Spec	250	155.4	25.9	181.3	201.0	206
8-1	Tour Multiple (1xB)	NAC, Point IR Spec	250	155.4	25.9	181.3	526.7	279
8-2	Tour Multiple (2xB)	NAC, Point IR Spec	250	155.4	25.9	181.3	759.9	339
8-3	Tour Multiple (3xB)	NAC, Point IR Spec	250	155.4	25.9	181.3	953.5	430
9	Tour CubeSats	NAC,Point IR Spec	250	290.7	65.1	355.8	505.5	363
10-MP	Intercept (Monoprop)	NAC, Point IR Spec, LFR	1000	287.0	207.0	494.0	528.5	352
10-BP	Intercept (Biprop)	NAC, Point IR Spec, LFR	1000	298.3	148.0	446.2	480.7	359
11	Rendezvous wNED	WAC, Point IR Spec	2000	430.9	478.4	909.3	933.9	438
11-E	Rendezvous SEP wNED	WAC, Point IR Spec	2000	308.3	57.5	365.8	390.4	326
12	Rendezvous wNED (2E)	WAC, Point IR Spec, LFR	2000	458.7	512.6	971.3	1000.9	512
12-E	Rendezvous SEP wNED (2E)	WAC, Point IR Spec, LFR	2000	299.2	56.9	356.1	385.6	376
13-DI	Kinetic Impact (DI)	NAC (2)	500	243.6	83.6	327.2	648.8	361
13-DART	Kinetic Impact (DART)	NAC, CubeSat cam	500	388.4	100.8	489.2	524.8	298
14	Kinetic Impact (SEP obs)	NAC, WAC	6000	345.1	194.6	539.7	1052.6	621
15-A-M	Ion Beam (MaSMi)	WAC	2000	329.6	88.4	418.0	422.9	336
15-B-M	Ion Beam (MaSMi)	WAC	2000	412.2	240.8	653.0	658.0	390
15-C-M	Ion Beam (MaSMi)	WAC	2000	677.5	832.1	1509.6	1514.5	598
15-A-S	Ion Beam (SPT-140)	WAC	2000	538.5	120.4	658.9	663.8	499
15-B-S	Ion Beam (SPT-140)	WAC	2000	576.4	266.1	842.5	847.4	555
15-C-S	Ion Beam (SPT-140)	WAC	2000	758.4	827.6	1586.0	1590.9	666

Option #	Name	Instruments	ΔV	Spacecraft Bus Mass (Margined)			Stack Mass	Mission Cost
				Dry	Propellant	Bus Total		
			m/s	kg	kg	kg	kg	$M FY2025
16	Gravity Tractor	WAC	2000	387.0	190.7	577.7	594.9	430
17	Ion Beam & Gravity Tractor	WAC	2000	389.1	217.8	606.8	624.1	443

4.2.2 FLYBY RECONNAISSANCE MISSION CONCEPT

In the short-warning time scenario (Appendix C), it may be critical to launch a reconnaissance intercept mission on a very short time scale, with one or more mitigation missions to arrive shortly thereafter. This class of quick-look recon would be intended to estimate the mass by way of shape modeling, inferring meteoritic analogs, or to measure the mass directly by other means (Christensen et al., 2021). This requires high resolution imagery (1-10 m sample distance), which is the product of the angular size of a NAC pixel and the flyby range. (For example a 10 μr NAC resolution produces 1 m image resolution at 100 km range; but with this geometry at 15 km/s the angular rate is 8.6 deg/s, requiring either a gimbaled camera or a highly-agile spacecraft attitude control.)

Another key objective is an improved orbit solution via collection of in situ OpNav data; a tertiary objective is to see whether the newly-discovered target is a singular object or in fact a binary, or if there were anything anomalous which would affect mitigation. The critical operational value of these parameters is made clear in the PDC exercises discussed in Appendix C.

The assumed trajectory and ΔV requirements, and encounter closing velocities are as studied parametrically (§4.1.1). This kind of high-velocity intercept (whether a recon flyby or KI) requires autonomous navigation as demonstrated on Deep Impact (Bhaskaran, 2012) and proposed for other scenarios (Chesley et al., 2013) and soon to be performed on DART with a smaller target at lower relative velocity (§2.2). For new demos, it is important to include mission design to create an encounter at high velocities (e.g., >15 km/s) (§4.1.1). A high degree of cross-track agility (high-impulse thrusters) is also required for these cases.

Primary instrumentation for these missions is a narrow angle camera (NAC) and point IR spectrometer, to help constrain bulk density (and hence mass) via meteorite analog type.

Option 1: Fast Flyby Recon: Simple recon mission, with just a NAC and a point IR spectrometer. This mission should be achievable with SmallSat-class spacecraft. We assumed that all avionics were single-string, and used commercial off the shelf (COTS) SmallSat components wherever possible. The concept requires only 250 m/s of ΔV, which is easily achievable with a blowdown monoprop system. For telecom, we assumed a 1m deployable high-gain antenna (HGA) plus an Iris radio, which would return the 30 Gb of flyby data at 20 kb/s at an Earth-spacecraft (S/C) range of 1 AU over about 3 months. Note that:

- These avionics assumptions (single-string, SmallSat) were applied to almost all other options (excluding Option 5)
- The telecom design (1m deployable HGA + Iris → 20 kb/s @ 1AU range) was assumed in almost all flyby options (excluding Option 9)
- If faster data return is required (due to planetary defense data latency needs), the Telecom design could be switched to be the same as in Option 2, with a mass and cost increase (~20 kg wet margined mass increase, ~$13 M total mission cost increase), to get all the data back in 2 weeks.
- The cost of this option was decremented by 1 year of Phase E costs ($7 M) to reflect the short TOF inherent in this option (§4.1.1).

4.2.3 RENDEZVOUS CONCEPTS

Rendezvous missions with small bodies have been studied extensively (Müller et al., 2017; Bambach et al., 2018; Venigalla et al., 2019; Papais et al., 2020) and been performed many times by various organizations. Rendezvous provides the ability to perform in-depth characterization by any number of methods, and also potential mitigations, with very good ability to meet many of the PD objectives. We consider many different implementations of rendezvous missions (§4.1.1), including high and low ΔV using chemical and Solar Electric Propulsion (SEP), multiple flight elements including deployed landers, and NED demonstrations. Instrumentation considered includes visible cameras, LIDAR, and RADAR (Table 4-1). A point IR spectrometer is assumed to meet minimal requirements; however, for these rendezvous missions an imaging IR spectrometer such as that being developed for the Lunar Trailblazer SIMPLEx mission may be a cost-effective upgrade. Gravity field measurements are also considered when there are two flight elements (Hesar et al., 2015), or can be estimated via a series of slow approaches as done on OSIRIS-REx and Hayabusa2.

As demonstrations, the drawbacks to these missions are that they are more expensive than other options and generally limited to visiting a single target. However, in a real scenario, a rendezvous mission is strongly preferred as it can perform both characterization and mitigation, and is inherently robust (Appendix C). Rendezvous missions also could be combined with tour concepts (§4).

Option 2: Rendezvous: Rendezvous mission, using a biprop chemical propulsion bus to provide 2km/s of ΔV. This option uses the same simple payload suite as Option 1 (NAC + point IR spec), but returns a larger data volume (400 Gb) due to a longer time spent at the body. We assumed that the telecom subsystem from Option 1 could be upgraded with a 25 W SSPA rather than using the Iris radio's default 4 W SSPA. With the same 1m deployable HGA, it would achieve a data rate of 100kb/s at an Earth-S/C range of 1AU, enabling it to return the 400Gb of data over about 7 months. Total margined launch mass was a bit under half of the Falcon 9's capability, at 739kg. Note that this telecom design (1m deployable HGA + Iris + 25 W RF SSPA for 100 kb/s @ 1AU range) was assumed in all rendezvous options (excluding Option 5).

Option 2-E: Rendezvous SEP: This was a SEP variant on Option 2. A rendezvous mission with a NAC + point IR spec, using a SEP spacecraft, providing 2 km/s of ΔV. This comes in as lower mass (and therefore lower estimated cost) than the biprop Option 2. Assumes use of a single MaSMi Hall thruster (Conversano et al., 2017), thrusting for 0.6 years at 85% duty cycle, consuming 779 W current best estimate (CBE) and with an I_{sp} of 1542 sec. This S/C used rigid solar arrays, assuming 87 W/kg at end of life (EoL) at 1AU, rated at 1414 W, and sized for spacecraft power requirements while simultaneously thrusting with the electric propulsion (EP) system and transmitting with the telecom system.

Option 3: Rendezvous high ΔV: A rendezvous mission, with the same simple NAC + point IR spec payload as Options 1 and 2, but assuming a higher ΔV of 4 km/s. Uses a SEP bus, and assumes two MaSMi Hall thrusters operating simultaneously to keep the burn duration under 1 year. Total burn duration of 0.8 years, at a duty cycle of 85%.

Option 4: Rendezvous Radar-Lidar: A rendezvous mission, with a biprop system providing 2km/s ΔV like in Option 2, but where the payload includes a Radar and a Lidar. The cost of the payload (and to some extent the ripple of the added mass and power through the S/C bus) pushes the cost over $500 M.

Option 4-E: Rendezvous SEP Radar-Lidar: A SEP variant of Option 4. Using SEP reduces the mass, and therefore the estimated cost. Like in Option 2-E, uses a single MaSMi thruster, at 779 W CBE power, and has rigid arrays sized to 1414 W. Burn time is 0.7 years at a duty cycle of 85%.

Option 5: Rendezvous high ΔV BigSat: Like Option 3 (high ΔV of 4 km/s, simple NAC + point IR spec payload, SEP design), but does not use SmallSat components, and uses a single SPT-140 thruster rather than a MaSMi. Note that this was the only option to use "traditional" footprint avionics rather than SmallSat components. All avionics were still single-string. Since the SPT-140 requires more power than a MaSMi, the arrays were larger at 7205 W, and we therefore used low-mass arrays to save mass and ultimately cost (see Design Assumptions section). The higher thrust of the SPT-140 means it only needs to burn for 0.3 years to achieve 4 km/s of ΔV. This bus was higher mass than in Option 3, but the mission still came in under the cost cap at $484 M. Note that using MaSMi rather than an SPT-140 would bring the est. mission cost down to $422 M.

Option 6: **Rendezvous 2FE**: Like Option 4 (rendezvous with Lidar and Radar), but now with an additional small deployed lander as well. The deployed lander was assumed to have a mass of 15 kg and cost of $44 M including its instruments. The instrument was assumed to be equivalent to the Hera Light radar.

Option 6-E: Rendezvous SEP 2FE: A SEP variant on Option 6. Using SEP brings down the estimate mass and therefore cost. As with Option 2-E and 4-E, the design used a single MaSMi thruster running at 779W CBE, with arrays sized to 1414 W EoL. Total burn time to reach 2 km/s of ΔV was 0.8 years at a duty cycle of 85%.

4.2.4 FLYBY TOUR CONCEPTS

In contrast with rendezvous missions, flyby tours provide much less detailed information but provide information about multiple targets. These tours have also been studied extensively (Rivkin et al., 2016). Here, we seek to constrain the suite of physical characteristics and compositions among the NEO population that may be relevant for Planetary Defense considerations and inform mitigation techniques (especially those <140 m in diameter). While straightforward and less expensive than other concepts, such missions have not yet been flown in the NEO regime. For system sizing/costing purposes, we assume the parametric trajectory information shown in §4.1.1 and flybys of four NEOs. Required instrumentation includes NAC and IR point spectrometer (Table 4-2). Variations include use of 1, 2, 3, or 4 spacecraft, or the addition of CubeSats; in all cases the objective is to collect imagery for shape model development, and data relevant for compositional assessment (e.g., mass).

Option 7: Flyby Tour: Tour of four selected NEOs per §4.1.1.1. This can be done with low ΔV of 250 m/s. Spacecraft and instruments are the same as Option 1 (NAC and IR Spec.)**.** However flyby closing velocities in this case are typically 6–8 km/s instead of the 15 km/s in Option 1.

Option 8-1: Tour Multiple (1xB): Launches two flyby spacecraft together, on a single launch; both spacecraft follow the same multi-body tour trajectory, and nominally fly by 4 distinct bodies. At each body, they each follow very slightly different trajectories to provide alternate perspectives (e.g., on different sides of the body). Spacecraft A has a NAC as well as a point IR spectrometer. Spacecraft B has only a NAC. Both S/C are assumed to launch together with a Dual Payload Adapter, though equally feasible using an ESPA ring. The data volume is assumed to be 30 Gbit per spacecraft per flyby, and it is assumed that there is time between and after the flybys to relay it all back, which would take (3 months/target) x (4 targets) = 12 months total.

Option 8-2: Tour Multiple (2xB): Like 8-1, but increases the quantity of Spacecraft B to 2, for a total of 3 spacecraft. All are assumed to launch together on an ESPA ring.

Option 8-3: Tour Multiple (3xB): Again, increasing the number of copies of Spacecraft B, now to 3, for a total of 4 spacecraft. All are assumed to launch together on an ESPA ring.

Option 9: Tour CubeSats: A multi-target "tour" concept like Option 8. A Mothership carries 4x 12U CubeSats, and launches one during each of 4 body flybys, to give an extra vantage point for the flyby. The CubeSats relay data back to the mothership via a UHF link. The CubeSats were assumed to have a margined wet mass of 29 kg and a first-unit cost of $14 M. The data return requirement for the Mothership would approximately double, so a 10 W_{RF} SSPA was assumed, to bring the data rate to 40 kb/s @ 1AU range and keep the downlink time to 3 months/target. Each CubeSat was carried in an 11 kg (margined) and ~$200 k dispenser.

4.2.5 INTERCEPT AND KI CONCEPTS

In the short-warning time scenario (Appendix C), it may be critical to launch an intercept mission on a very short time scale. This may include a flyby for reconnaissance and/or one or more KI mitigation missions (JPL CNEOS, 2019a). The success of hypervelocity impacts carries some inherent risk, and new analysis of their efficacy is presented in Appendix C.

Hypervelocity Kinetic Impact missions have also been studied extensively (Hernandez & Barbee, 2012; Bhaskaran & Kennedy, 2014; Dearborn et al., 2020) and executed previously (Frauenholz et al., 2008). We consider several implementations based on the NEO tour trajectories and encounters (§5.1.1), including observer spacecraft (either stationary or flyby similar to Deep Impact). Instrumentation includes WAC and NAC. KI technologies will be further demonstrated with DART launching later this year (Cheng et al., 2012; Adams et al., 2019). Future demonstrations should include intercepts at much higher closing velocities (e.g., >15 km/s). Demonstration of GN&C requirements will drive the NAC design, autonomous intercept processing, and cross-track agility.

This family of concepts also includes high-velocity intercepts with a NED; in this case the objective is not to demonstrate the explosive device, but rather the radar-driven trigger to support detonation at the desired distance a few hundred milliseconds before impact. While extreme, this scenario represents our response to a worst-case situation in which there is inadequate warning time for detailed characterization or to perform rendezvous missions, and inadequate time (or KI energy) for deflection by other means.

Option 10-MP: Intercept (monoprop): A demonstration for a NED intercept mission. The S/C would carry a non-deployable NED simulator with a trigger, and would do a very close flyby of the body, with a small radar to measure distance. It would demonstrate “detonating” with the trigger, though it would not actually contain any explosives. This sub-option uses a monoprop propulsion system to provide 1 km/s of ΔV, with a margined launch wet mass of 529 kg.

Option 10-BP: Intercept (biprop): A sub-option, like Option 10-MP but using a biprop propulsion system to provide the same 1 km/s of ΔV. The modeled total wet mass was lower (for a margined launch wet mass of 481 kg), but the dry mass was very slightly higher due to the higher complexity of the biprop system, and the modeled total cost was therefore slightly higher. There is an un-modeled propulsion cost upper that could push the cost up even further in a grass-roots estimate.

Option 13-DI: Kinetic Impact (DI): A two-element kinetic impact demo, architecturally similar to Deep Impact. A monoprop mothership spacecraft (500 m/s ΔV capability) deploys a kinetic impact spacecraft one day prior to close approach, and observes the impact. The kinetic impact vehicle carries 190 kg of additional “dumb” mass to bring its total margined mass to at least 300 kg. Note that this “dumb” mass was assigned a fixed $100 k cost, and was not fed to the bus cost model; but there are additional “taxes” on that mass, because the impactor was sized to carry it (affecting structures, propulsion, thermal, attitude control system [ACS]); and the mothership was sized to carry that impactor.

Option 13-DART: Kinetic Impact (DART): A two-element kinetic impact demo, architecturally similar to the DART mission. A monoprop mothership serves as the impact vehicle, but releases a small observer spacecraft before impact. The observer is estimated to be similar in mass to a 12U CubeSat, and it is assumed that it could fit in a 12U form factor, and released from a 12U dispenser; therefore a CubeSat design specification (CDS) board was used (rather than the SmallSat box assumed in other concepts). The mothership impactor carries 45 kg of additional “dumb” mass to bring the impacting mass above 300 kg. This option is more mass-efficient than Option 13-DI, and therefore lower estimated cost, because the naturally heavier spacecraft (mothership with propulsion) is the impactor. The total amount of “dumb” mass can be reduced vs. 13-DI, and therefore the “taxes” (structure, propulsion, thermal, ACS) to carry that extra mass are reduced, for a reduction in costed dry mass. This concept is enabled by the ability to observe and send back data from the small observer spacecraft, which requires capable pointing (using SmallSat ACS components) and a deep-space communication system in a small form factor (the same Iris + 1m deployable HGA as in other flyby concepts).

Option 14: Kinetic Impact (SEP obs): This concept is a kinetic impact demonstration that uses two spacecraft that separate from each other immediately after release from the Launch Vehicle. A SEP observer spacecraft, carrying only a Narrow Angle Camera for observing the impact, takes a 6 km/s low-thrust trajectory to rendezvous with the body. It uses 2x MaSMi engines, running simultaneously at 779 W CBE each, with low-mass (132 W/kg) solar arrays sized at 2639W EoL, and takes 1.3 years of thrusting (at a duty cycle of 85%) to achieve 6 km/s of ΔV. Meanwhile, a monoprop impactor S/C uses an impulsive (1 km/s) trajectory to target the body, and needs 25 kg of additional "dumb mass" to bring its total margined mass to over 300 kg. The "dumb mass" was increased further to 50 kg, such that it plus the predicted bus mechanical and structure mass (85 kg) would exceed the 133 kg mass of a 6-port ESPA ring. It was then assumed that the impactor spacecraft could use an ESPA ring (or similar tube structure) for the bulk of its primary structure, and could carry the load of the SEP spacecraft above it on the launch vehicle, obviating the need for a dual payload adapter (DPA). The use of two relatively high ΔV spacecraft pushes this concept over $500 M, despite modest camera-only payloads. Note that there is additional mission design work needed to show that such a concept is indeed feasible.

4.2.6 NED AND SLOW PUSH/PULL RENDEZVOUS CONCEPTS

A rendezvous mission inherently provides the opportunity for demonstration of a NED simulator. Nuclear devices have long been considered for PD purposes and may provide a highly-reliable mitigation alternative delivered via rendezvous, without prior demonstration due to the maturity of nuclear effects modeling (Bruck Syal et al., 2013; Dearborn et al., 2020) As with all rendezvous missions, detailed characterization of the NEO is available prior to the deflection activity, allowing it to be tuned to the actual target body (e.g., adjustment of nuclear stand-off range at detonation).

Rendezvous missions are generally more complex so we add an additional 3 months to phase B and 12 months to phase E for those cases to be more in family with typical schedules (Table 4-3). Rendezvous cases were incremented by $20 M (the average annual Phase E costs) to cover this additional year of operations

Option 11: Rendezvous w/NED: Carries a non-deployable NED simulator, as in Option 10; but performs a rendezvous with the body, rather than a flyby. After asteroid characterization, would make a very close approach to the body, and activate the NED simulator's trigger. Uses a biprop propulsion system, providing 2 km/s of ΔV.

Option 11-E: Rendezvous SEP w/NED: A SEP variant on Option 11. Comes in at lower mass and therefore lower estimated cost. As in Option 2-E, 4-E, and 6-E, uses a single MaSMi thruster running at 779 W CBE and I_{sp}=1542 sec, with rigid solar arrays sized to 1414 W. The total burn time is 0.7 years at a duty cycle of 85% to reach the total 2 km/s of ΔV.

Option 12: Rendezvous w/NED (2E): A rendezvous concept which includes a deployed NED simulator. The spacecraft would release the NED simulator on a trajectory towards the asteroid, and the simulator would trigger when in close proximity or contact. This exercises the simulation for precise deployment (like the Hayabusa2 small carry-on impactor experiment), retreats for safety, and then re-approaches for characterization and mitigation assessment. It is more expensive than Option 11 because of the addition of a radar, which was for characterization of the body (and not for triggering).

Option 12-E: Rendezvous SEP w/NED (2E): A SEP variant of Option 12. Switching to a SEP design brings the estimated mass down enough that the estimated cost is now under $500 M. SEP design is still a single MaSMi @ 779 W CBE, 1414 W arrays. Total burn time of 0.7 years for 2 km/s of ΔV.

Option 15: Ion Beam: (with 6 sub-options) is for an Ion Beam demonstration. It involves a SEP spacecraft, with at least two engines that can be operated simultaneously. The engines are assumed to be on outriggers, such that they can be used simultaneously for both propulsion and deflection. Ion Beam deflection is best performed with an *Ion engine*, with as tight an exhaust ion beam as possible; however, it was assumed that it can also be demonstrated (at lower effectiveness, 50% assumed) with Hall thrusters (which have wider exhaust spread). This assumption perhaps merits additional scrutiny. It was also assumed that 1 mm/s of change to the body's velocity would be detectable, and preliminary analysis (Appendix D) indicated that this would require only 32 kg of Xenon propellant from a Hall thruster (SPT-140, or MaSMi at a high throttle setting) for a demonstration on a 50 m diameter body. Because there is uncertainty in this figure, we ran three primary sub-cases, with varying Xe quantities for the demonstration: A) 32 kg B) 150 kg C) max out the F9 launch allocation (~600 kg). Further, sub-cases were run with both MaSMi engines (M) and SPT-140 engines (S), for a total of 6 sub-options. In all cases it was assumed that the ΔV budget prior to the start of the deflection demonstration was 2 km/s. All options use low-mass arrays (132 W/kg EoL). The second two MaSMi options (15-B-M and 15-C-M) add extra inactive thrusters to avoid exceeding the MaSMi's 100 kg rated throughput limit; however, if this limit is revised upwards, the dry mass and cost can come down.

Option 16: Gravity Tractor: This is a Gravity Tractor demonstration, with 100 kg of Xe for the deflection. The spacecraft configuration is assumed to be the same as in Option 15, with two MaSMi engines on outriggers. In addition to a WAC, it carries a point IR spec for characterizing the body, and a small radar to maintain spacing with the body. It is assumed that 1 mm/s of ΔV in the body is measurable; to achieve this in 1.5 years, with a low-mass (and therefore low-cost) spacecraft, the spacecraft must fly very close to the body. For the point design in this study, it must fly as close as 10 m from the surface of a 50 m diameter body with a 400 kg spacecraft (see Mission Design report for analysis description and body assumptions). In all cases it was assumed that the ΔV budget prior to the start of the deflection demonstration was 2 km/s.

To reduce the thrust (to keep the S/C from just flying away from the body), and to avoid plume impingement on the body, the two MaSMi engines must be throttled down to their lowest level (~0.01 N each) and canted off to the side by 79°. This angle is 46° to clear the limb, plus an assumed 10° to clear the Hall thruster plume spread, and 26° additional to reduce thrust further so that the spacecraft mass can be kept low. This canting reduces the efficiency of the system, and most of the impulse is lost; but it still closes with a reasonable propellant budget of 100 kg.

Whether flying this close is actually achievable may be questionable, especially since that body will not actually be a sphere and may have protrusions beyond 10 m from its "mean" surface.

Flying so close to the asteroid, not all force vectors are available, due to the asteroid blocking the Sun, affecting array power. Other notes include:

- With both MaSMi thrusters on, each at 779 W CBE, it achieves 2 km/s of ΔV for cruise and rendezvous in 200 days. The gravity tractor demonstration is then conducted at minimum throttle, with an assumed duty cycle of 100%, and takes 1.5 years.
- Note that higher S/C mass helps. We added 15 kg to make this concept close, and there is plenty of room on the LV to add additional "dumb" mass, but the ripple effects on the spacecraft will quickly drive it over the cost cap.
- It is recommended to evaluate a wide population of known asteroids, and to find the distribution of achievable deflection ΔV's.
- Did not consider enhanced gravity tractor (EGT) due to cost considerations for complexity of operations and instrumentation required. Put cost well over $500 million cost study target.

Option 17: Ion Beam & Gravity Tractor:. This option combines Options 15 and 16 into a single demo of both Ion Beam (32 kg Xe) and a gravity tractor (100 kg of Xe), since both propellant amounts were fairly low. Uses 2x MaSMi thrusters, in the same outrigger configuration as Options 15 and 16, and the payload (with radar) of Option 16. Key notes include:

- In cruise and rendezvous, it takes 169 days to achieve a ΔV of 2 km/s
- The Ion Beam deflection takes 70 days, at maximum power (1064 W CBE per engine, Isp of 1790 sec), to achieve 1mm/s deflection using 32kg of Xe.
- The Gravity Tractor deflection takes 1.5 years, at minimum power (226W CBE per engine, Isp of 947 sec), 10 m from the surface of the 50 m diameter asteroid, to achieve 1mm/s deflection using 100 kg of Xe.
- Note that this was run with the calculated propellant loads for the ion beam and gravity tractor concepts, both of which have a good deal of uncertainty. It is therefore possible

that a technically feasible "double demo" would be over the cost target; or even that there is no technically feasible concept (especially the gravity tractor portion)

As it is an order of magnitude less efficient than IBD, use of the Gravity Tractor (GT) concept is difficult to justify. The Enhanced Gravity Tractor (EGT) concept of detaching a large portion (e.g., 100,000 kg) of the asteroid by early arrival of one dedicated spacecraft and transferring it later to the IBD spacecraft may be problematic and risky for a number of reasons (Brophy et al., 2018). Given limited resources, these complexities, and lower performance than IBD, we do not recommend demonstration of GT or EGT.

4.3 OBSERVABILITY OF DEFLECTION DEMONSTRATIONS

Mitigation demonstrations are only valuable if their effectiveness can be measured. Appendix D provides analyses of the ΔV required from IBD and KI deflection experiments, given different scenarios. Ideally post-deflection orbit determination is supported by an in-situ spacecraft that can facilitate high-accuracy tracking as performed on OSIRIS-REx; this happens naturally for rendezvous missions, but requires a second flight element for KI demos. Post-deflection tracking can also be done from the ground, but requires much more time and is much less accurate.

4.4 HYBRID MISSIONS

While the *objectives* of the various missions considered in this section have significant differences, we note that the *mission designs* clearly fall into two categories: rendezvous and flyby/intercept. This invites consideration of how different objectives may share the same trajectory, instruments, and spacecraft. In the case of rendezvous missions, this aggregation can be especially synergistic because of the ability to do both characterization activities prior to mitigation demonstrations, on the same target asteroid with the same spacecraft. Examples of combinations that may save money and provide the best-value include:

(1) Hybrid Rendezvous Mission. This combines SEP-based characterization with an IBD Mitigation demonstration (Option 15-A-S). IBD may be a very important weapon in the PD arsenal and should be validated via demonstration (§3, Appendix C). Given that platform and the inherent proximity operations, Options 4-E (expanded instrumentation) or 6-E (deployed assets) could be added at marginal cost. Addition or contribution of a low-cost lander is also an obvious option. These could all be combined for the order of $500 M, with the added benefit of detailed characterization of the selected target before and after the IBD experiment.

(2) Hybrid Intercept Mission. A high-velocity (>15 km/s) KI demo may be helpful to mature our base mitigation technology, at much higher intercept velocities than planned for DART. While it is desirable to have a stationary observer to inspect and monitor the target following the impact (Option 14) this exceeds the desired $500 million cost target, so observations by another small flyby spacecraft could be used instead (Option 13-DART). Starting with this, we note that a NED trigger intercept demo inherently requires an Opnav-driven intercept similar to the KI, so adding that would add only marginally to the demo cost but serve both purposes (Option 10). Also, given the NAC payload and intercept capabilities, such a spacecraft could possibly also perform a flyby/recon demonstration (Option 1) prior to the KI event if a compatible trajectory could be found that supported two different high-velocity "intercepts". Using Table 4-4, and the costing details from Appendix E, we estimate that this combined hybrid mission could be done for the order of $400 M. However, this option would require further study to find the necessary trajectories and consider the hardware implications in more depth.

(3) Hybrid SEP Tour/Rendezvous. As a benefit of the large ΔV available from SEP, it would be straightforward to select a NEO characterization tour as described for options 7 and 8, and then end the tour with a rendezvous (Option 2-E). From Table 4-4, we see that this could be done for $300 - $400 M, depending on the number of flight elements and instrument suite desired.

Many other hybrid combinations are also possible by consideration of commonality of implementation; these could be studied in the future for "best value" solutions as PD capabilities mature.

4.5 LOWER COST ALTERNATIVES

The sizing and costing shown in §4 above assumed DART as an appropriate model; single-string Class-C with a short design life, and using existing, mature instrument and spacecraft hardware. The intention is to provide conservative cost estimates for flight systems capable of meeting the expected performance requirements with margin. A more aggressive approach was also used to estimate cost sensitivity assuming a higher risk tolerance (Malphrus et al., 2021), including Class-D and more advanced instruments that were half the mass and power of the conservative baseline assumed in §4.1.4 (Storm et al., 2017; Freeman et al., 2021). The effects of the smaller instrument demands on the spacecraft were estimated using physics/mass-based scaling relationships, which estimated propellant mass using the rocket equation and spacecraft dry mass from a ratio of propellant mass to spacecraft dry mass based on prior missions. Costing for the smaller instruments was re-run as before using NICM. We did this for the mission concepts that are most amenable to this style of mission: Options 1, 2, 4, 6, 8-2, and 10-MP (Table 4-4). Total system costs were then estimated as in §4.2. The results are that the lower-cost/higher-risk strategies could reduce the baseline costs by approximately 50%. This more aggressive approach is appropriate if a higher risk tolerance is allowed, and if a capability-driven, cost-limited strategy is preferred.

As a validation/sanity check, the costing was also performed using published information for the two current SIMPLEx projects, Janus and Lunar Trailblazer. This yielded cost estimates similar to those allowed under SIMPLEx, so we believe the costing is approximately valid and certainly valid in a relative sense to the baseline estimates in §4.2.

5 RAPID RESPONSE CONCEPTS

The bulk of this report is with regards to PD *demonstration* missions, which would focus on technology maturity, and increasing operational readiness and knowledge pertinent to PD objectives. However, for future scenarios it would be important to have fast-response *operational* missions available.

The ability to protect the planet from the threat of impact by NEOs is enhanced by the ability to detect and characterize them with "long" warning times. While a substantial effort is underway to detect and classify the existing population of potentially hazardous objects, there is no guarantee that will be comprehensive. This means that threatening NEOs may not be detected with substantial time before impact, creating a "short warning" scenario (Appendix C).

Without adequate preparation, execution of rapid response programs could be reminiscent of NASA's "Faster, Better, Cheaper" paradigm (Jolly, 2008) where there were significant cost savings, but the 62% success rate demonstrated in those years would not be appropriate for *operational* PD missions (Ward, 2012). While not cost-limited, an emergency rapid response would still be limited by fundamental schedule limitations (esp. long-lead components and minimum Integration & Test [I&T] time).

Operational missions are subject to the appropriate mission high-reliability requirements (Appendix C). They typically build on a long succession of development and early-generation prototypes over many years (e.g., GOES, GPS); hence the demonstration missions considered herein. (These demonstrations are necessary, but do not provide sufficient conditions to develop a reliable operation capability, which should be a goal for future decades).

Table 5-1. Comparison of rapid response architecture options. Storing in space is lowest mission risk and fastest response. Rolling phase A/B design maintenance could save 1–2 yr and require lower investment.

Strategy	Tailored to Target	Limited to COTS parts	Available Time for I&T	Flight/ Ops Proven	Trajectory Accessibility	Response Time	Storage/ Operations Cost	Total Project Cost
Rolling Phase A/B design	Somewhat	No	Short	No	Ground to Target using LV C3	2–3 yr	None	(Least)
Build SC On Demand	Completely	Likely	Short	No	Ground to Target using LV C3	3–5 yr	None	(Medium*)
Repurposed components/ subsystems	Constrained by available h/w	No	Short	No	Ground to Target using LV C3	2–3 yr	None	(Medium*)
Build/store entire SC Inventory	Somewhat	No	Long	No	Ground to Target using LV C3	~ 1 yr	Ground Storage	(Most*)
Build/store modular SC components Inventory	Somewhat	Possibly	Short	No	Ground to Target using LV C3	1–3 yr	Ground Storage	(Medium*)
Store in Space Awaiting Target	No (requires over-design)	No	Longest	Yes	Depart from in-space stored location	~ 1 mo	Ops check-ins required	(Medium*)

**Absolute cost estimates are out of the scope of this study but are on the order of the project costs estimated in §4 ($0.1B to $1B). More importantly, the context here is that for these rapid-response concepts to be effective, these costs need to be a-priori investments in the absence of responding to specific threats. Key: Positive Discriminator   Negative Discriminator*

Five rapid response architecture concepts are described below. Their efficacy and associated benefits and drawbacks are considered qualitatively and summarized in Table 5-1. It is difficult to provide quantitative assessments because of the very wide range of system designs and technologies considered in this report (e.g., Table 4-4), and project scheduling details are very case-specific depending on the necessary technologies and components. In order to develop accurate information for particular designs, we recommend further evaluation of the options as parts of future flight demonstrations (§7). Especially, designs for a rapid reconnaissance mission would be most critical, and are better bounded, simpler, and more mature than many of the other demos considered.

5.1 ROLLING PHASE A/B DESIGN

Flight project phase A/B typically requires ~ 25% of a project's budget, and the first two years is used in requirements definition, flowdown, and allocation, trades, interface definition, performance margin analysis/modeling, and development of specifications and subcontracts. If focused on schedule drivers and critical paths, it is plausible that a great deal of schedule risk reduction could be accomplished with ~10% of a project's budget, such that pausing at that point could provide an advanced starting point to a rapid-response design/build crisis. This strategy could be pursued for several of the most-critical system designs in parallel (e.g., rapid recon, rapid intercept, and KI or IBD deflection) for less than the cost of a single flight project. Ideally, key subcontracts would be in place for long-lead items such that system I&T could be entered in 1–2 years instead of the typical 3–4 years.

5.2 BUILD-ON DEMAND

The default concept for a rapid response spacecraft is to build it "on demand' as fast as possible once a decision is made to proceed. The primary advantage of this development process is that the spacecraft and instrumentation can be tailored specifically to the target. However, there are several limitations to this approach, especially the availability of long-lead parts and components. Thus the construction may be limited to COTS components in order to minimize lead time, and the customizability advantage is somewhat dampened. Portions of the build-on-demand schedule may be accelerated by government directive to commandeer compatible items, as described below.

5.3 REPURPOSED/COMMANDEERED

If caught completely unprepared for rapid response, an ad-hoc design would be initiated. A variation of build-on-demand is to "repurpose" (commandeer) parts, components, subsystems, or even entire spacecraft from existing flight projects. In the event of national emergency, the Government may utilize the code of federal regulations to direct actions needed for national defense. The Defense Priorities and Allocations System (DPAS) implements the priorities and allocations authority of the Defense Production Act, including use of that authority to support emergency preparedness activities pursuant to Title VI of the Robert T. Stafford Disaster Relief and Emergency Assistance Act (42 U.S.C. 5195 et seq.) (Bureau of Industry and Security, 2018), and the priorities authority of the Selective Service Act and related statutes, all with respect to industrial resources. The DPAS establishes procedures for the placement, acceptance, and performance of priority rated contracts and orders and for the allocation of materials, services, and facilities.

The best schedule case would be to invoke such Government directive to transfer existing hardware from other flight projects. The feasibility of doing this effectively depends on the level of integration; low-level parts and components are very likely usable for the required mission, but the ability to use existing subsystems, spacecraft and instruments becomes less likely due to the specificity of space mission requirements and interfaces. Pulling items from other projects implies integration of components, subsystems and instruments that were not designed to go together and not designed for the critical mission to be launched in just a few years. This may not be practical for a real-life situation and induce undesirable risk, potentially leading to failure. Moreover, the compressed timeline may force the program to forgo necessary verification and validation tests, further increasing mission risk. It would be advisable to avoid this situation to the extent affordable using a more deliberate approach.

5.4 BUILD-TO-INVENTORY

An effective preparatory strategy is to develop several complete spacecraft, in advance of any rapid response scenarios, and store them on the ground. This set of spacecraft would be architected so that they efficiently cover the range of expected scenarios, enabling them to be somewhat tailored to the body of interest, but also fully complete and ready to fly. Besides the cost of the initial investments, there are also additional storage costs and complexities associated with keeping a fleet of spacecraft maintained and ready to fly.

5.5 STANDARDIZATION AND MODULARITY

Similar to the build-to-inventory concept, an alternative may be to design and build an inventory of components which are modular and compatible. This still allows for testing and validation in advance of launch, but enables much more customizability and flexibility in tailoring the spacecraft to the target. There is some precedent for this in the telecommunications industry with programs such as Boeing's 702 bus, SSL/Maxar's 1300 Series bus, and Lockheed's A2100 bus. The average development time for these busses is 3.3 years, which is a large time reduction over their 'first build' editions which often take twice this long. For example, the shortest ever developed was a Lockheed A2100 bus which flew within 2 years of its initial contract start date (Davis & Filip, 2015). All of these telecom-style busses are intended to accommodate a wide range of payloads by using highly matured components and will scale based on payload or customer requirements. The higher degree of in-advance qualification, in addition to modularity, enables these busses to minimize non-recurring engineering (NRE) costs and time. Their scalability is well demonstrated, for example the SSL-1300 bus has flown with as few as a single X-band transponder (Krebs, 2017), and as many as 56 (C and Ku-Band) transponders (SATBEAMS, 2020).

For planetary defense, the architecture would build an inventory of compatible modular components/subsystems which can be integrated based on required scale and instrumentation specific to the newly detected threat. Subsequent to system design trades, the correct components are selected and can pass through assembly, integration, and testing quickly due to the heritage and already completed testing. However, brute-force solutions like this are expensive because a wide range of possible component building-blocks are required and the total cost can resemble that of several whole spacecraft.

A variation of this concept is SmallSats, which constitute a relatively new approach in planetary missions, taking advantage of the miniaturization of space-based electronics, sensors, instrumentation, and control systems, in order to create a fully functional spacecraft which is more affordable, quicker, and easier to build than previous monolithic spacecraft. The inventory of COTS SmallSat components is growing quickly, and many manufacturers have operational production lines which one can procure flight ready components in months such as Tyvak which has platforms available within 6–18 months (Tyvak, 2020). Such spacecraft can be developed quickly and with fairly low associated costs ($500 k to $2 M), making them well suited for higher risk tolerant applications often associated with rapid response scenarios. However, this would place limitations on available performance that may not be adequate for the specific PD mission needs.

5.6 STORE IN SPACE

On-orbit storage is a concept which has been used in a variety of different contexts. One example that provides precedent is the GOES program, which uses a fleet of geostationary spacecraft to track and warn against rapidly developing weather. The dangers of a missed detection or monitoring led the program to include spares in the fleet. The program elected to store spares on-orbit rather than storing them on the ground. This was because ground-stored spacecraft would require 9 to 12 months of recertification and test before launch, in addition to 3 months of commissioning post launch, before they could be active in the fleet: an unacceptable response time for monitoring an urgently developing weather scenario. In addition, the potential for a launch failure when a spare was urgently required to replace a failing/failed orbital asset was unacceptable to the program (NASA, 2009). This strategy is indicative of a mature operational program and a potential model for future PD.

The GOES-14 spacecraft was launched in 2009 and held on orbit as a replacement for when other members of the GOES fleet might reach end of life or an unexpected failure (Clark, 2021). It was stored effectively for approximately 3 years until it was reactivated replacing a failed spacecraft, and also would later serve as a stand-in for a different failed asset. Current GOES spacecraft, such as GOES-R, are designed for up to 5 years of on-orbit storage before they begin their 10+ year primary mission, as well as additional 5 years on-ground storage prior to flight (Walsh, 2010).

The ESA Comet Interceptor program will also demonstrate on-orbit storage architecture. This will provide several advantages over the build-on-demand and build-to-inventory concepts. Once on-orbit there is no longer risk associated with launch vehicle failure or launch window, and the storage location provides significant flexibility to depart from the L2 Lagrange point and intercept an object (provided adequate ΔV is available). This spacecraft will be stored in space awaiting a previously undiscovered, serendipitous target to be detected. Detection and initial characterization will likely be accomplished with one of several GBOs (Grav et al., 2016) currently in operation or being built. Once the target is identified, the mission planning team will need to design and execute maneuvers to place the spacecraft on an intercept trajectory. Upon intercept of the object, the spacecraft and its daughter probes will study the object at a range of observation distances, including distances commensurate with demonstration of (but not execution of) kinetic impact mitigation, relaying all such information back to Earth. Although primarily a science-driven project, ESA's successful completion of this mission provides a framework for process and lessons learned which will be highly informative to any future planetary defense efforts.

Comet Interceptor is a good example that effective rapid-response missions are best executed by parking assets in space, such as halo orbits around the Sun-Earth L1 or L2 libration points (Williams et al., 2000; Roberts, 2011; Dunham et al., 2014). If a halo radius of >700,000 km is used, then the

ΔV needed to leave that orbit and travel away from the Earth system is nearly zero; in this case the LV C3 of 2 km^2/s^2 assumed in the example intercept trajectories (§4.1) could be supplied instead by a large, integral propulsion system, or by an upper stage attached to the spacecraft. The fact that NEO intercept trajectories are not far from Earth orbit thus enables the storage in space concept for rapid response.

A regular cadence of replacing the on-station assets every several years would address potential lifetime issues while mitigating obsolescence and accommodating technology advancements. This would also offer the ability to perform live experiments using the obsolete vehicle in an exercise as a means of disposal. For these reasons, as well as the unique ability for very rapid response, storage in space is the best alternative if appropriate funding were available on a long-term basis.

6 TECHNOLOGY READINESS

Inherent in the cost modeling in §42 is usage of existing hardware and software, so technology readiness in general is considered adequate today for most aspects of these missions. Obvious exceptions to that include the objectives of the demos themselves, such as:

- Terminal guidance appropriate for small asteroid intercept (Barbee et al., 2018; Barbee et al., 2020). Fast-flyby reconnaissance and KI closing velocities are typically in the range of 15 km/s or higher. This drives NAC design, autonomous targeting speed, and attitude control and cross-track agility beyond previous implementations. This would be a key objective of the proposed recon demonstration mission.
- Technologies to measure asteroid gravity (and hence mass) during a flyby (Bull et al., 2021; Christensen et al., 2021). This may or may not be included in the Fast-fly recon demo, depending on technology maturity.
- SEP and GN&C systems appropriate for low-altitude hovering to perform IBD (or GT) (Brophy et al., 2018). While proximity operations have matured extensively in the last decade, implementation for IBD requires very long duration autonomous operations, active altitude sensing, and altitude control via throttling of the SEP thrusters. This would be the focus of a slow push flight demonstration.
- Nuclear Electric Xenon Ion System (NEXIS) thruster development (Brophy et al., 2018) and/or other gridded ion thruster implementations (e.g., NEXT). This could be done in parallel with the proposed slow push demo, which would be focused on GN&C technologies.
- Active RADAR NED trigger. This would trigger at shorter ranges (e.g., 100–200 m) and higher closing velocities (e.g., 15 km/s), and on smaller targets than currently available triggering systems. Most of this development could be done in a terrestrial environment. However, it is important because of the number of worst-cases that may require NED with no other options (red area in Figure 3-2). Demonstration of this technology would be a natural companion with a future KI demonstration, since both would require the same intercept/terminal guidance capability.
- Landed instruments to test geophysical and geotechnical properties (Watanabe et al., 2017).

Also, the more aggressive costing described in §4 assumes more advanced instrument designs, so additional investment in those technologies helps to reduce flight system cost indirectly. Similarly, continued investment/maturation in smaller spacecraft will incrementally help reduce costs, assuming the capabilities are compatible with deep space missions and PD objectives.

7 RECOMMENDATIONS

Specific recommendations in response to this study request:

1) The highest priority recommendation is a rapid-response, flyby reconnaissance mission targeted to a challenging NEO, representative of the population of highest-probability hazardous objects. Such a mission should test flyby characterization methods to assess their capabilities and limitations, to better prepare for a short-warning-time NEO threat. If additional PD objectives can also be achieved by this mission, that is highly desirable and as this study shows, would be cost effective.

2) At least one other priority mitigation and characterization mission should also be developed within the decade. These include, in no particular order:

   a. A characterization tour mission to gain characterization information required for future deflection/disruption missions and to exercise characterization capabilities for a range of targets;

   b. A kinetic impact mission on a smaller NEO and at a higher closing speed than the Double Asteroid Redirection Test (DART) mission to acquire the needed experience of kinetic impact mitigation missions;

   c. A slow-push mitigation mission demonstration, such as ion beam deflection, to develop several different technologies that can be available and optimized for specific mitigation situations that may arise. This may readily be combined with characterization of the target asteroid.

   This study concludes that such missions are technically feasible and provides a range of cost estimates to guide development of the PD budget. Hybrid missions that combined multiple PD objectives in a single mission are shown to be cost-effective investments (§4.4).

3) Risk reduction investments for the technology readiness issues described in §6 would facilitate their insertion into the pertinent demo missions. Guidance, Navigation, and Control and instrumentation appropriate for hypervelocity flybys/intercepts and GN&C for long-duration hovering over asteroids are fundamentally important for near-term objectives.

4) Study of practical limitations of specific rapid-response strategies (§5) should be considered as part of specific flight demos as possible; for example, exactly what preparations would be necessary to launch 1, 2, 3, or 4 years from time of alert, and what would those cost? Concepts for on-orbit storage (e.g., at L2) of PD spacecraft critical for short-warning responses should be matured (§5). Especially, the feasibility of a single rapid recon design that would envelope most expected need cases should be studied in more depth and consideration given to reducing the life cycle time if needed vs the investment required e.g., executed through Phase A, Phase B, Phase C, or Phase D .

A ACRONYMS

AAS	American Astronautical Society
ACE	Advanced Composition Explorer
ACS	Attitude Control System
AIAA	American Institute of Aeronautics and Astronautics
AIDA	Asteroid Impact and Deflection Assessment
APL	Johns Hopkins University Applied Physics Laboratory
ASDS	Automated Spaceport Drone Ship
ASE	Association of Space Explorers
AU	Astronomical Unit
C3	Launch Specific Energy
CBE	Current Best Estimate
CDS	CubeSat Design Specifications
CML	Concept Maturity Level
CNEOS	Center for NEO Studies
Comet-I	Comet Interceptor
COTS	Commercial Off the Shelf
CSS	Catalina Sky Survey
CuSP	CubeSat for Solar Particles
DART	Double Asteroid Redirection Test
DPA	dual payload adapter
DPAS	Defense Priorities and Allocations System
EEE	Electrical, Electronic, and Electromechanical
EGT	Enhanced Gravity Tractor
EoL	End of Life
EP	Electric Propulsion
ESA	European Space Agency
ESM	European Service Module
ESPA	Evolved Expendable Launch Vehicle (EELV) Secondary Payload Adapter
F9	Falcon 9
FY	Fiscal Year
GBO	Ground-Based Observatories
GN&C	Guidance Navigation and Control
GOES	Geostationary Operational Environmental Satellite
GPS	Global Positioning System
GT	Gravity Tractor
HFR	High Frequency Radar
HGA	High Gain Antenna
I&T	Integration & Test
IBD	Ion Beam Deflection
ICPS	Interim Cryogenic Propulsion Stage
IR	InfraRed
ISIS	Intelligent Spacecraft Interface Systems
IXPE	Imaging X-ray Polarimetry Explorer

JAXA	Japan Aerospace Exploration Agency
JPL	Jet Propulsion Laboratory
KI	Kinetic Impact
L1/2	Lagrange Points 1 or (Sun-Earth system)
LEO	Low Earth Orbit
LFR	Low Frequency Radar
LIDAR	Light Detection and Ranging
LLNL	Lawrence Livermore National Laboratory
LOLA	Lunar Orbit and Landing Approach
LPI	Lunar and Planetary Institute
LSP	Launch Services Program
LSST	Large Synoptic Survey Telescope
LV	Launch Vehicle
M-Argo	Miniaturised Asteroid Remote Geophysical Observer
MLPS	Mid- and Long- wave infrared Point Spectrometer
MMX	Moons Exploration mission
MSSS	Malin Space Science Systems
MYA	Million Years Ago
NAC	Narrow Angle Camera
NASA	National Aeronautics and Space Administration
NASEM	National Academies of Sciences, Engineering, and Medicine
NDIA	National Defense Industrial Association
NEA	Near-Earth Asteroid
NED	Nuclear Explosive Device
NEO	Near-Earth Object
NEOSM	NEO Surveyor Mission
NEOWISE	Near-Earth Object Wide-Field Infrared Survey Explorer
NEXIS	Nuclear Electric Xenon Ion System
NICM	NASA Instrument Cost Model
NOAA	National Oceanic and Atmospheric Administration
NPR	NASA Procedural Requirements
NRE	Non Recurring Engineering
OSIRIS-REx	Origins, Spectral Interpretation, Resource Identification, Security-Regolith Explorer
PD	Planetary Defense
PDC	Planetary Defense Conference
PHA	Potentially Hazardous Asteroid
PHO	Potentially Hazardous Object
POC	Point of Contact
RADAR	Radio Detection and Ranging
RMA	Rapid Mission Architecture
RTLS	Return to Launch Site
S/C or SC	Spacecraft
SOHO	Solar and Heliospheric Observatory

SSL	Space Systems Loral
SSPA	Solid State Power Amplifier
TOF	Time of Flight
UHF	Ultra High Frequency
USA	United States of America
VNIR or Vis/NIR	Visible and Near-Infrared
WAC	Wide Angle Camera

B LITERATURE SURVEY SUMMARY

Research papers, white papers, technical reports, and slide packages were pulled from various sources including both available online repositories, JPL-internal libraries, and certain papers which are not publicly released due to competition sensitivity. The literature study assembles these papers, categorizes them, and includes statistics regarding age, target body, maturity, and other data. The materials collected were considered and informed the architecture trade space presented in §4. The data base is summarized in Table B-1.

To date there are over one hundred-thirty entries within the database, and more than half of those fall under the characterization category (62%). A majority of characterization papers are written around the study of a specific target or targets. A portion of the characterization missions are associated with revisiting a body; the intent being to either explore how a previous mission changed the environment, or to re-study a body with new, more targeted, instrumentation. Many targets are studied, but the most popular targets which were studied in multiple papers include Apophis and the binary 65803 Didymos, probably due to their relative interest.

The characterization category is dominated by rendezvous opportunities as they provide the best balance between technical difficulty, cost, and scientific return. Flyby opportunities are also well represented due to their relative ease of modeling along with lower cost and complexity. The 'other' category is primarily populated by ground-based observation studies, or by meta-analyses of an existing dataset in order to improve propagation techniques or characterization. Papers on multiple target tours are rare, due to the complexity of target selection and mission design. In general, secondary (multiple) body studies are often secondary follow-on events, which are opportunistically identified and pursued only after the primary mission has completed.

The mitigation category includes a variety of technologies. Many mitigation papers overlap in terms of the subjects they focus on, often as trade studies between efficacy options for specific types of bodies, cost brackets, and urgency levels. It is also worth noting that a portion of the mitigation papers are written on completed missions such as Deep Impact, and not directly intended for planetary defense.

A quick look at funding and progress of various missions, primarily from the characterization category, can be misleading. 'Funded or Completed' missions included Deep Impact, Hayabusa and Hayabusa2, Rosetta, OSIRIS-REx, Dawn, and Psyche. Missions with an 'anticipated launch date' and some funding allocated include mostly smaller spacecraft launching as ride-shares or as higher profile mission auxiliaries; examples include M-Argo, Aster, NEA-Scout, NEOSM, Janus, DART, and Hera. Finally, the 'proposed' spacecraft include Discus, NEACO, NEACORE, and PrOVE, all which remain seeking funding. At first blush, this appears indicative that there is significant activity in the field, however, most of the missions described above are not direct contributors to planetary defense. These missions are aimed at scientific exploration of a single very specific body, rather than the study or detection of high priority targets for planetary defense to provide precise orbit, mass, material properties, internal structure, etc. The major existing contributors to planetary defense include only NEOSM, Hera, and DART.

Table B-1. See following pages for literature search data base summary

Primary Focus	Title	Publication Year	Main Author	Secondary Author List	Notes to Reader	General Description of Paper	Spacecraft Name	Target Body	Characterization				Mitigation			
									Rendezvous	Tour (Multi-Target)	Flyby (Single Target)	Other	Kinetic Impact	Nuclear Explosive Device (or Simulator)	Gravity Tractor	Other (laser, enhanced gravity tractor, etc)
						103			Count: 41	Count: 5	Count: 16	Count: 50	Count: 18	Count: 13	Count: 6	Count: 17
Mitigation	Deep Impact Mission: Looking Beneath the Surface of a Cometary Nucleus	2005	Russell			The Deep Impact mission will provide the first data on the interior of a cometary nucleusand a comparison of those data with data on the surface. Two spacecraft, an impactor and a flybyspacecraft, will arrive at the comet. These data will provide unique information on the structure of the nucleusnear the surface and its chemical composition. They will also used to interpret the evolutionary effectson remote sensing data and will indicate how those data can be used to better constrain conditions inthe early solar system							**X**			
Mitigation	Deep Impact as a World Observatory Event: Synergies in Space, Time, and Wavelength	2009	Kaufl	Sterken, Leibundgut		Comet 9P/Tempel1 was the focus of an unprecedented worldwide long-term multi-wavelength observation campaign. The comet was also studied through its perihelion passage by various spacecraft including the Deep Impact mission itself, the HubbleSpace Telescope, Spitzer, Rosetta, XMM and all major ground-based observa-tories in a wavelength band from the cm-wave radio astronomy to x-rays.The purpose of this meeting was to bring together an audience of observers across the electromagnetic spectrum to make full use of the massive ground-based observing dataset.							**X**			
Mitigation	ENHANCED GRAVITY TRACTOR TECHNIQUE FOR PLANETARY DEFENSE IAA-PDC-15-04-11	2015	Mazanek			Describes technique of either domino-ing, or amassing, several smaller bodies together in order to deflect a larger PHO/NEO more rapidly.										**Enhanced Gravity Tractor**
Characterization	Architecture Trades for Accessing Small Bodies with an Autonomous Small Spacecraft	2020	Papais	Bandyopadhyay; Bhaskaran; Hockman; Karimi; Nesnas	More focused on origins of the solar system than planetary defense.	How to best utilize new, small technologies to perform characterization missions to small bodies.			**X**		**X**					
Characterization	DISCUS - The Deep Interior Scanning CubeSat mission to a rubble pile near-Earth asteroid	2018	Bambach			Conceptual design study for a tandem 6U CubeSat carrying a bistatic radar as the main payload designed to determine the internal macro-porosity of a 260–600 m diameter Near Earth Asteroid. It aims for a single-unit (1U) radar design equipped with a half-wavelength dipole antenna developed for ESA's technologyprojects GINGER and PIRA. Additionally, an initial designstudy of the platform and targets accessible within 20 lunar distances is presented.	**Discus**		**X**							
Characterization	What's Inside a Rubble Pile Asteroid? DISCUS - a Tomographic Twin Radar Cubesat to Find Out	2018	Bambach			Using DISCUS's radar, and using inversion methods developed for medical tomography the data would allow to reconstruct the large scale interior structure of a small body.			**X**							
Characterization	Hayabusa-2 mission target asteroid 162173 Ryugu (1999 JU3):Searching for the object's spin-axis orientation	2016	Müller	J. ˇDurech2, M. Ishiguro3, M. Mueller4, T. Krühler1, H. Yang3, M.-J. Kim5, L. O'Rourke6, F. Usui7, C. Kiss8, B. Altieri6, B. Carry9, Y.-J. Choi5, M. Delbo10, J. P. Emery11, J. Greiner1, S. Hasegawa12, J. L. Hora13, F. Knust1, D. Kuroda14, D. Osip15, A. Rau1, A. Rivkin16, P. Schady1, J. Thomas-Osip15, D. Trilling17, S. Urakawa18, E. Vilenius19, P. Weissman20, and P. Zeidler21		. We reanalysed previously published Subaru-COMICS and AKARI-IRC obser-vations and merged them with aSpitzer-IRS data set. In addition, we used a large set ofSpitzer-IRAC observations	**Hayabusa-2**	**Ryugu (1999 JU3)**	**X**							
Characterization	Envelop of reachable asteroids by M-ARGO CubeSat	2020	Topputo			Assessment of the target list of attainable asteroids for the M-ARGO mission. Pulls from hundreds of candidates from the Minor Planet Center Database and implements a realistic thruster model. The analysis shows that approximately 150 minor bodies are potentially reachable. A manual inspection of the transfer features led to a subset of 41 targets	**M-Argo**		**X**							
Characterization	Miniaturised Asteroid Remote Geophysical Observer (M-ARGO) - Astand-alone deep space CubeSat system for low-cost science and exploration missions	2018	Walker			A slideset summary of the mission objectives, phases, and programmatics of the M-Argo spacecraft mission concept.	**M-Argo**		**X**							
Characterization	Preliminary mission profile of Hera's Milani CubeSat	2021	Ferrari	Franzese; Pugliatti; Giordano; Topputo		Investiages feasibility and preliminary mission profile of Hera's Milani Cubesat. Identifies design challenges and discusses design criteria to find suitable solutions in terms of mission analysis, operational trajectories, and GN&C design.	**Hera**	**Didymos**	**X**							
Characterization	THE NEAR-EARTH ASTEROID CHARACTERIZATION ANDOBSERVATION (NEACO) MISSION	2017	Venigalla			The Near-Earth Asteroid Characterization and Observation (NEACO) mission pro-poses to explore the fast-rotating asteroid (469219) 2016 HO3with a SmallSatspacecraft and perform an early scientific investigation to enable future, more in-depth missions.	**NEACO**	**2016 HO3**	**X**							
Characterization	Near-Earth Asteroid Characterization and Observation (NEACO) Mission to Asteroid (469219) 2016 HO3	2019	Venigalla			. In this study, a SmallSatspacecraft performs a scientific investigation that characterizes the asteroid at a sufficient degree to enable future,more in-depth missions.	**NEACO**	**2016 HO3**	**X**							
Characterization	The aster mission: Exploring for the first time a triple system asteroid	2011	Macau			2001 SN263 is a triple system asteroid, this interesting system was chosen as the target for the Aster mission: The first Brazilian space exploration undertaking of a small spacecraft.	**Aster**	**Triple asteroid 2001-SN263**	**X**							

									Characterization				Mitigation			
Primary Focus	**Title**	**Publication Year**	**Main Author**	**Secondary Author List**	**Notes to Reader**	**General Description of Paper**	**Spacecraft Name**	**Target Body**	**Rendezvous**	**Tour (Multi-Target)**	**Flyby (Single Target)**	**Other**	**Kinetic Impact**	**Nuclear Explosive Device (or Simulator)**	**Gravity Tractor**	**Other (laser, enhanced gravity tractor, etc)**
						103			**Count: 41**	**Count: 5**	**Count: 16**	**Count: 50**	**Count: 18**	**Count: 13**	**Count: 6**	**Count: 17**
Characterization	Reviewed plan of the ALR, the laser rangefinderfor the ASTER deep space mission to the tripleasteroid 2001-SN263	2016	Brum			The Brazilian deep space mission ASTERplans to send a small spacecraft to investigate the triple asteroid 2001-SN263. In this effort,a preliminary design of a laser altimeter to meet the mission needs was created and presented in 2010-2011	**Aster**	**Triple asteroid 2001-SN263**	**X**							
Characterization	Apophis 2029: Decadal Opportunity for the Science of Planetary Defense	2021	Binzel			A potentially hazardous asteroid as large as Apophis encountering Earth this closely is, on average, a once-per-thousand year event. In this White Paperwe outline our current best understanding, and uncertainties, for scientific advances in the physical study of potentially hazardous asteroids that may be achievableby measuring physical changes of Apophis' spin, surface structure, and/or shape configuration in response to Earth's tidal torques.		**Apophis**	**X**							
Characterization	APOPHIS 2029 PLANETARY DEFENSE MISSION OPTIONS.	2015	Cheng			This report summarizes the mission options and scientific opportunities to be gained from study of Apophis 2029.		**Apophis**	**X**							
Characterization	The Main-belt Asteroid and NEO Tour with Imaging and Spectroscopy (MANTIS)		Rivkin			MANTIS addresses many of NASA's highest priorities as laid out in its 2014 Science Plan and provides additional benefit to the Planetary Defense and Human Exploration communities via a low-risk, cost-effective tour of the near-Earth region and inner asteroid belt.	**MANTIS**	**Main-belt Asteroids**			**X**					
Characterization	Nanospacecraft Exploration of Asteroids by Collision and flyby Reconnaissance (NEACORE) IAA-LCPM-19-05-06	2019	Walker			This concurrent design study acts as a feasibility study for a new concept of nanosatellite mission framework which is intended to allow reconnaissance of a large number of NEAs while minimizing cost. The presented mission framework consists of pairs of nanosatellites travelling together on multi-target flyby trajectories, and is designed to be flexible to suit many different target sets	**NEACORE**	**Many**			**X**					
Characterization	Near Earth Asteroid Scout Mission	2014	Castillo-Rogez			This report provides an introduction to NEAScout, the target, science definition and implementation: NEAScout characterizes 1 NEA with a 6U cubesat as an imager to strategic knowledge gaps.	**NEA-Scout**	**1991 VG**								
Characterization	Janus: A NASA SIMPLEx mission to explore two NEO Binary Asteroids	2020	Scheeres			Janus is a NASA SIMPLEx mission currently in Phase B. The Janus mission concept plans to take advantage of the NASA Psyche launch to send two spacecraft to fly by Near Earth Objects of interest. A specific point design has been developed that sends two spacecraft to two binary asteroid systems, (175706) 1996 FG3 and (35107) 1991 VH, both of which have been observed repeatedly with photometry, spectrometry and radar.	**Janus**	**(175706) 1996 FG3 and (35107) 1991 VH,**		**X**						
Characterization	Overview of Primitive Object Volatile Explorer (PrOVE) CubeSat orSmallsat concept	2018	Clark			Here we describe the Primitive Object Volatile Explorer (PrOVE),a smallsatmission concept to study the surface structure and volatile inventory of comets in their perihelion passage phase when volatile activity is near peak	**PrOVE**	**Comet**		**X**						
Characterization	Primitive Object Volatile Explorer (PrOVE) – Waypoints and Opportunistic Deep Space Missions to Comets	2018	Hewagama	S. Aslam2, P. Clark3, M. Daly4, L. Feaga1, D. Folta2, N. Gorius5,2, T. Hurford2, T. A. Livengood1,2, B. Malphrus6, M. Mumma2, C. Nixon2, J. Sunshine1, G. Villanueva2, and A. Zucherman6, 1Dept. of Astronomy, University of Maryland, College Park, MD 20742 (correspondence: tilakh@umd.edu), 2Planetary Systems Laboratory, NASA/GSFC, Greenbelt, MD 20771, 3Solar System Exploration Directorate, JPL, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA 31109, 4Dept. of Earth & Space Science & Engineering, 102 Petrie Science and Engineering Bldg, 4700 Keele Street, Toronto, Canada M3J 1P3, 5Dept. of Physics, Catholic University of America, 620 Michigan Ave., Washington, DC 20064, 6Space Science Center, Morehead State University, Morehead, KY 40351.		UMD/GSFC/MSU/CUA/JPL team proposed Primitive Object Volatile Explorer (PrOVE) to the SIMPLEx-1 program, the mission was to a volatile rich Jupiter Family Comet46P/Wirtanen with a ecliptic plane crossing with 0.09AU of Earth, Launch platform delays were an identified risk, and .spacecraft and Propulsion were not investigated for more remote missions.	**PrOVE**	**Comet**		**X**						

Primary Focus	Title	Publication Year	Main Author	Secondary Author List	Notes to Reader	General Description of Paper	Spacecraft Name	Target Body	Characterization: Rendezvous	Characterization: Tour (Multi-Target)	Characterization: Flyby (Single Target)	Characterization: Other	Mitigation: Kinetic Impact	Mitigation: Nuclear Explosive Device (or Simulator)	Mitigation: Gravity Tractor	Mitigation: Other (laser, enhanced gravity tractor, etc)
						103			Count: 41	Count: 5	Count: 16	Count: 50	Count: 18	Count: 13	Count: 6	Count: 17
Characterization	Ground-Based Radar for Planetary Science and Planetary Defense	2021	Taylor		Reader should take caution that this paper, and any other papers that reference the Arecibo Observatory, are likely out of date. The Arecibo Observatory is permanently offline as of Aug 2020. The importance of this observatory in PD is highlighted in this report, indicating a gap in detection efforts.	Planetary radar is a unique method for studying solid bodies in the Solar System and arguably the most powerful method for post-discovery, remote physical and dynamical characterization of near-Earth objects. ith dedicated planetary radar facilities unlikely on the decadal timescale, it is imperative that the shared-use, single-dish radio telescopes utilized for planetary radar remain viable throughthe next decade. This includes transmitters at Arecibo Observatory and the Goldstone Solar System Radar, along with the Green Bank Telescope, which is often used as a receiver in conjunction with the transmitting telescopes.						Earth-Based Survey				
Characterization	NEOWISE Observations of Near-Earth Objects: Preliminary Results	2011	Mainzer			With the NEOWISE portion of theWide-field Infrared Survey Explorer(WISE) project, we have carried out a highly uniform survey of the near-Earthobject (NEO) population at thermal infrared wavelengths ranging from 3 to 22μm, allowing us to refine estimates of their numbers, sizes, and albedos.						LEO Based Survey				
Char / Mitig	Future Spacecraft Missions for Planetary Defense Preparation	2020	Barbee	Abell; Binzel;Brozovic; Cahill; Chodas; Daly; Davis; Mainzer; Mazanek; Park; Sotirelis; Venditti		Recommendations for the next 3-missions that should be flown for Planetary Defense.					X	Space-Based IR Observatory	X	X		
Characterization	Near-Earth Object Characterization Priorities For Planetary Defense	2020	Barbee			National Plan Actions Relevant to PD NEO Characterization Priorities: Action 1.2: Identify technology and data processing capabilities and opportunities in existing and new telescope programs to enhance characterization of NEO composition and dynamical and physical properties. • Action 1.4: Establish and exercise a process for rapid characterization of a potentially hazardous NEO. • Action 2.2: Ascertain what information each participating organization requires on what timeframe, identify gaps, and develop recommendations for modeling improvements. • Action 2.5: Assess the sensitivities of these models to uncertainties in NEO dynamical and physical properties. Action 3.1: Assess technologies and concepts for rapid-response NEO reconnaissance missions. • Action 3.3: Create plans for the development, testing, and implementation of NEO reconnaissance mission systems. • Action 5.2: Establish a procedure and timeline for conducting a threat assessment upon detection of a potential NEO impact, and for updating the threat assessment based on improved data. • Action 5.6: Establish a procedure and timeline for conducting a risk/benefit analysis for space-based mitigation mission options following a NEO threat assessment. • Action 5.7: Develop benchmarks for determining when to recommend NEO reconnaissance, deflection, and disruption missions.			X		X					
Characterization	Vision and Voyages for Planetary Science in the Decade 2013-2022	2013	Committee on the Planetary Science Decadal SurveySpace Studies Board			Previous planetary decadal study. Large body of work including some small body characterization. Start on Page 87 (Primitive Bodies). Primary Goals: Decipher the record in primitive bodies of epochs and processes not obtainable elsewhere; Understand the role of primitive bodies as building blocks for planets and life.										
Characterization	Autonomous SmallSats for Small Bodies	2020	A-Team	Freeman; Castillo-Rogez; Nesnas		Participants presented on the science related to Small Bodies and gave an overview of autonomy capabilities. A number of architectures were identified that would return compelling science. There was a discussion about how autonomy would enable such architectures. Feasibility of a number of architectures from an engineering point of view were discussed. There was also a discussion about how autonomy would enable such architectures. Quad charts were generated for four different architectures that could be enabled by autonomous SmallSats.			X		X					
Characterization	The Case for a Planetary Defense-Optimized NEO Characterization Tour	2020 (or 2021?)	Rivkin			Meta-Analysis: Suggestions for characterization mission objectives for rendesvous, tours, and flybys.			X	X	X	Earth-Based Observation				

									Characterization				Mitigation			
Primary Focus	**Title**	**Publication Year**	**Main Author**	**Secondary Author List**	**Notes to Reader**	**General Description of Paper**	**Spacecraft Name**	**Target Body**	**Rendezvous**	**Tour (Multi-Target)**	**Flyby (Single Target)**	**Other**	**Kinetic Impact**	**Nuclear Explosive Device (or Simulator)**	**Gravity Tractor**	**Other (laser, enhanced gravity tractor, etc)**
						103			**Count: 41**	**Count: 5**	**Count: 16**	**Count: 50**	**Count: 18**	**Count: 13**	**Count: 6**	**Count: 17**
Characterization	Small Bodies Assessment GroupCommunity Decadal Survey Summary	2020	Burrati	Sykes; SBAG Steering Committee		Meta-Analysis: Questionnaire identifying top characterization priorities for NEOs based on Small-Body community respondants. Top three priorities: 1. Population identification and physical/compositional characterization; 2. Understanding the characteristics and evolution of individual objects; 3. Determining early conditions in the Solar System (e.g., compositional gradient, early formation history)						**Multiple mission class priorities listed but includes all small bodies (not only NEO/PHO; not focused on Planetary Defense)**				
Mitigation	Technology Development for Planetary Defense In Situ Spacecraft Missions to Near-Earth Objects	2020 (or 2021?)	Barbee	Paul A. Abell (NASA Johnson Space Center (JSC)) Justin Atichson (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Olivier Barnouin (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Shyam Bhaskaran (CalTech/Jet Propulsion Laboratory (JPL)) John Brophy (CalTech/Jet Propulsion Laboratory (JPL)) Joshua Cahill (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Paul Chodas (CalTech/Jet Propulsion Laboratory (JPL)) Dan Mazanek (NASA/Langley Research Center (LARC)) Ryan Park (CalTech/Jet Propulsion Laboratory (JPL)) Cathy Plesko (Los Alamos National Laboratory (LANL)) Joshua Schare (Sandia National Laboratory (SNL)) Thomas S. Sotirelis (Johns Hopkins University Applied Physics Laboratory (JHUAPL))		Technology development recommendations for reconnaissance and mitgation missions			**X**		**X**	**Earth observatories**	**X**	**X**	**X**	**X**
Characterization	Research and Analysis for Planetary Defense In Situ Spacecraft Missions to Near-Earth Objects	2020 (or 2021?)	Barbee	Paul A. Abell (NASA Johnson Space Center (JSC)) Daniel R. Adamo (Independent Astrodynamics Consultant) Justin Atchison (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Olivier Barnouin (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Joshua Cahill (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Paul Chodas (CalTech/Jet Propulsion Laboratory (JPL)) Terik Daly (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Amanda Davis (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Jessie Dotson (NASA/Ames Research Center (ARC)) Dan Mazanek (NASA/Langley Research Center (LARC)) Ryan Park (CalTech/Jet Propulsion Laboratory (JPL)) Clemens Rumpf (STC at NASA/Ames Research Center (ARC)) Thomas S. Sotirelis (Johns Hopkins University Applied Physics Laboratory (JHUAPL)) Lorien Wheeler (NASA/Ames Research Center (ARC))		Meta-Analysis: What are the key spacecraft & launch vehicle properties to mount a realitic assessment mission (particularly early assessments) that offers valuable information about NEOs? What are they key NEO properties that need to be understood to mount a mitigation mission?						**Spacecraft & LV properties to mount a characterization mission**				**NEO properties to mount a mitigation mission**

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						103			Count: 41	Count: 5	Count: 16	Count: 50	Count: 18	Count: 13	Count: 6	Count: 17
Characterization	Near-Earth Object Characterization Priorities and Considerations for Planetary Defense	2020 (or 2021?)	Abell	Carol Raymond (CalTech/Jet Propulsion Laboratory (JPL)); Terik Daly (Johns Hopkins University Applied Physics Laboratory (JHUAPL)); Daniel R. Adamo (Independent Astrodynamics Consultant); Brent W. Barbee (NASA Goddard Space Flight Center (GSFC)); Megan Bruck Syal (Lawrence Livermore National Laboratory (LLNL); Kyla Carte (George Mason University); Nancy L. Chabot (Johns Hopkins University Applied Physics Laboratory (JHUAPL)); Jesse L. Dotson (NASA Ames Research Center (ARC)); Lee D. Graham (NASA Johnson Space Center (JSC)); Rob R. Landis (NASA Johnson Space Center (JSC)); Amy Mainzer (University of Arizona); Mike C. Nolan (University of Arizona); Edgard Rivera-Valentin (Lunar and Planetary Institute, USRA); Andy S. Rivkin (Johns Hopkins University Applied Physics Laboratory (JHUAPL)); Patrick A. Taylor (Lunar and Planetary Institute, USRA); Flaviane C. F. Venditti (Arecibo Observatory - University of Central Florida); Lorien Wheeler (NASA Ames Research Center (ARC))	The concept of rapidly responding to a potential threat by making available some specifically targeted reconnassiance system is often discussed, however, less work has been done on the specific architecture of needs, and less on a spacecraft or ground asset which can fullfil that need	Properties for Assessing impact threats: Orbit elements and uncertainties, Size, Spectral class, Evidence for multiple objects. Properties for planning an effective response: Precise orbit, Mass, Presence of multiple objects, Detailed shapre, Material properties, Internal structure, Mineral composition, Detailed surface topology, Rational state, Dust/coma/ejecta			X		X	Ground & Space-Based Observation				
Characterization	Apophis 2029 Planetary Defense Application	2018	A-Team	Kim Reh, Jim Bell, Illana Gat, Justin Boland, Randii Wessen		The asteroid, Apophis, will have a close flyby to Earth on Friday, April 13th, 2029. Given this close Earth approach, there is a great opportunity to study this asteroid's interior, seismic behavior from Earth's gravitational forces, and other characteristics that will help us learn understand other asteroids. The Solar System Program and Planetary Defense are interested in a mission to study Apophis in 2029 (and perhaps sooner). This study is to examine the trade space of science, mission concepts, technology feasibility, and utilization of current and future ground and space-based assets to define a compelling high science return and low-cost mission for the opportunity to study the before, during, and after approach to propose to the Solar System Science programs and Planetary Defense.		Apophis	X			Ground & Space-Based Observation				
Characterization	Visible and near-infrared observatinos of asteroid 2012 DA14 during its closest approach of February 15, 2013	2013	Leon	Ortiz, Pinilla-Alonso, Cabrera-Lavers, Alvarez-Candal, Morales, Duffard, Santos-Sanz, Licandro, Perez-Romero, Lorenzi, Cikota		Ground-based telescopes used to observe an object passing near to the Earth to better understand its physical properties. This article describes what was done for these observations and their findings. Key results show the asteroid is an L-Type (rare composition) similar to that of carbonaceous chondrites and that it is very elongated and irregularly shaped with an equivalent diameter of aroudn 18m.		2012 DA14				Ground Based Observation				
Characterization	Time-seriese photmetry of Earth flyby asteroid 2012 DA14	2013	Terai	Urakawa, Takahashi, Yoshida, Oshima, Aratani, Hoshi, Sato, Ushioda, Oasa		Observations of 2012 DA14 to study the solar phase angle which varied widely around its closest approach but was almost constant during the following night. Key results show the phase curve slope of the asteroid is significantly shallower than those of other L-Type asteroids.		2012 DA14				Ground Based Observation				
Characterization	The northern χ-Orionid meteroid stream and possible association with the potentially hazardous asteroid 2008XM1	2013	Madiedo	Trigo-Rodriguez, Williams, Ortiz, Cabrera		This article explores the orbital data of the χ-Orionid fireball observed on 12/06/2011 to assess its potential parent body. The article points out two likely candidate parent bodies: 2002XM35 and 2008XM1. Key results, it is likely 2008XM1 that is the parent body but further analysis is needed.		2008XM1								
Characterization	Spectral properties of eight near-Earth asteroids	2011	Popescu	Birlan, Binzel, Vernazza, Barucci, Nedelcu, DeMeo, Fulchignoni		Attempting to characterize physical properties based on visible and infrared wavelength observations. Key results results show most of the target bodies are S-Type asteroids and have primitive properties		1917 (Cuyo), 8567 (1996 HW1), 16960 (1998 QS52), 164400, 188452 (2004 HE62), 2001 SG286, 2010 TD54				Ground-Based Observation (Visible and IR)				
Characterization	Origin of the near-Earth asteroid Phaethon and the Geminids meteor shower	2010	Leon	Campins, Tsiganis, Morbidelli, Licandro		Exploration of the dynamical connection between Pallas & Phaethon to understand their origin. Key results found nine asteroids belonging to the Pallas family have visible spectra that are different from that of Pallas and strikingly similar to that of Phaethon. Pallas is the most likely paraent body of Phaethon and the associated Geminids meteor shower stream.		Pallas, Phaethon (3200)				Ground-Based Observation (Visible and IR)				

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Mitigation	Optimization of deflection of a big NEO through impact with a small one	2014	Zhu	Huang, Wang, Niu, Wu		Paper describes how to select a small body to use for kinetic impact from a database. Diverting the small body is done using a spacecraft kinetic impact. Comparisons are shown between using the same spacecraft to kinetically impact the larger NEO. A specific case study is performed for the deflection fo the Apophis NEO by diverting NEO 2004HE as the smaller body; the efficiency of the method is assessed. Results suggest furhter study is warranted.		**Apophis**					**X**			
Characterization	On the possible values of the orbit distance between a near-Earth asteroid and the Earth	2012	Gronchi	Valsecchi		The article considers all possible trajectories of a NEA corresponding to the entire set of heliocentric orbital elements with perihelion distance q≤1.3 au and eccentricity of e≤1 (NEA class). The results show the maximal orbit distance between an object in the NEA class and the Earth is attained by a parabolic orbit with apsidal line orthogaonal to the ecliptic plane. The paper shows certain geometric properties of confocal conics can explain some selection effects in the orbital distribution of the known NEAs.						**Orbital simulations based on survey data.**				
Characterization	Nonextensive distributions of asteroid rotation periods and diameters	2011	Betzler	Borges		Investigates the distribution of asteroid rotationperiods from different regions of the solar system and diameter distributions of NEAs. Key results: the rotation periods of asteroids follow a q-Gaussian quth q=2.6 regardless of taxonomy, diameter, or region of the solar system. According their distribution of diameters, there are expected to be 994 ± 30 NEAs with a diameter greater than 1km.						**Analysis based on Planetary Database Sytem (PDS)**				
Characterization	Near-Earth Objects: Community white paper to the Planetary Decadal Survey, 2013-2022	2013	Nolan			This highlights the importance of, and risks associated with, our lack of understanding and collected data on the physical characterstics of NEOs.										
Characterization	Near-Earth object hazardous impact: A multi-criteria decision making approach	2016	Sanchez-Lozano	Fernandez-Martinez		The paper provides a multi-criteria decision making (MCDM) approach to classifying hazardous NEOs.						**Dataset meta-analysis**				
N/A	*Near-Earth object 2004CK39 and its associated meteor showers*	*2012*	*Babadzhanov*	*Williams, Kokhirova*	*Orbital analysis and calculations that match up the NEO and an associated meteor shower.*	*Investigation of the orbital evolution of 2004CK39. Research shows 2004CK39 is a dormant or dead comet nucleus and it is a quadruple crossr of the Earth's orbit.*		***2004CK39***				***Dataset meta-analysis***				
Characterization	Near-Earth asteroid (3200) Phaethon: Characterization of its orbit, spint state, and thermophysical parameters	2016	Hanus	Delbo, Vokrouhlicky, Pravec, Emery, Ali-Lagoa, Bolin, Devogele, Dyvig, Galad, Jedicke, Kornos, Kusnirak, Licandro, Reddy, Rivet, Vilagi, Warner		Attempting to create a reliable convex shape model of Phaethon by interpreting space- and ground-based thermal IR and sSpitzer spectra data. Key results include a new convex shape model and rotational state of Phaethon.		**Phaethon**				**Dataset meta-analysis**				
Characterization	Jadeite in Chelyabinsk meteorite and the nature of an impact event on its parent body	2014	Ozawa	Miyahara, Ohtani, Koroleva, Ito, Litasov, Pokhilenko		Paper investigates the formational history of NEOs using the Chelyabinsk asteroid as an example. Evidence shows the parent body of the Chelyabinsk meteor may have been impacted by an object larger than 0.19km at a speed of at least 0.4km/s which may have caused the spearation of the Chalyabinsk meteorite.						**Impact meteorite sample analysis**				
Characterization	Impact probability computation of near-Earth objects using Monte Carlo line sampling and subset simulation	2020	Romano	Losacco, Colombo, Lizia	Characterization (risk) method	Paper uses two Monte Carlo (MC) based sampling methods to improve the performance of standard MC analyses in the context of asteroid impact risk assessment.						**Dataset meta-analysis**				
Characterization	Rapid Response to Targets of Opportunity	2020	Donitz	Costillo-Rogez; Bhaskaran; Matousek; Chien; Karapetian; Mages		Three mission architectures identified to enable rapid response to targets: in-spaced storage; on-ground storage; and build-after-detection.					**X**					
Characterization	An investigation of the low-ΔV near-Earth asteroids (341843) 2008 EV5 and (52381) 1993 HA: Two suitable targets for the ARM and MarcoPolo-M5 space missions	2016	Perna	Popescu; Menteiro; Lantz; Lazzaro; Merlin	Background info and method are still useful, but the targets are specific to ARM's (defunct mission) timefrmame.	Paper aims to further characterize the physical properties of the two targets for sample return space missions. Key results: new observations are in agreement with the C-type classification of 2008 EV5 (a requirement for the ARM mission) and discovered the synodic rotation period of 1993 HA is 4.107 ± 0.002 h, a value that is optimal for the execution of the sample return mission.		**2008 EV5; 1993 HA**				**Ground-Based Observations**				

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Characterization	280 one-opposition near-Earth asteroids recovered by the EURONEAR with the Isaac Newton Telescope	2017	Vaduvescu	L. Hudin4, T. Mocnik1, F. Char5, A. Sonka6, V. Tudor1, I. Ordonez-Etxeberria1,7, M. Díaz Alfaro1,8, R. Ashley1, R. Errmann1, P. Short1, A. Moloceniuc9, R. Cornea9, V. Inceu10, D. Zavoianu11, M. Popescu6,12, L. Curelaru9, S. Mihalea9, A.-M. Stoian13, A. Boldea14,15, R. Toma16,9, L. Fields16, V. Grigore9, H. Stoev1, F. Lopez-Martinez1,17, N. Humphries1, P. Sowicka1,18, Y. Ramanjooloo1, A. Manilla-Robles1, F. C. Riddick1, F. Jimenez-Lujan1, J. Mendez1, F. Aceituno19, A. Sota19, D. Jones2,3, S. Hidalgo2,3, S. Murabito2,3, I. Oteo20,21, A. Bongiovanni2,3, O. Zamora2,3, S. Pyrzas2,3,22, R. Génova-Santos2,3, J. Font2,3, A. Bereciartua2,3, I. Perez-Fournon2,3, C. E. Martínez-Vázquez2,3, M. Monelli2,3, L. Cicuendez2,3, L. Monteagudo2,3, I. Agulli2,3, H. Bouy23,24, N. Huélamo24, M. Monguió25, B. T. Gänsicke26, D. Steeghs26, N. P. Gentile-Fusillo26, M. A. Hollands26, O. Toloza26, C. J. Manser26, V. Dhillon27,2, D. Sahman27, A. Fitzsimmons28, A. McNeill28, A. Thompson28, M. Tabor29, D. N. A. Murphy30, J. Davies31, C. Snodgrass32, A. H. M. J. Triaud33, P. J. Groot34, S. Macfarlane34, R. Peletier35, S. Sen35, T. ˙Ikiz35, H. Hoekstra36, R. Herbonnet36, F. Köhlinger36, R. Greimel37, A. Afonso38, Q. A. Parker39,40, A. K. H. Kong41, C. Bassa42, and Z. Pleunis43	Useful only for cataloging / dataset anslysis	Dataset curation and analysis for a catalog.						**Ground Based Observation**				
Characterization	The Future of Planetary Defense A. Mainzer	2017	Mainzer			This article basically highlights the importance of expanding PD beyond NASA and how important it is to expand and rely on international collaboration.						**Ground Based Observation**				
N/A	From Project Management to Planetary Defense: Implementation of a Systems Engineering Approach Using Integrated Product and Process Development (IPPD)	2018	Alaa Adnan Hussein		Rapid-Response Team Coordination	This represents a simulated response task, given to a large multi-national group, in order to study how the team could organize themselves and structure a response.										
Mitigation	A benchmarking and sensitivity study of the full two-body gravitational dynamics of the DART mission target	2020	Agrusa	Richardson; Davis; Fahnestock; Hirabayashi; Chabot; Cheng; Rivkin; Michel; DART Dynamics Working Group		Study performs high-fidelity rigid full two-body simulations of the mutual dynamics of the kinetic impactor system. Key results: the orbit phase (angular position or true anomaly) of the secondary is highly sensitive to the initial rotation phase of the primary, making prediction of the secondary's location from numerical simulation challenging. Also shown: the DART impact should induce significant free and forced librations on the secondary.		**Binary Asteroid 65803 - Didymos**					**X**			
Char / Mitig	National Near-Earth Object Preparedness Strategy & Action Plan	2018	INTERAGENCY WORKING GROUP FOR DETECTING AND MITIGATING THE IMPACT OF EARTH-BOUND NEAR-EARTH OBJECTS of the NATIONAL SCIENCE & TECHNOLOGY COUNCIL		Strategic plans. Need to download file and open as PDF. Word online has trouble with this one for some reason.	This is the White House Report (written 2018 by IWGFDMIENEO of the NSTC) which provides a strategy and action plan for the US and outlines expected international collaborations for NEO PD.										
Characterization	The Hera Mission: Multiple near-Earth asteroid sample return	2002	Sears	Allen; Britt; Brownlee; Franzen; Gefert; Gorovan; Pieters; Preble; Scheeres; Scott		Hera mission review as of fall 2002.	**Hera**		**X**							
Characterization	The Hera near-Earth asteroid sample return mission: Science requiremetns of the sample collector	2003	Sears	C.C. Allen b, M.S. Bell b, D. Bogard b, D. Britt c, D.E. Brownlee d, C. Chapman e, B.C. Clark f, R. Dissley g, M.A. Franzen a, J. Goldstein h, K. Nishiizumi i, L. Nyquist b, C.M. Pieters j, D. Scheeres k, E.R.D. Scott l, A. Treiman m		Review of the nature of sample to be collected by a touch-and-go collector to allow maximum science return while utilizing the simplest engineering designs. The article summarizes the results of a small workshop convened to discuss the topic.	**Hera**		**X**							
Characterization	Creep Stability of the DART/Hera mission target 5803 Didymos: II The role of cohesion	2021	Zhang	Michel; Richardson; Barnouin; Agrusa; Tsiganis; Manzoni; May		Analysis of the cohesive strength of the fast-spin-primary Didymos. The paper investigates formation and evolution of this system, structural stability, and cohesive strength based on current observational information.		**65803 Didymos-Dimorphos**				**Ground-Based Observations**				
Mitigation	Roadmap for Earth Defense Initiatives	2015	Hussein	International Space University Space Studies Program		Recommendations in five areas of planetary defense including detection & tracking, deflection techniques, global collaboration, outreach and education, and evacuation & recovery.						**Ground-Based Observations**		**X**		

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N/A	REPORT ON NEAR-EARTH OBJECT IMPACT THREAT EMERGENCY PROTOCOLS	2021	INTERAGENCY WORKING GROUP FOR DETECTING AND MITIGATING THE IMPACT OF EARTH-BOUND NEAR-EARTH OBJECTS of the NATIONAL SCIENCE & TECHNOLOGY COUNCIL		Protocols for threat mitigation. Doesn't really go into how the threat will be mitigated, just a description of steps to take after detection of a PHO. Description is what federal agencies and politicians would do in the case a PHO were detected.	Report contains several flow charts for decision trees for various situations pertaining to NEO/PHOs. Also contains some background information in the form of Q&A in Appendix B (pg 22)										
Mitigation	High energy density soft X-ray momentum coupling to comet analogs for NEO mitigation	2016	J.L.Remo	R.J.Lawrence, S.B.Jacobsen, M.D.Furnish	Ground based system using X-ray momentum coupling (instead of traditional laser/ablation style) for NEO deflection.	Ground based system using X-ray momentum coupling (instead of traditional laser/ablation style) for NEO deflection.						**Ground-Based Observations**				**X**
Mitigation	Decision program on asteroid threat mitigation	2009	Schweickart			Provides explicit criteria developed in order to assist the decision making process of NEO Planetary Defense, and highlights major pitfalls in existing decision making architecture including the lack of multinational collaboration.										
Mitigation	Mission concepts and operations for asteroid mitigationinvolving multiple gravity tractors	2013	Foster	Julie Bellerose, David Mauro, Belgacem Jaroux		Analysis and justification for multi-spacecraft gravity tractor architecture.									**X**	
Char / Mitig	Hazardous near Earth asteroid mitigation campaign planningbased on uncertain information on fundamental asteroidcharacteristics	2014	Sugimoto	G Radice, M Ceriotti, JP Sanchez		Report has many facets including characterization tree flow chart, uncertainty quantification, and application of those towards all of the different methods of deflection.							**X**	**X**	**X**	**X**
Mitigation	Hypervelocity nuclear interceptors for asteroid disruption	2012	Wie			This is a novel combined nuclear kinetic impactor.								**X**		**X**
Mitigation	Evaluation of NEA deflection techniques. A fuzzy Multi-Criteria Decision Making analysis for planetary defense	2020	Sanchez-Lozano	Fernandez-Martinez; Saucedo-Fernandez; Trigo-Rodriguez		Assessment of four deflection techiques to be applied to mitigate a hypotheical impact with an asteroid smaller than 250 meters in diameter (the most common class of asteroid to have impacts that occure in timescales less than 100 years). Key results show kinetic impactor is the best option to deflect mid-size NEAs but larger asteroids are better mitigated by enhanced gravity tractor and laser ablation.							**X**		**X**	**Ion Beam, Enhanced Gravity Tractor, Laser Ablation**
Characterization	Effects of NEO composition on deflection methodologies	2012	Sugimoto	Radice, J.P. Sanchez		The study aims to evaluate the reliability and robustness of different deflection methodologies subject to uncertainty in the asteroid composition. A typical S-type rubble pile configuration is used as the baseline asteroid composition for the study.							**X**	**X**		**Solar sublimation**
Characterization	THE PHYSICAL CHARACTERIZATION OF THE POTENTIALLY HAZARDOUS ASTEROID 2004 BL86: AFRAGMENT OF A DIFFERENTIATED ASTEROID	2015	Reddy	Reddy1,12, Bruce L. Gary2, Juan A. Sanchez1,12, Driss Takir1,12, Cristina A. Thomas3,12, Paul S. Hardersen4,12,Yenal Ogmen5, Paul Benni6, Thomas G. Kaye7, Joao Gregorio8, Joe Garlitz9, David Polishook		Spectrographic analysis of PHA (357439) 2004 BL86 during a close flyby of Earth in an attempt to link 2004 BL86 directly to Vesta, the assumed parent body.		**ASTEROID 2004 BL86**				**Ground Based Observation**				
Mitigation	Modeling tether–ballast asteroid diversion systems, includingtether mass and elasticity	2014	French	Mazzoleni		Study to determine if a tether-ballast system is capable of diverting Earth-Threatening asteroids. Detailed parametric studies are presented which illustrate how system performance depends on tether mass and elasticity. Key results include dangers imposed by periods during which the tether goes slack and ways to avoid this.										**Tether-Ballast System**
N/A	The European Union funded NEOShield project: A global approach to near-Earth object impact threat mitigation	2012	Harris	M.A. Baruccib, J.L. Canoc, A. Fitzsimmonsd, M. Fulchignonib,I,S.F. Greene, D. Hestrofferf, V. Lappasg, W. Lorkh, P. Micheli, D. Morrisonj,D. Paysonk,1, F. Sch¨aferl		This paper describes some results from the NEOShield investigation which was an EU commisioned program to help understand the open questions related to NEO risks.										
Char / Mitig	Physical properties of Near-Earth Objects thatinform mitigation	2012	Michel			This report details how different aspects of NEO characterization provide information for downselection of an appropriate mitigation method. It provides a cross-chart identifying the characterization criteria necessary for each mitigation type to be successful.							**X**	**X**	**X**	**X**
Mitigation	Conceptual design of a hypervelocity asteroid intercept vehicle(HAIV) and its flight validation mission	2013	Pitz	B. Kaplingern, G. Vardaxis, T. Winkler, B. Wie		This paper describes the conceptual development and design ofa baseline HAIV system and its flight validation mission architecture for three missioncost classifications (e.g., $500 M, $1 B, and $1.5 B). This is a novel combined nuclear kinetic impactor.								**X**		
Mitigation	Activities in Europe related to the mitigation of the threatfrom near-Earth objects	2014	Koschny	Drolshagen		This paper summarizes the current NEO-related activities within ESA's Space Situational Awareness programme. I										
Mitigation	Deflection of Fictitious Asteroid 2017PDC: Ion Beam vs. Kinetic Impactor	2017	Bombardelli	Claudio Bombardelli,Emilio José Calero,Juan Luis Gonzalo Detlef Koschny,Javier Roa, John Brophy		Mission scenarios for the deflection of fictitious asteroid 2017 PDC are investigated. Two deflection options, kinetic impactor (KI) and ion beam shepherd (IBS), are studied and compared on the basis of deflection performance, safety, as well as mission schedule and political aspects.		**2017 PDC**					**X**			**Ion Beam Deflection**

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Mitigation	CHARACTERISTICS OF A HIGH-POWER ION BEAM DEFLECTION SYSTEM NECESSARY TO DEFLECT THE HYPOTHETICAL ASTEROID 2017 PD	2017	Brophy	Nathan Strange, Dan M.Goebel, Shawn Johnson, Daniel Mazanek, David Reeves		This paper deals with a hypothetical asteroid 2017 PDC, that is the subject of an emergency response exercise, with just over ten years to implement a deflection approach. The analyses herein allocates four years to the design, fabrication, assembly, test and launch of a notional high-power Ion Beam Deflection (IBD) vehicle to meet a launch readiness date no later than May 2021.		2017 PDC								Ion Beam Deflection
Mitigation	Defending Against Asteroids and Comets	2015	Miller	Dearborn		Thissection discusses the technical considerations associated with options to prevent or mitigate such adisaster beyond a civil-defense response. The principal approaches to avert an impact include deflectingthe object and/or breaking it up and dispersing the pieces							X	X	X	X
Char / Mitig	Options and uncertainties in planetary defense: Impulse-dependent response and the physical properties of asteroids	2019	Dearborn	Megan Bruck Syala, Brent W. Barbeec, Galen Gislerb, Kevin Greenaughd,Kirsten M. Howleya, Ronald Leungc, Joshua Lyzhoftc, Paul L. Millera, Joseph A. Nuthc,Catherine S. Pleskob, Bernard D. Seeryc, Joseph V. Wasema, Robert P. Weaverb, Melak Zebenay		With the current state of technology, kinetic impactors are thepreferred but not the complete solution. If the time to impact is short, or the threatening body too large, nucleardeflection serves as afinal option. The nuclear approach is considered within the context of current capabilities, posing no need to test, as extantand well-understood devices are sufficient for the largest known Potentially Hazardous Objects (PHOs)								X		
Mitigation	Limits on the use of nuclear explosives for asteroid deflection	2012	Bruck Syal	Dearborn, Schultz		Successfully deflecting a small body via NED, while avoiding fragmentation, becomes a challenging problem when therequired kinetic energy increment is a substantial fraction of the body's potential. Thispaper addresses the challenge of preventing the production of substantial low-speeddebris while deflecting small bodies with an impulsive method								X		
Char / Mitig	Planetary Defense	2021	Bruck Syal			This slide set provides a comprehensive overview of characterization, existing measurements from spacecraft missions, and the related mitigation methods with a focus on NED efficacy.						X		X		
Mitigation	Options and uncertainties in planetary defense: Mission planning andvehicle design forflexible response	2017	Barbee	Megan Bruck Syalb, David Dearbornb, Galen Gislerc, Kevin Greenaughd,Kirsten M. Howleyb, Ron Leunga, Josh Lyzhofta, Paul L. Millerb, Joseph A. Nutha,Catherine Pleskoc, Bernard D. Seerya, Joseph Wasemb, Robert P. Weaverc, Melak Zebenay		This paper is part of an integrated study by NASA and the NNSA to quantitatively understand the response timeframe should a threatening Earth-impacting near-Earthobject (NEO) be identified. The study compares NED and Kinetic Impactor solutions.										
Characterization	NEOCAM Near-Earth Object Camera	2016	Mainzer	Watkins, Abell, Bauer, Boslough, Bottke, Brozovic, Buratti, Carey, Chesley, Chodas, Cutri, DeMeo, Eisenhardt, Emery, Fernandez, Forrest, Grav, Masci, Maseiro, McMurtry,Nugent, Pipher, Reddy, Ressler, Spahr, Statler, Sykes, Wright.		The NEOCam observatory is a 50-cm, passively cooled, mid-infrared telescope designed to discover and characterize asteroids and comets. The mission will assess the hazard to Earth from near-Earth objects (NEOs) and will study the origin, evolution, and fate of the asteroids and comets.						X				
Characterization	Estimating Asteroid Mass from Optically Tracked R	2021	Christensen	Park, Bell		ThisPaper presentsthe feasibilityof estimating themass of an asteroidby tracking a number of probesejected froma host spacecraft during a flyby. The probes are designed to fly by at a much closer distance to the asteroid than thehost spacecraft, which lowers the risk of endangering the overall mission.						X				
Characterization	MODELING THE PERFORMANCE OF THE LSST	2018	Grav	Mainzer, Sparh		We have performed a detailed survey simulation of the LSST performance with regards to near-Earth objects (NEOs) using the project's current baseline cadence. The survey shows that if theproject is able to reliably generate linked sets of positions and times (a so-called "tracklet") using twodetections of a given object per night and can link these tracklets into a track with a minimum of3 tracklets covering more than a ~12 day length-of-arc, they would be able to discover 62% of thepotentially hazardous asteroids (PHAs) larger than 140 m in its projected 10 year survey lifetime. Thiscompleteness would be reduced to 58% if the project is unable to implement a pipeline using the twodetection cadence and has to adopt the four detection cadence more commonly used by existing NEOsurveys.						X				

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						103			Count: 41	Count: 5	Count: 16	Count: 50	Count: 18	Count: 13	Count: 6	Count: 17
N/A	GENESIS HALO ORBIT STATION KEEPING DES	2000	Wiliams	Barden, Howell, Lo, Wilson		As the fifth mission of NASA's Discovery Program, Genesis is designed to collect solar wind samples for approximately two years in a halo orbit near the Sun-Earth LI Lagrange point for return to the Earth. The design of the maneuvers required for the station keeping in the halo orbits is described. An overview of the Genesis mission is provided with a brief description of the Genesis spacecraft and operational constraints, and a discussion of the contingency plans in the event of spacecraft or ground system anomalies.			X			L1 Loitering, libration				
N/A	LONG TERM MISSIONS AT THE SUN-EARTH LIE	2011	Roberts			Three heliophysics missions □the Solar Heliospheric Observatory (SOHO), the Advanced Composition Explorer (ACE), and the Global Geoscience WIND□have been orbiting the Sun-Earth interior libration point L1 conti-nuously since 1996, 1997, and 2004, respectively. The L1 orbits and the mission histories of the three spacecraft are briefly reviewed, and the station-keeping techniques and orbit maneuver experience are discussed.			X			L1 Loitering, libration				
Characterization	ARCHITECTURE OF A FAULT-TOLERANT AND V	2018	Frazier	Rice, Mitchell		In this paper we pre-sent a system architecture as part of a proposed mission to Triton, called "Trident". The concept includes multiple layers of insurance against one or more failures while still achieving a successful flyby. Elements of this robustness include: flyby mission design, large timing margins built into the encounter sequence, multiple redundant science obser-vations with adequate data storage, an instrument suite providing overlapping measure-ments, active redundancy, and conservative GN&C design			X							
Characterization	SMALL BODY GRAVITY FIELD ESTIMATION USI	2015	Hesar	Parker, McMahon, Born		This paper presents a new navigation technique for estimating the gravity field of a smallbody. The proposed technique takes advantage of autonomous onboard optical navigationsupplemented with in-situ satellite-to-satellite radiometric measurements. Simulated in-siturelative radiometric measurements are generated between a navigation satellite and a radiobeacon orbiting the asteroid 433 Eros. In general, relative observations alone are not suf-ficient to provide a unique orbit determination solution. However, taking advantage of theasymmetric gravity field of an asteroid by solving for its gravity field, relative measurementscan converge on a unique solution					X					
Char / Mitig	How Much Fault Protection is Enough – A Deep Im	2005	Barltrop	Kan		For the Deep Impact Project, a myriad of Fault Protection (FP) Monitors, Symptoms, Alarms and Responses is engineered into the spacecraft FP software, common and yet custom to the Flyby and Impactor mother-daughter spacecraft. Device faults and functional faults are monitored, which are mapped 1-to-n into FP Symptoms, per instance of the fault. Symptoms are then mapped n-to-1 to FP Alarms, further down mapped n-to-1 to FP Responses. Though the final statistics of 49 Monitors, 921 Symptoms, 667 Alarms, and 39 Responses appear to be staggering, it remains debatable whether the amount of on-board autonomous Fault Protection is sufficient and friendly to operate						X				
Char / Mitig	Deep Impact Navigation System Performance	2008	Frauenholz	Bhat, Chesley, Mastrodemos, Own, Ryne		Thisdefinitive work provides a mission overview, summarizes the navigation requirements, compares the achievednavigation performance with a baseline design that reflects in-flight updates, and identifies operational proceduresthat may benefit future comet-bound navigators.					X					

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									Rendezvous	Tour (Multi-Target)	Flyby (Single Target)	Other	Kinetic Impact	Nuclear Explosive Device (or Simulator)	Gravity Tractor	Other (laser, enhanced gravity tractor, etc)
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N/A	SOLARSYSTEMHUMANEXPLORATIONAIDEDBY LIBRATION-POINTORBITS, LUN		Dunham	Farquhar, Eismont, Chumacheno, Aksenoc, Fedorenko, Furfaro, Kidd, Mogk		igh-energy Earth orbits that can be drastically modified with lunar swingbys and small propulsive maneuversare used, especially near the collinear Sun-Earth and Earth-Moon libration points. Thiswork builds on ideas developed by the International Academy of Astronautics' exploration study group. The firsthumanmissionsbeyond low-Earth orbitmaygo to the vicinity of the translunar Earth-Moon libration point, although a lunar distant retrograde orbit (DRO), as envisioned as the destination for the small asteroid (or asteroidal boulder) returned by the proposed Asteroid Redirect Mission may also be used. This paper will concentrate on the next possible step, the first one into interplanetary space, a one-year return mission to fly by a Near-Earth Object (NEO). Details are presented of a trajectory that leaves a halo orbit about the Earth-Moon L2 libration point, then uses three lunar swingbys and relatively small propulsive maneuvers to fly by the approximately 200m asteroid 1994 XL1, and return to the Earth-Moon L2 halo orbit for a □V of only 432 m/s. Next,rendezvous missions to some other NEO's will be presented. Finally, trajectories to reach Mars, first to Phobos or Deimos, will be outlined.						**L1 Loitering, libration**				
Characterization	The population of near-earth asteroids revisited and	2021	Harris	Chodas		In this paper we update, extend, and improve upon the recent paper on Near-Earth Asteroid (NEA) population by Harris and D'Abramo (2015). We update the population estimate taking into account discoveries to August 3, 2020. Shortly after the previous paper was published, we identified a problem in our previous studies due to rounding off of absolute magnitude H by the Minor Planet Center to 0.1 magnitude that implicitly shifted our bin boundaries by 0.05 magnitude. Here we correct the problem by choosing H bin boundaries at 0.25–0.75 magnitude, rather than 0.00–0.50 magnitude thereby eliminating the round-off shift. We also introduce an updated model distribution of NEA orbits (Granvik et al. 2018) in our survey simulations. This new population model includes orbital distributions as a function of size, allowing us to test our presumption that distributions are homologous with respect to size.						**X**				
Mitigation	2019 Planetary Defense Conference: Impact Exerc	2019	Chodas			A hypothetical asteroid impact scenario will be presented at the 2019 IAA Planetary Defense Conference (PDC), to be held in College Park, Maryland, USA, April 29 - May 3, 2019. Although this scenario is realistic in many ways, it is completely fictional and does NOT describe an actual potential asteroid impact.										**X**
Mitigation	2021 PDC Exercise Final Inject: October 14, 2021(6 days to impact)Asteroid 2021 PDC Detected by Radar: Size Smaller Than Previously Thought, Impact Energy Likely About 40 Mt	2021	Chodas			An asteroid is discovered on April 19, 2021, at apparent magnitude 21.5, and confirmed the following day. It is assigned the designation "2021 PDC" by the Minor Planet Center. (To reinforce the fact that this is not a real asteroid, we are using three letters in the designation, something that would never be done for an actual asteroid.)										**X**
Mitigation	Quantifying the Risk Posed by Potential Earth Impa	2002	Chesley	Chodas, Milani, Valsecchi, Yeomans		Predictions of future potential Earth impacts by near-Earth ob-jects (NEOs) have become commonplace in recent years, and therate of these detections is likely to accelerate as asteroid surveyefforts continue to mature. In order to conveniently compare andcategorize the numerous potential impact solutions being discov-ered we propose a new hazard scale that will describe the risk posedby a particular potential impact in both absolute and relative terms.										**X**

									Characterization				Mitigation			
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Char / Mitig	Comet Interceptor: An ESA mission to a Dynamica	2020	Sanchez	Jones, Snodgrass,		Comet Interceptor (Comet-I) was recently selected as ESA's first fast-track class mission and aims to explore a pristine comet, which will ideally be visiting the inner Solar System for the first time. Comet-I will hitch a ride to a Sun-Earth L2 quasi-halo orbit, as a co-passenger in ESA's M4 ARIEL's launch, in 2028. It will then remain there waiting for the right departure conditions to definitively leave the L2 point and intercept a newly discovered comet. Comet-I will be the first mission to be design and, possibly launched, without an identified target. Nevertheless, a Monte Carlo analysis modelling the uncertainties of the long period comet population and the spacecraft transfer capabilities demonstrate the high likelihood of completing the mission within 6 years. A few days before the closest approach Comet-I will release two small independent probes (~30 kg each) into fly-by paths with close approach distances in the order of a few hundred kilometres, while the main spacecraft (~700 kg) will take a safer path (~1000 km) to protect it from the dust environment. Comet-I will thus involve three spacecraft elements working together to ensure a low-risk, bountiful, interdisciplinary scientific return through unprecedented multipoint measurements.			**X**		**X**	**L1 Loitering, libration**				**X**
N/A	GOES-R spacecraft/instrument overview	2010	**Walsh**			The Geostationary Operational Environmental Satellite-R (GOES-R) is a high performance evolutionary follow-on satellite system to the existing GOES-I/M and NOP series satellites currently operating over the Western Hemisphere.										
N/A	NOAA moves spare satellite in position over Atlanti	2012	Clark			NOAA on Monday began moving a backup weather satellite into position to replace an observatory knocked out of service in September										
N/A	GOES-14 (O) Moving into On-Orbit Storage Aroun	2009	Gutro			The Geostationary Operational Environmental Satellite named GOES-14, is being placed in on-orbit storage this month to await its call to duty.										
N/A	Autonomous Space Robotics:Enabling Technologie	2000	Hollander			Exploitation of robotic techniquesdeveloped for industrial applications can providecapabilities to perform fully autonomous on-orbitfueling, electrical system modification, and remotesimple assembly operations in space. Roboticoperations in a space environment need not precludemeeting high reliability standards imposed bytraditional space programs. Significant developmentis required, and common interface standards need tobe defined that will have the widest range ofapplicability to all potential users.										
N/A	How Long Does It Take to Develop and Launch Go	2015	Davis	Filip		This report is aimed at providing insight into government satellite acquisition processes to potentially identify focus areas to reduce satellite development timelines. There seems to be a perception that it takes 10 years or more to develop and launch a government satellite system										
Characterization	Initial characterization of interstellar comet 2I/Boris	2019	Guzik	Drahus, Rusek, Waniak, Cannizzaro, Pastor-Marazuela		Interstellar comets penetrating through the Solar System had been anticipated for decades1,2. The discovery of asteroidal-looking 'Oumuamua3,4 was thus a huge surprise and a puzzle. Furthermore, the physical properties of the 'first scout' turned out to be impossible to reconcile with Solar System objects4,5,6, challenging our view of interstellar minor bodies7,8. Here, we report the identification and early characterization of a new interstellar object, which has an evidently cometary appearance. The body was discovered by Gennady Borisov on 30 August 2019 ut and subsequently identified as hyperbolic by our data mining code in publicly available astrometric data.										
Char / Mitig	The European Space Agency'sComet Interceptor li	2019	Snodgrass	Jones, Snodgrass,		The European Space Agency (ESA) recently selectedComet Interceptoras itsfirst 'fast' (F-class) mission. It will be developed rapidly to share a launch withanother mission and is unique, as it will wait in space for a yet-to-be-discovered comet.			**X**			**L1 Loitering, libration**				
Char / Mitig	Request for Information for the provision of small s	2020	Nicola Rando			The objective of the Comet Interceptor (Comet-I) mission is to characterize a pristine comet by performing multi-point observations during a dedicated fly-by.			**X**			**L1 Loitering, libration**				

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Characterization	NEOSM Survey Cadence and Simulation	2020	Sonnett	, Mainzer, Grav, Spahr, Lilly, Masiero		The Near-Earth Object Surveillance Mission (NEOSM) is a planned space-based infrared mission that will nominally launch in 2025 and librate at the Earth-Sun L1 Lagrange point. The NEOSM Project was formulated to address the need to detect, catalog, and characterize near-Earth objects (NEOs) to support informed decision making for any potential mitigation activity. NEOSM detects NEOs, obtains high quality orbits for them, provides physical characterization of the NEOs and their source populations, and provides more detailed physical characterization for individual targets with significant impact probabilities. Specifically, NEOSM will detect, track, and characterize 2/3 of potentially hazardous asteroids (PHAs) larger than 140m						X				
Characterization	Finding Near Earth Objects from Space with NEOV	2019	Mainzer			Asteroids and comets periodically impact the Earth; the key questions are how often do they do so, and with what energy? Systematic telescopic searches to find, track, and characterize these objects are key to addressing these questions. The issue of identifying when a potential impact might occur is answered by discovering the objects and tracking them for sufficient time to enable a reliable prediction of close approaches to be made over the next century.						X				
Char / Mitig	Radar observations and a physical model of binary	2020	Naidu	Benner, Brozovic, Nolan, Ostro, Margot, Giorgini, Hirabayashi, Scheeres, Pravec, Scheirich, Magric, Jao		Near-Earth asteroid Didymos is a binary system and the target of the proposed Double Asteroid Redirection Test (DART) mission (Cheng et al., 2016), which is a planetary defense experiment. The DART spacecraft will impact the satellite, causing changes in the binary orbit that will be measured by Earth-based observers	Dart	Didymos	X			X	X			
Char / Mitig	European component of the AIDA mission to a binary asteroid: Characterization and interpretation of the impact of the DART mission	2018	Michel	Kueppers, Sierks, Carnelli, Cheng, Mellab, Granvik, Kestila, Kohout, Miononen, Nasila, Penttila, Tika, Tortoa, Ciarletti, Herique, Murdoch, Asphaug, Karatekin		The European component of the joint ESA-NASA Asteroid Impact & Deflection Assessment (AIDA) mission has been redesigned from the original version called Asteroid Impact Mission (AIM), and is now called Hera. The main objectives of AIDA are twofold: (1) to perform an asteroid deflection test by means of a kinetic impactor under detailed study at NASA (called DART, for Double Asteroid Redirection Test); and (2) to investigate with Hera the changes in geophysical and dynamical properties of the target binary asteroid after the DART impact	Hera	Didymos	X			X	X			
Char / Mitig	Impact modeling for the Double Asteroid Redirection Test (DART) mission	2020	Rainey	Stickle, Cheng, Rivkin, Chabot, Barnouin, Ernst,		We present results from numerical simulations of the DART impact using the CTH shock physics code with 2D homogenous asteroid models.	Dart	Didymos	X			X	X			
Characterization	OSIRIS-REx: Sample Return from Asteroid (10195	2017	Lauretta	Balram-Knutson, Boynton, d'Aubigny, DellaGuistina, Enors, Golish, Hergenrother, Howell, Bennett, Morton, Nolan, Rizk, Roper,		ASA selected the Origins, Spectral Interpretation, Resource Identification, andSecurity–Regolith Explorer (OSIRIS-REx) asteroid sample return miion as the third mission in the New Frontiers program.	Osiris Rex	Bennu	X							
Characterization	Rosetta crashes into comet	2016	Gibney			he European Space Agency’s comet-orbiting Rosetta spacecraft was successful to the last. It crash-landed on the comet 67P/Churyumov–Gerasimenko within one minute of its scheduled impact time,	Rosetta	67p	X							
Char / Mitig	Deep Impact – A Review of the World's Pioneering	2015	Henderson	Blume		On July 4th, 2005, in celebration of our nation's birthday, NASA's Deep Impact Impactor spacecraft collided with comet Tempel1 at 10 km/sec – marking the first hypervelocity impact of a celestial body by a human-made spacecraft. With closing speeds of 23,000 mph, the Impactor's active guidance system steered it to impact on a sunlit portion of the comet's surface.	Deep Impact	Tempel-1	X				X			
Char / Mitig	DEEP IMPACT: WORKING PROPERTIES FOR THE TARGETNUCLEUS – COMET 9P/TEMPEL 1	2004	BELTON	Meech, A'hearn, Groussin, LUCY MCFADDEN3,CAREY LISSE3,YANGA R. FERN´ANDEZ2,JANA PITTICHOV ´A2,HENRY HSIEH2,JOCHEN KISSEL4,KENNETH KLAASEN5,PHILIPPE LAMY6,DINA PRIALNIK7,JESSICA SUNSHINE8,PETER THOMAS9and IMRE TOTH		In 1998, Comet 9P/Tempel 1 was chosen as the target of theDeep Impactmission (A’Hearn,M. F., Belton, M. J. S., and Delamere, A.,Space Sci. Rev.,2005) even though very little was knownabout its physical properties. Efforts were immediately begun to improve this situation by theDeepImpactScience Team leading to the founding of a worldwide observing campaign (Meechet al.,Space Sci. Rev.,2005a).	Deep Impact	Tempel-1	X				X			
Characterization	Hayabusa2Mission Overview	2017	Watanabe	Tsuda, Yoshikawa, Tanaka, Saiki, Nakazawa		TheHayabusa2mission journeys to C-type near-Earth asteroid (162173) Ryugu(1999 JU3) to observe and explore the 900 m-sized object, as well as return samples col-lected from the surface layer.	Hayabusa-2	Ryugu	X							

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Characterization	The Stardust – a successful encounter with theremarkable comet Wild 2	2004	Brownlee	Anderson, Atkins, Bhaskaran, Cheuvront		On January 2, 2004 the Stardust spacecraftcompleted a close flyby of comet Wild2 (P81).Flying at a relative speed of 6.1 km/s within 237kmof the 5 km nucleus, the spacecraft took 72 close-inimages, measured the flux of impacting particles anddid in-situ compositional analysis of freshly releaseddust with a time-of-flight mass spectrometer.	**Stardust**	**Wild 2**	**X**			**X**				
Characterization	Stardust: Comet and interstellar dust sample return mission	2003	Brownlee	Andersone, Hanner, Newburn, Sekanina, Clark, Horz, Zolenssky, Kissel, McDonnell, Sandford, Tuzzolino		Stardust, the 4th Discovery mission launched in February 1999, will collect coma samples from the recently deflected comet 81P/Wild 2 on 2 January 2004 and return them to Earth on 15 January 2006 for detailed laboratory analyses. Stardust will be the first mission to bring samples back to Earth from a known comet and also the first to bring back contemporary interstellar particles recently discovered	**Stardust**	**Wild 2**				**X**				
Characterization	The Giotto mission to Comet Halley	1987	Reinhard			Giotto encountered Comet Halley on 14 March 1986, about four weeks after the comet's perihelion passage when it was most active. Giotto passed the comet nucleus at a distance of 600 km on the sunward side. The mission's scientific objectives, the spacecraft and the mission from launch to encounter are described	**Giotto**	**Halley**			**X**					
Characterization	The Near Earth Asteroid Rendezvous Missionto Asteroid 433 Eros:A Milestone in the Study of Asteroids and their Relationship to Meteorites	2002	McCoy	Robinson, Nittler, Burbine		A milestone in the study of asteroids occurred on 14 Feb 2000, when the NEARspacecraft entered orbit around the asteroid 433 Eros for a year of detailed observation ofthe geology, mineralogy and chemistry of the surface, before landing on the surface on 12Feb 2001 and conducting the first science on the surface of a small Solar System body.	**NEAR**	**Eros**	**X**							
Characterization	The NEAR Shoemaker mission to asteroid 433 Eros	2002	Prockter	Murchie, Cheng, Trombka		The Near Earth Asteroid Rendezvous (NEAR) mission inaugurated NASA's Discovery Program. It was the first mission to orbit an asteroid and made the first comprehensive scientific measurements of an asteroid's surface composition, geology, physical properties, and internal structure	**NEAR**	**Eros**	**X**							
Characterization	Technical Challenges and Results for Navigation of NEAR Shoemaker	2002	Williams			WTechnical Challenges and Results for Navigation of NEAR ShoemakerBobby G. Williamshen the NEAR Shoemaker spacecraft began its orbit about the asteroid 433 Eros on 14 February 2000, it marked the beginning of many firsts for deep space navigation.	**NEAR**	**Eros**	**X**							
Characterization	Halley Comet Missions	1986	Stelzried, Efron, Ellis			Description of the various missions to Halley's Comet in the 'Halley's Armada' time period inclding DSN interactions, TDA technology development, and navigation systems.		**Halley**			**X**					
Characterization	The near-Earth asteroid population from two decades of	2016	Tricarico			Determining the size and orbital distribution of the population of near-Earth aster-oids (NEAs) is the focus of intense research, with the most recent models convergingto a population of approximately1000NEAs larger than 1 km and up to approximately109NEAs with absolute magnitudeH <30. We present an analysis of the combinedobservations of nine of the leading asteroid surveys over the past two decades, andshow that for an absolute magnitudeH <17.75, which is often taken as proxy for anaverage diameter larger than 1 km, the population of NEAs is920±10, lower thanother recent estimates.						**X**				
N/A	NASA's Search for Asteroids to Help Protect Earth	2014	NASA NEO Program			NEOs are asteroids and comets with orbits that come within 28 million miles of Earth's path around the sun, and NASA has been studying them since the 1970s. NASA's NEO Observations Program, at NASA Headquarters in Washington, is responsible for the Agency's efforts at finding, tracking, and characterizing NEOs. The agency's Jet Propulsion Laboratory (JPL) in Pasadena, California, hosts the NEO Program Office for Headquarters.										
Characterization	An empirical examination of WISE/NEOWISE asteroid analysis and results	2018	Myhrvold			Asteroid observations by the WISE space telescope and the analysis of those observations by the NEOWISE project have provided more information about the diameter, albedo, and other properties of approximately 164,000 asteroids, more than all other sources combined	NEOWise					**X**				

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Characterization	Pan-STARRS – The PS1 & PS2 Wide Area Survey for NEOs Kenneth C. Chambers	2018	Chambers			he Panoramic Survey Telescope and Rapid Response System or Pan-STARRS is a wide field sky survey system developed at the University of Hawaii that now includes both the PS1 and PS2 telescopes and extensive cyber-infrastructure for image processing, machine learning, and very large hierarchical databases. The emergent Pan-STARRS infrastructure is described together with the survey goals for the next five years of Pan-STARRS static sky and time domain science						X				
N/A	Coupling of system resource margins through the use of electricpropulsion: Implications in preparing for the Dawn mission toCeres andVesta	2007	Rayman	Frashetti, Raymond, Russell		The Dawn project is progressing toward its 2007 launch on a mission to orbit main belt asteroids (1) Ceres and (4) Vesta.Designed to provide insights into important questions about the evolution of the solar system, Dawn will spend more than0.5 years in orbit about each of these bodies. This challenging mission is enabled by an ion propulsion system. In contrast tomissions that use conventional chemical propulsion, the use of this system creates a strong coupling of allowable flight systemmass and available power, thereb yrequiring different methods of managing these and other technical resources	Dawn	Ceres, Vesta	X							
Char / Mitig	Autonomous Navigation for Deep Space Missions	2012	Bhaskaran			Autonomous navigation (AutoNav) for deep space missions is a unique capability that was developed at JPL and used successfully for every camera-equipped comet encounter flown by NASA (Borrelly, Wild 2, Tempel 1, and Hartley 2), as well as an asteroid flyby (Annefrank). AutoNav is the first on-board software to perform autonomous interplanetary navigation (image processing, trajectory determination, maneuver computation), and the first and only system to date to autonomously track comet and asteroid nuclei as well as target and intercept a comet nucleus. In this paper, the functions used by AutoNav and how they were used in previous missions are described. Scenarios for future mission concepts which could benefit greatly from the AutoNav system are also provided			X	X	X	X	X	X		
Char / Mitig	Predicting Close Approaches and Estimating Impact Probabilities for Near-Earth Object	1999	Chodas	Yeomans		Recent popular movies have raised public consciousness of the very real possibility ofa comet or asteroid collision with the Earth. A news story last year further caught the public's eye when it implied that asteroid 1997 XFII had a distinct chance of hitting the Earth in the year 2028. The possibility of impact disappeared the very next day, and the public perceived either that astronomers had made mistaken calculations, or that the pre-discovery observations found that day had been responsible for the revised prediction. In fact, the original report of the possibility of impact in 2038 was due to an incomplete analysis.										
Char / Mitig	Potential Impac tDetection fo rNear-Earth Asteroids:The Case of 99942 Apophis (2004MN4)	2005	Chesley			OrbitdeterminationforNear-EarthAsteroidspresentsuniquetechnicalchallengesduet otheimperative ofearlydetectionandcarefulassessmentoftheriskposedb yspecificEarthcloseapproaches.Thisarticlepresentsa casestudyofasteroid99942Apophis,a 300-400meterobjectthat,fora shortperiodinDecember2004,heldanimpactprobabilityof morethan2%in2029.Now, withanorbitbasedonradarrangingandmorethana yearofopticalobservations,wecanconfidentlysaythatit willpasssafelybytheEarthin2029,althoughata distanceofonlyaboutsixEarthradiifromthegeocenter		**Apophis**								

C PERTINENT OPERATIONAL SCENARIOS

New considerations of operational scenarios provide context for the Planetary Defense system architectures studied for risk reduction in §4, and for the recommendations in §7.

C.1 THREAT POPULATIONS

The focus in this study is on NEOs as described in §1.1. These provide very convenient test articles that are readily accessible and have known trajectories, making them well-suited for a variety of demonstration missions.

The operational threat spectrum also includes potential impacts from long-period comets such as C/Hale-Bopp, and Inter-Stellar Objects (ISOs) such as 1I/Oumuamua and 2I/Borisov, both of which allow little warning time (Ounuamua was discovered after perihelion and Borisov was discovered just three months before its perihelion). In both cases, the statistical probability of Earth impact by an undiscovered object is much lower (~1%) than from NEOs (Stokes et al., 2017). Given that detection/mitigation of impact from ISOs and long-period comets is extremely challenging, and their relatively low impact probability, we currently assume acceptance of this risk for the next decade, and focus on the more manageable defense of NEOs. However, as planetary defense and survey technologies continue to improve, defense from ISOs and long-period comets may become more feasible.

C.2 OPERATIONAL MISSIONS

While not explicitly requested for this study, a full understanding of operational PD processes and decision-making protocols is useful to defining the key parameters for mitigation missions. When an NEO is first discovered, a very preliminary orbit is determined using astrometric measurements (position on the sky) and a rough estimate of the object size is made using photometric (brightness) measurements and a range of typical albedos (reflectivities). Follow-up observations are critical for refining the orbit and size estimates. The possibility of a future impact can often be detected after only a few days of observations, but continued tracking over the subsequent days, weeks and months are required to establish reasonably accurate orbit solutions. Impact probability is continually reassessed whenever new observations become available. If an object really is an impactor, the impact probability will generally increase over time, as the orbit solution improves and predictions of future close approaches become more accurate.

According to the proposed logic flow from the Interagency Working Group on NEO Impact Threat Emergency Protocols (2021) (Figure C-1), a threshold impact probability of 1% is used for either a warning issuance (if the asteroid size is between 10 and 50 m in diameter) or a recommendation for preparation of a reconnaissance mission (if the asteroid size is larger than 50 m in diameter and the potential impact is less than 50 years away). In this latter case, when the impact probability reaches 10% and the hazard is significant, the protocol recommends that preparations begin for a mitigation mission.

The mass of the threatening object is a critically important parameter that directly drives the deflection campaign design (if needed) and damage assessment modeling if mitigation is not possible. The primary purpose of a reconnaissance mission would be to obtain an accurate as possible estimate of the object mass. A secondary objective would be to obtain in situ tracking measurements that would enable the precise impact location of the undeflected asteroid trajectory to be pinpointed, if that has not already been accomplished through ground tracking. In some circumstances ground-based radar can estimate the size and shape of an asteroid allowing for a rough estimate of mass through assumptions on a range of bulk densities. But in most scenarios, obtaining an accurate mass estimate would require a dedicated reconnaissance mission (§4.2.2). The timeline for executing the decision tree in Figure C-1 is driven by the amount of warning time available between discovery of the object and the decision point in the first red circle.

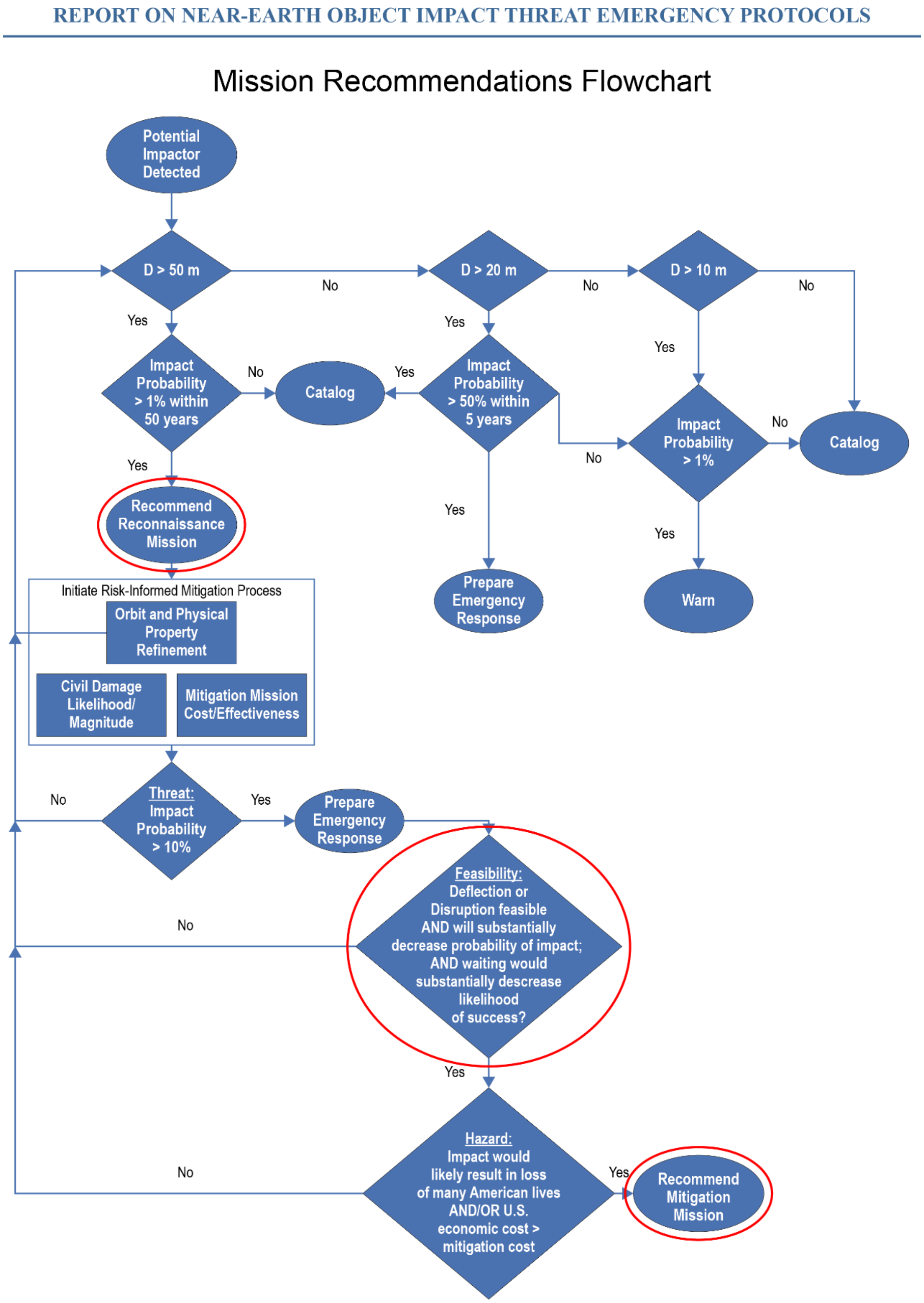

Figure C-1. Proposed decision tree for consideration of characterization / mitigation missions (Interagency Working Group on NEO Impact Threat Emergency Protocols, 2021). This study report is intended to support risk reduction at the key decision steps shown in red circles.

C.2.1 SHORT WARNING TIME

We define warning time as the time from when the impact probability reaches the threshold level for initiating a flight project to the time of the potential impact. For an asteroid estimate to be at least 50 m in diameter, the Space Mission Planning and Advisory Group (SMPAG), sets a threshold at 1% for beginning to plan space missions to the threatening asteroid (Figure C-1).

The concept of "short warning" means the minimum warning time within which a space mission could mitigate the threat. It involves the time for decision-making, spacecraft development, launch vehicle acquisition, launch, cruise, mitigation, and enough time for the mitigation effect to accumulate a large enough change in the asteroid orbital position so that it no longer will impact. The latter parameter is highly dependent on the asteroid orbital period, which in most cases lies in the range from less than a year to more than 3 or even 4 years. Experience from hypothetical impact exercises suggests the threshold warning time subjectively ranges roughly from 5 years to perhaps 12 years. "Short warning" herein also implies a crises situation in which multiple missions might be implemented in parallel, rather than having the time to develop them more optimally in series. The degenerate case of a warning time that is certainly too short to allow mitigation (e.g., ~1–3 years) are considered out of scope of this study and present a finite risk that must be managed via civil defense methods given present capabilities.

Short warning time situations are dominated by the implementation of preparatory programmatic/policy issues, in addition to a potentially dire technical situation. If a "rapid-response" project inventory was previously invested in and therefore available, then the development time required to launch may be reduced from 3–5 years to less than one year (see §5). Ideally the suite of missions would include at least one rendezvous to perform characterization, and potentially including adequate time for a variety of deflection attempts (JPL CNEOS, 2019b).

A fast flyby/reconnaissance mission (§4.2.2) could perform the critical tasks of (1) a rough estimate of the target size (and constraints on mass), (2) verify that it is a single body and not part of a doublet, and (3) provide a better asteroid orbit solution. If the warning time is less than ~5 years, then one or more rapid-response deflection mission(s) would need to be launched as soon as possible, with a focus on robust deflection techniques (potentially in parallel with the recon mission just noted). In the most extreme cases, it would be too late for deflection and the only remaining option would be disruption and destruction to the greatest degree possible, while accepting the risk of multiple, smaller impacts (Bruck Syal et al., 2013; Barbee et al., 2018). Performance or demonstration of deliberate disruption/destruction is an extreme case that is not addressed in this study, but has been examined previously (Miller & Dearborn, 2015).

These types of scenarios are rehearsed biennially as part of the Planetary Defense Conference series and are highly illustrative of the challenges in rapid responses. For example, Figure C-2 shows how a newly-detected asteroid could recede far from Earth after many months and get too faint to be observed, or approach too close to the direction of the Sun; orbit updates may be suspended for many months until the asteroid comes back into view. In this example, many additional observations are collected during 2019 to reduce the b-plane uncertainty to estimate a 70% probability of impact before the asteroid becomes invisible in 2020. Since the 10% threshold has been passed, emergency procedures are enacted (Figure C-1). In late 2020, more observations are available and the uncertainty ellipse shrinks to less than one Earth radius and the impact point is indeed on the Earth.

At the same time, optimal deflection opportunities may be missed until an intercept can be launched (Figure C-3). A deflection of >2 Earth radii may initially be required because KI deflection relies on slowing down the target asteroid; this creates a "handedness" in the ability to move the impact point across the Earth disk. In the worst case if the impact point were near to edge of the Earth, we may be constrained to move it all the way across the disc to the other side (ref PDC 19). However as the impact is calculated with more accuracy in this example, it is learned that a deflection of less than 2 Re is required, but since the asteroid mass is still largely unknown the amount of velocity change (ΔV) that can be imparted by each KI mitigation mission is highly uncertain.

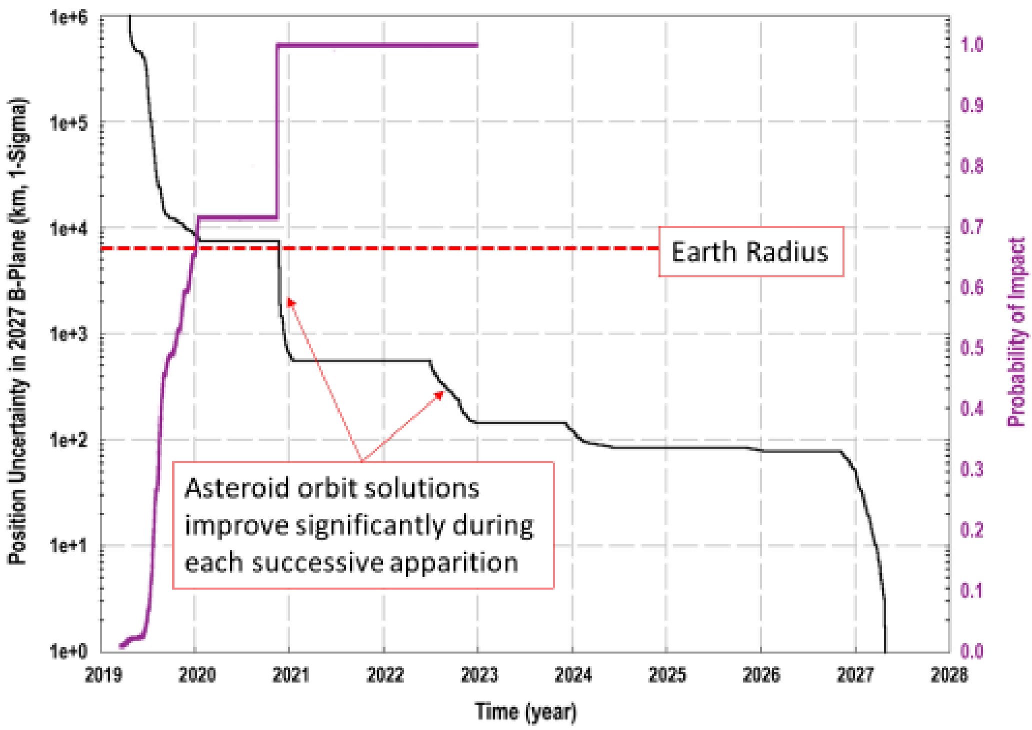

Figure C-2. Impact uncertainty refinement depends on target apparitions. As the b-plane errors are reduced we learn whether the asteroid will impact the Earth, and where. This is critical to mitigation planning (ref PDC 19).

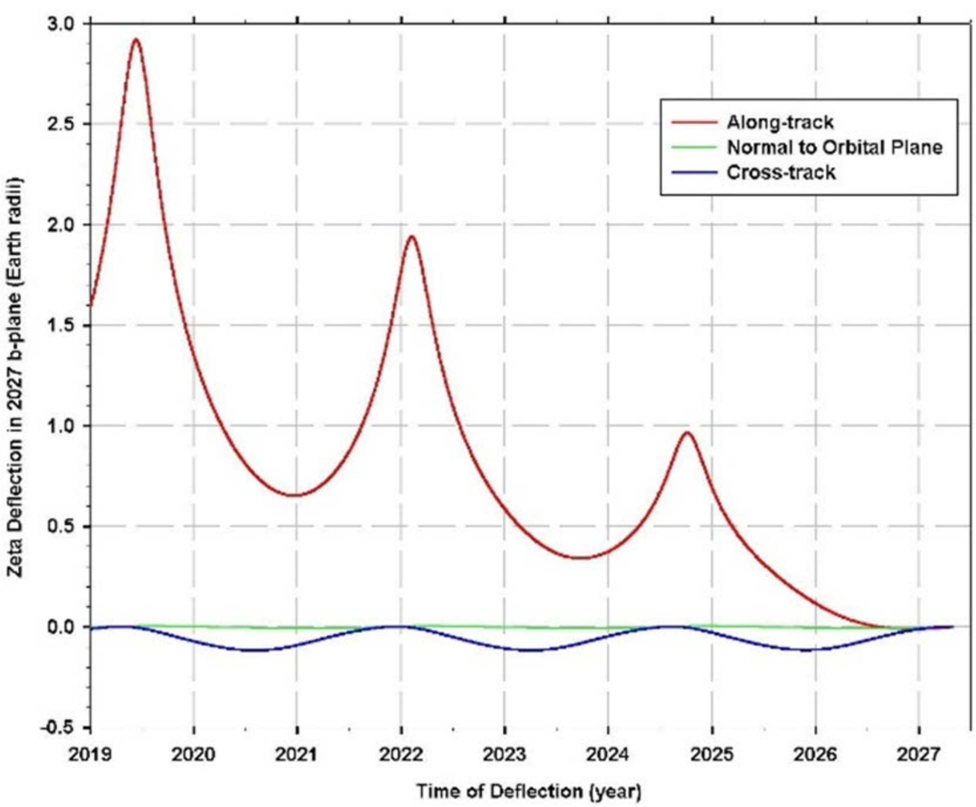

Figure C-3. Optimal asteroid deflection efficiency for a 1 cm/s ΔV depends on the direction in which it is applied (e.g., along-track), and the ability to apply the deflection at optimal times and as early as possible (ref PDC 19).

This dichotomy of delayed impact knowledge creates a rather stressful situation of not having adequate knowledge of impact probability while at the same time losing precious time in developing an effective response. In this exercise, eight different possible responses were developed as shown in Figure C-4 (although not all were exercised). This hypothetical case imagines design, build, and test of complex spacecraft and instruments in approximately half the time that is normally required. The longer schedules are normally the result of long-lead parts procurement and a careful design, integration, and test approach that experience shows leads to successful missions. Cutting this time in half implies a great deal of additional risk. If completely unprepared, the best-case solution to the lead-time problem would be to commandeer existing hardware from other flight projects and integrate components, subsystems and instruments that were not designed to go together and not designed for the critical mission to be launched in just a few years. While very illustrative in this exercise, this may be overly optimistic in a real-life situation and would imply higher likelihood of failure than typical. For this reason, rapid response capabilities should be developed from a deliberate preemptive/preventive programmatic posture (§5). The "build" portion of the timeline in Figure C-4 is then highly dependent on preceding programmatic assumptions and may range over many years, but this illustrates the need for improved rapid response capability.

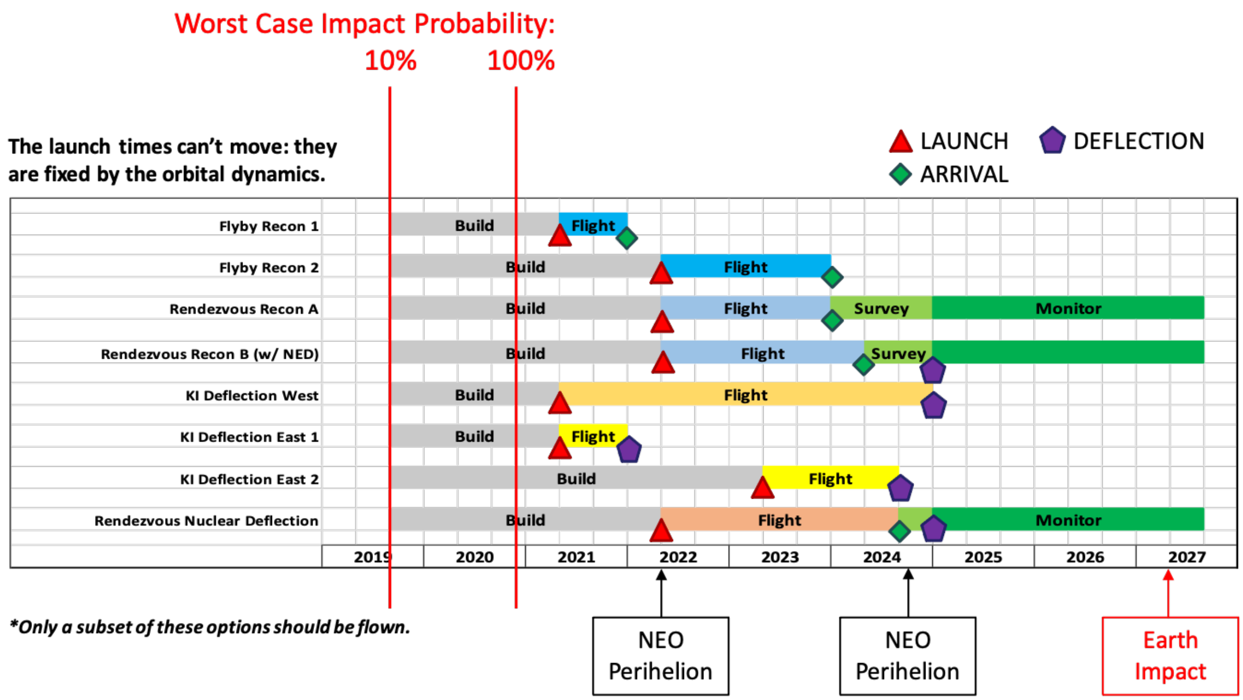

Figure C-4. Example of rapid-response scheduling in parallel with impact statistics improvement in Figure C-2. All concepts are started after the first apparition, before impact probability reaches 100% (ref PDC19).

Not shown in this example, but present in some real-world cases is the possibility that the newly-found threat will narrowly miss the Earth in a b-plane "keyhole" which creates a difficult-to-predict orbit perturbation that can substantially bias the subsequent impact trajectory toward or away from the Earth [Chesley (2005) and Chodas & Yeomans (1999)]. While unlikely, this case is much more complicated than the scenario used here.

C.2.2 LONG WARNING TIME

If the solutions of the new threat trajectory indicate a possible impact in more than ~10–20 years, then a deliberate, serial process could be followed by launching a characterization mission first, followed by a mitigation mission (assuming the impact probability continued to grow). A fundamental aspect of the characterization mission would be to rendezvous and perform proximity operations in order to obtain a precise estimate of the object mass, as well as an improved orbit solution and estimate of the impact location, if that was not already available from ground-based observations. The estimate of the object mass is needed to assess how much deflection a given mitigation mission can impart, and the precise estimate of the impact location is needed to assess the required total deflection. Together these parameters would facilitate development of an optimum deflection strategy, including adequate time to perform the deflection well before impact, while such deflection is highly effective (Hernandez & Barbee, 2012).

C.2.3 EFFICACY OF DEFLECTION TECHNOLOGIES

Selection of appropriate deflection technologies for a given scenario is a delicate balance between generating adequate deflection, given the asteroid mass and available warning time, without causing unwanted disruption. Additionally, we impose a practical limitation of deflection capability with a single high-performance launch (ideally with at least one backup).

Kinetic Impact

Kinetic impact is probably the most obvious and certainly simplest means to deflect an asteroid, although it is limited by a number of factors, especially the available momentum in a single impact, and the risk of disrupting the single body into multiple pieces that would become even harder to deflect (Barbee et al., 2018). Previous studies have suggested the hypothetical situation of a large number of launches and deflections in a short period of time, but this is not feasible with today's infrastructure or any likely future scenario (Barbee et al., 2018) and can lead to clearly unrealistic scenarios (Woo & Gao, 2021). However multiple launches on a longer time scale (e.g., multiples of the asteroid orbit period) are certainly feasible.

Given the uncertainties with the hypervelocity impact, it would be highly desirable to precede it with a reconnaissance mission to assess the size and mass of the target. Preferably this would be a rendezvous mission in order to provide an accurate estimate of target mass, its precise trajectory and impact location, and to verify the presence or absence of satellites. Knowing the mass and the impact location more precisely is critical to designing a KI intercept (§4.2.5). If there is inadequate time for the rendezvous, a "fast flyby" recon (§4.2.2) would be the only alternative to provide at least approximate information of these key parameters. A rendezvous recon mission has the additional advantage that it could also remain on station as an observer of the deflection, and measure the achieved velocity change, confirming the deflection and post-verifying integrity.

A fundamental limitation of KI deflection is that its deflection direction is determined by the intercept geometry, and so generally non-optimal. Impulsive deflection techniques like KI also have the additional limitation that the ΔV required to move the asteroid trajectory away from impact may be larger than the threshold ΔV which causes disruption and fragmentation of the asteroid. KI deflection also suffers from a "handedness handicap" in which, as a consequence of orbit dynamics, deflection in one direction is significantly harder to accomplish than deflection in the other direction. If the predicted impact is at a location near the Earth limb, but deflection towards that limb is along the "hard" direction, a much larger KI deflection may have to be executed in the easier-but-longer direction. To be successful KI deflections may need to be larger than deflections with other methods.

Nuclear

Although there is not a great deal in the public literature, the possibility of using the radiation from a nuclear blast in close proximity to an asteroid has been known for at least thirty years (Ahrens & Harris, 1992). This technique has the advantage of allowing a much higher amount of momentum to be transferred to the asteroid than is feasible with a single KI (Miller & Dearborn, 2015). Due to the importance of precisely controlling the trigger altitude, it would be strongly preferable to deliver the device as part of a rendezvous mission as opposed to a hypervelocity intercept, which may appear excessively risky. In this manner, a nuclear deflection can push the asteroid in any direction desired.

Ion Beam Deflection

Although the available forces are small, IBD has been proposed as a natural consequence of using electric propulsion thrusters to provide a slow, controlled deflection (Brophy et al., 2018; Bombardelli et al., 2019). This has not been examined extensively by the PD community so we attempt to simulate its effectiveness below. IBD inherently requires rendezvous and extended proximity operations; this carries the benefits of detailed characterization and a very controllable deflection not available with the other techniques.

Comparison of Deflection Techniques

We combine modeling of KI and IBD, including disruption limitations similar to previous work by Miller & Dearborn (2015), only using numerical examples of actual asteroid trajectories to yield a stochastical data set of discrete realizations (Figure C-5). This indicates that KI and IBD have similar performance envelopes, except that the risk of asteroid disruption limits the impulse available to a single KI deflection. All cases assume a single Falcon Heavy launch and accurately compute the deflection capability at each time in the simulation. Less obvious in Figure C-5 is that depending on the assumption of disruption threshold, for many of the orbits considered, KI fails to provide any capability for deflection without disruption over the 35-year simulation period. For example, assuming a 10% disruption threshold, only half of the cases simulated provided any pure-deflection capability (of course a series of smaller KI deflection missions could be carried out to avoid the large ΔVs which might disrupt the asteroid, but this violates the single-mission assumption). As asteroid diameter grows in Figure C-5, the mass grows with the cube of the diameter so both IBD and KI become performance limited in the range of 100–300 m diameter targets with 10–30 year of warning time.

The simulations done to create Figure C-5 use accurate asteroid orbital models to compute actual deflections in the asteroid impact b-planes. For each asteroid orbit the simulation shows the maximum size asteroid (assuming a nominal density of 2000 kg/m^3) that could be deflected off of a collision trajectory, as a function of warning time. The modeling to date uses fifteen Earth-impacting orbits already loaded into the CNEOS NEO Deflection App (NDA) (JPL CNEOS, 2021b). For a more statistically-sound analysis, a larger set of one-hundred orbits are currently being simulated; however the current smaller set is useful for illustrative purposes.

Although the orbits of the asteroids and Earth are realistic in the simulations, the calendar dates are removed; all times are simply days before impact. Since each of these orbits impact the Earth, orbit position in the b-plane (the b-vector) lies within the “capture circle” of the Earth, which is larger than the Earth disc due to gravitational focusing. Working in the b-plane may seem to be an unnecessary complication, but it has the advantage that orbit displacements at a given deflection time are linearly related to the ΔV applied and not dependent on the starting position of the b-vector; if the ΔV is doubled, the Δb is also doubled regardless of where the asteroid was originally headed. The relationship between ΔV and Δb is very much dependent on when the ΔV is applied and in which direction. Generally, the earlier the ΔV is applied the greater the Δb, but there is also a strong dependence on where the asteroid is on its orbit when the ΔV is applied: the Δb produced by a given ΔV is largest when the asteroid is near perihelion (e.g., Figure C-3). The simulations assume that only a single deflection mission is launched, and the launches are consistent with the launch mass and C3 performance of a Falcon Heavy launch vehicle. Stacking multiple deflection missions was not considered, but could be implemented in some fashion for each method over appropriate time scales.

Kinetic Impactor Simulations

The KI intercept trajectories are realistic, ballistic trajectories. A wide range of possible launch dates and deflection dates were considered, up to 30 years before impact, producing thousands of potential mission trajectories for each orbit. KI missions were restricted to a times-of-flight less than 5.5 years, and arrival solar phase angle of less than 120 deg (180 deg would be arriving from the night side of the asteroid and would make terminal guidance extremely difficult). The deflection is conservatively assumed to be inelastic: the spacecraft mass becomes embedded in the asteroid and transfers all of its momentum to the asteroid, with no ejecta magnification (i.e., beta=1). The arrival direction of the spacecraft at intercept determines the direction of the ΔV, which is generally not the optimal direction for deflection. A minimum displacement of 2 capture radii is used as the metric for a successful deflection, because KI deflection is uni-directional, this represents the worst-case requirement for a successful deflection.

For an impulsive deflection method such as KI, there is a possibility that the asteroid may be disrupted, fragmented or even destroyed. Simulations suggest that disruption occurs when the imparted ΔV exceeds a threshold fraction of the escape velocity of the asteroid, conservatively ~ 10% (Miller & Dearborn, 2015). Other threshold levels from 0 to 100% were also considered to observe the sensitivity to this assumption. The maximum and minimum asteroid sizes that could be deflected without being disrupted was calculated, as a function of launch time, across all possible KI missions. Launch time was turned into a warning time by simply adding 2 years as a minimal mission development time.

When KI missions deflect asteroids decades before impact, the ΔV necessary to avoid impact is often quite small (~1 cm/s) and the maximum size of asteroid that can be deflected depends principally on the maximum mass that can be launched onto the intercept trajectory. When warning times are less than roughly 15 to 20 years, the ability to deflect the asteroid, without exceeding the disruption threshold, and using a single KI mission, is significantly constrained and may not even be possible for most impactor orbits.

IBD Simulations

A single high-power ion-beam deflection spacecraft was assumed to have rendezvoused with the asteroid after a three-year cruise, a typical time-of-flight for these trajectories. The launch time was turned into an estimate for warning time by adding an assumed 3 years of development time (six years total before deflection can begin).

The ion thrusters are assumed to be the 20 kW NEXIS thrusters developed to TRL 4 in support of the Jupiter Icy Moons Orbiter (JIMO) mission concept (Brophy et al., 2018). This thruster has a high specific impulse and a small beam divergence angle (< 4 deg), which happens to be ideal for this application. (Other ion thrusters could also be used, and ion grids can be added to focus the beams.) The thrusters would be operated in pairs pointing in opposite directions and with identical thrust levels. The spacecraft would be stationed at a reasonable standoff distance (~2–4 times the asteroid diameter), close enough that 95% of the ion beam of one of the thrusters can be assumed to intercept the asteroid.

IBD simulations were performed assuming both 60 kW of power at 1 AU with 1 pair of NEXIS thrusters, and 100 kW at 1 AU, equipped with 2 pairs of NEXIS thrusters. (Note that a 60 kW array is presently being developed for NASA's Lunar Gateway, and 25 kW SA modules have recently been installed on the International Space Station).

1500 kg of xenon propellant is assumed to be available at the start of deflection operations. The total propellant load at launch would be greater, but a good fraction would be used to rendezvous with the asteroid (in all cases the launch wet mass is well within the capability of the Falcon Heavy LV). Thrust power is throttled according to available power throughout the asteroid orbit, with a minimum cut-off of 20 kW for the thruster pair. The deflection thrust impulse and ΔV on the asteroid was computed for each day, mapped to the displacement in the b-plane produced for that day, and displacements then summed to obtain the total b-plane deflection. The direction of thrusting was optimized each day so as to produce the largest total deflection. The days chosen for thrusting were also optimized to maximize deflection, generally centered around each perihelion time, when maximum power is available and the deflection ΔV has maximum effect. IBD mission lifetimes were capped at 15 years.

The criterion for a successful deflection is a displacement of 1 capture radius in the b-plane, half of that used for the KI case because IBD deflection can choose to take the shortest path to displace the trajectory off the Earth, which is 1 capture radius, worst case (IBD is not uni-directional as is KI). Since b-plane displacement is proportional to the mass of the asteroid, the maximum mass, and therefore size, of asteroid that can be deflected a threshold distance of 1 capture radius, can be computed and plotted as a function of deflection start time, creating the underlying envelopes illustrated in Figure C-5. The maximum size steps upwards as each perihelion is included in the thrusting period. This is true even when propellant and lifetime limits are imposed, because deflection is generally more productive for increasingly early perihelion periods. Since IBD is slow and continuous for long periods, it is not limited by disruption risk as are impulsive methods.

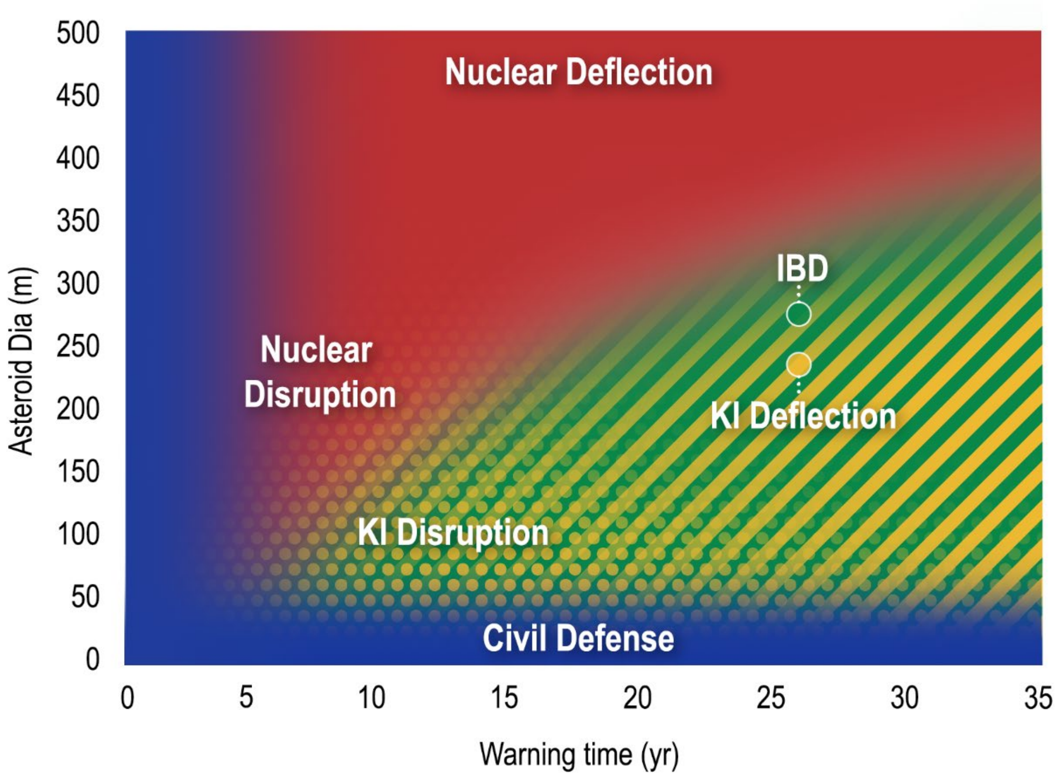

Figure C-5. Numerical results from simulating deflection capabilities of various deflection techniques across a variety of asteroid sizes following fifteen different Earth-impacting orbits. KI techniques (yellow) largely overlap the region where IBD is effective (green). Assuming a 10% disruption threshold, only half of the KI scenarios offered any pure-deflection capability within 35 years of impact (area of solid yellow stripes). If the warning time is very short and/or the asteroid relatively small, deliberate KI disruption may be the only viable non-nuclear technique (yellow dots).

C.2.4 *MISSION RELIABILITY*

In operational defense of planet Earth, a very high probability of success would be expected compared with extant PD demonstration/science missions (e.g., DART, Deep Impact). The traditional conceptions of reliability are well-understood and readily applied subject to project budget resources: component reliability in presence of random part failures, redundancy (to mitigate random failures), and robustness (performance margins). Another very important dimension of robustness is design diversity, for avoidance of common-mode failures and/or common design errors. Ideally, critical functionality can be accomplished using at least two components (or entire spacecraft) designed, built, and tested independently of each other ("design diversity").

A more challenging issue in unknown domains like PD is *the validity of design assumptions in presence of an unknown target/environment*. This leads to desire for "characterization" missions prior to "mitigation" (deflection) for long warning time scenarios. For short warning times this source of risk becomes even more significant because of lack of understanding the target characteristics (e.g., mass, number of bodies, etc.).

With few exceptions, operational reliability of critical mitigation missions has not been thoroughly addressed in the PD literature. Reliability in the sense of random failures has been considered in (Barbee et al., 2018), and reliability of the physics of various mitigation methods has been considered in many studies (Bruck Syal et al., 2013; Dearborn et al., 2020).

However there does not appear to have been substantial consideration of the overall confidence in success of a truly critical deflection mission, in the same sense that a military operation would be reviewed, or a Class-A NASA flight project. In particular, designing a mission that has a very specific success criteria (adequate deflection) in the presence of "unknown unknowns" of an unfamiliar target and engagement scenario deserves additional scrutiny. In the case of a hypervelocity KI mitigation, this review may be expected to focus on reliability of the time-critical sequence of events leading up to the impact (Barltrop & Kan, 2005). This class of system design is notoriously complex, and nearly impossible to V&V adequately (Frazier et al., 2019). Although it can be done successfully (Frauenholz et al., 2008) these critical sequences are only done out of necessity and are not inherently a low-risk architecture. KI is well-accepted as a science or demonstration mission (Holsapple & Housen, 2007), but may appear risky in an actual PD operational context, compared with a deflection that is delivered via a slow rendezvous.

KI mission risk would be compounded if multiple launches and impacts are required, target physical properties (integrity) and surrounding environment (dust and debris) could change and its changed trajectory would become increasingly unpredictable with each successive impact. For this reason, we limited deflection modeling to what is feasible with a single high-performance LV (Falcon Heavy Expendable). Time permitting, a series of smaller KI deflections on successive asteroid orbits might be practicable, but such a strategy would also require that an observer rendezvous spacecraft be on-station to observe each successive KI impact and assess the imparted ΔV, the changed orbit and new impact location, because those critical parameters could not be determined from the ground.

For the long warning time cases, deflection via a rendezvous and "slow-push" method is more robust and tolerant of faults and problems, because in that case the deflection occurs over much longer time scales and allows substantial time for fault diagnosis and recovery that otherwise is impossible with a KI. Slow-push deflection is also robust to almost any unexpected target characteristics (e.g., rubble pile) than KI, which may yield unexpected/undesirable results. Further, KI deflection allows little choice of the intercept geometry, and is also constrained by lighting conditions, so may or may not be capable of producing deflection in the desired direction. In comparison, slow-push deflection can be applied in any direction so is also more robust in that sense. Finally, KI deflection would almost certainly require a separate rendezvous observer mission to assess the imparted ΔV and verify the success of the deflection (or disruption); for a verified successful deflection, the KI technique will most likely require at least two launches, one into a rendezvous trajectory and another into a hyper-velocity encounter trajectory. An IBD mission, on the other hand, could serve as its own observer spacecraft, assessing the imparted ΔVs at regular intervals, and in theory accomplishing a verified successful deflection with just a single launch.

If delivered via rendezvous, nuclear deflection also offers a high confidence of mission success and as with the slow push, may be applied in any direction and is relatively robust to target characteristics (Bruck Syal et al., 2013) and offers the option of relatively safe disruption if necessary (Barbee et al., 2018). For the most-stressing short warning time cases and larger targets, a NED is the only viable option (top portion of Figure C-5). In this case, delivery via rendezvous would be strongly preferred for reliability reasons if there is adequate time available.

For these reasons, long-term development/demonstration of slow-push IBD (and nuclear deflection) technologies is an important path towards an optimum mitigation strategy that ultimately would provide the kind of confidence appropriate for real-life threats to the Earth given adequate warning time. Since these strategies inherently require rendezvous, a high ΔV capability most likely using Solar Electric Propulsion (SEP) is also implied as used by the Dawn mission to rendezvous with both Vesta and Ceres (Rayman et al., 2007). This same type of propulsion system can also be used for the deflection (Brophy et al., 2018) and is considered in the last family of cases presented in §4 and our recommendations in §7.

D OBSERVABILITY OF DEFLECTION DEMONSTRATIONS

Mitigation demonstrations are only valuable if their effectiveness can be measured. This appendix provides analyses of the ΔV required from IBD and KI deflection experiments, given different scenarios. This information is useful for planning and costing.

D.1 DEFLECTION SIMULATIONS

To provide information useful to calibrate modeling, it is important that any deflection demonstrations be designed such that the effects are measurable. This depends on the accuracy of the ephemeris before and after the deflection attempt. Since none of smaller NEOs that are of interest to this study have orbits determined to the accuracy needed, we have studied the effects of instead tracking a spacecraft flying in close proximity for a long period of time before and after the deflection experiment. This was done by picking a representative target and extrapolating from the actual orbit determination data of Bennu using the OSIRIS-REx spacecraft. The sample asteroid orbit is based on 2021 CG3, a fairly typical asteroid discovered in February and tracked for 7 weeks.

Asteroid tracking is modeled as derived from spacecraft tracking, with a new pseudo-delay observation every 15 days, starting 15 days after rendezvous. This is similar to what was done for Bennu during the OSIRIS-REx orbit phases. The difference with Bennu is in the delay uncertainty. With OSIRIS-REx, there was a spacecraft in orbit around a larger NEA and this allowed delay uncertainties of 15 ns. For purposes of this study addressing smaller bodies, we assume that the object is too small to orbit so that station keeping will be used, and that the spacecraft will have relatively large solar arrays. These factors will inject a significant amount of noise compared to what we had with OSIRIS-REx, and so we assumed 150 ns uncertainty for the tracking.

We included the Yarkovsky effect, assuming a 140-meter asteroid and estimating a scale factor on the acceleration. The effect is not solidly detected during the rendezvous period, but the fact that it is acting on the asteroid serves to inflate uncertainties in an appropriate way. This is quite conservative, since once the spin state and shape are known (post-rendezvous) the Yarkovsky effect can be modeled better than it can be estimated.

We have used this basic model to estimate the observability of Ion Beam Deflection (IBD) and Kinetic Impact (KI).

D.2 ION BEAM DEFLECTION OBSERVABILITY

To estimate the observability of a slow-push deflection, IBD thrusting is started 180 days after rendezvous and continued for 31 days. The IBD acceleration was all in the orbital transverse direction. From the tracking data, we estimated the effect of the thrusting, which leads to an uncertainty in the measured acceleration. When this uncertainty falls (well) below the level imparted by the spacecraft then one has detected the deflection). The gray region marks the thrusting interval. Bulk density is assumed to be 1500 kg/m^3. We assumed a commonly-used thruster would provide a force of 140 mN acting on the asteroid by the IBD thrusting (Snyder et al., 2020).

In the acceleration plot (Figure D-1) one can see that the acceleration is just barely detected for the 200-meter asteroid a year after the rendezvous. For the 100-meter asteroid the detection is made about two weeks after thrusting ends. For the 50-meter asteroid the detection is made even before thrusting. This is an artifact of the 15-day pseudo-delay cadence, and it means that the detection is doable with <<15 days of thrusting, probably only a few days. Put another way, one might say that the deflection can be detected almost in real time for the 50-meter asteroid. And the thrusting could start after only 2–3 months, rather than 6 months. However, there may be operational challenges with such a small target, particularly the hovering altitude to get most of the beam to hit a 50-meter target, and the potential station keeping challenges that may pose. These challenges are expected to be reduced by using a gridded ion-beam development to deliberately focus the beam, currently in development at JPL.

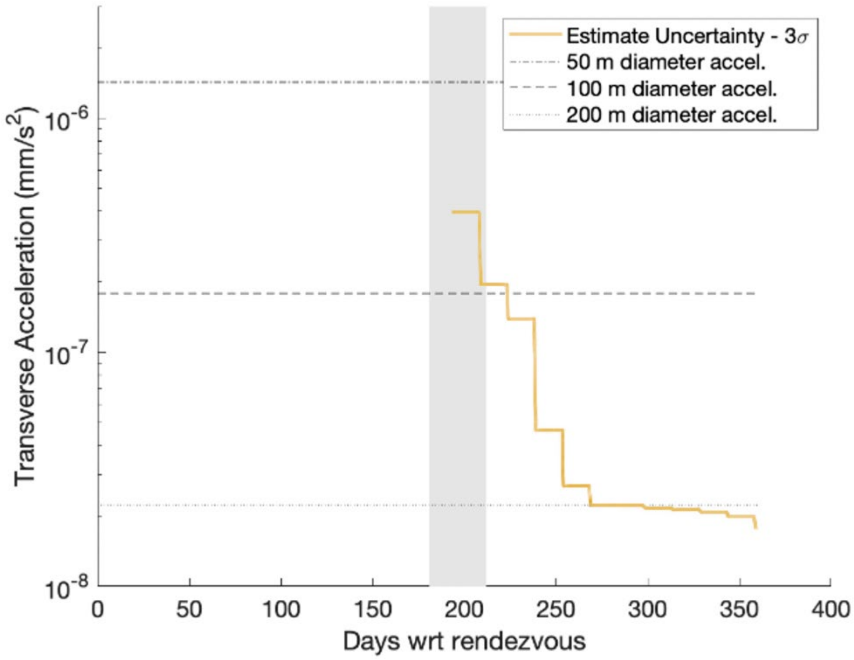

Figure D-6. Observability of small asteroid acceleration using IBD for a month

D.3 OBSERVABILITY WITHOUT RENDEZVOUS

If a KI or nuclear deflection demonstration is done, then the spacecraft that delivered the deflection will vanish and another is required to provide the tracking-via-proximity function assumed above (Options 13 and 14 in §4). However, given adequate circumstances, it is possible to observe the deflection with existing ground-based tracking (e.g., DART). A previous analysis suggested that KI demonstrations creating a deflection ΔV of ~1 cm/s should be observable via subsequent changes in the target heliocentric radius (Hernandez & Barbee, 2012). Here we use a different method to revisit this conclusion and to estimate deflection observability from the Earth. We modeled five random NEOs (but all ~200 m diameter), three with 10 year observed arcs and 2 with 5 year arcs. Three of these five have radar observations. The velocity uncertainty is plotted as a function of time after the last observation). With reasonably well observed arcs (5–10 years), the velocity uncertainty (typically) does not grow rapidly with time, and the uncertainty is generally on the order of a few or several cm/s. These example asteroids are all around 200 m in diameter, so are moderately observable. Larger asteroids are easier to get long arcs and smaller asteroids are harder, but a 10-year tracking arc should offer good observations at some times. The conclusion is that a deflection of at least 1 cm/s can probably be detected from the ground within a decade if the target is reasonably observable (so this would be a target selection criteria). A deflection of 10 cm/s would be readily visible given subsequent apparitions of the target asteroid.

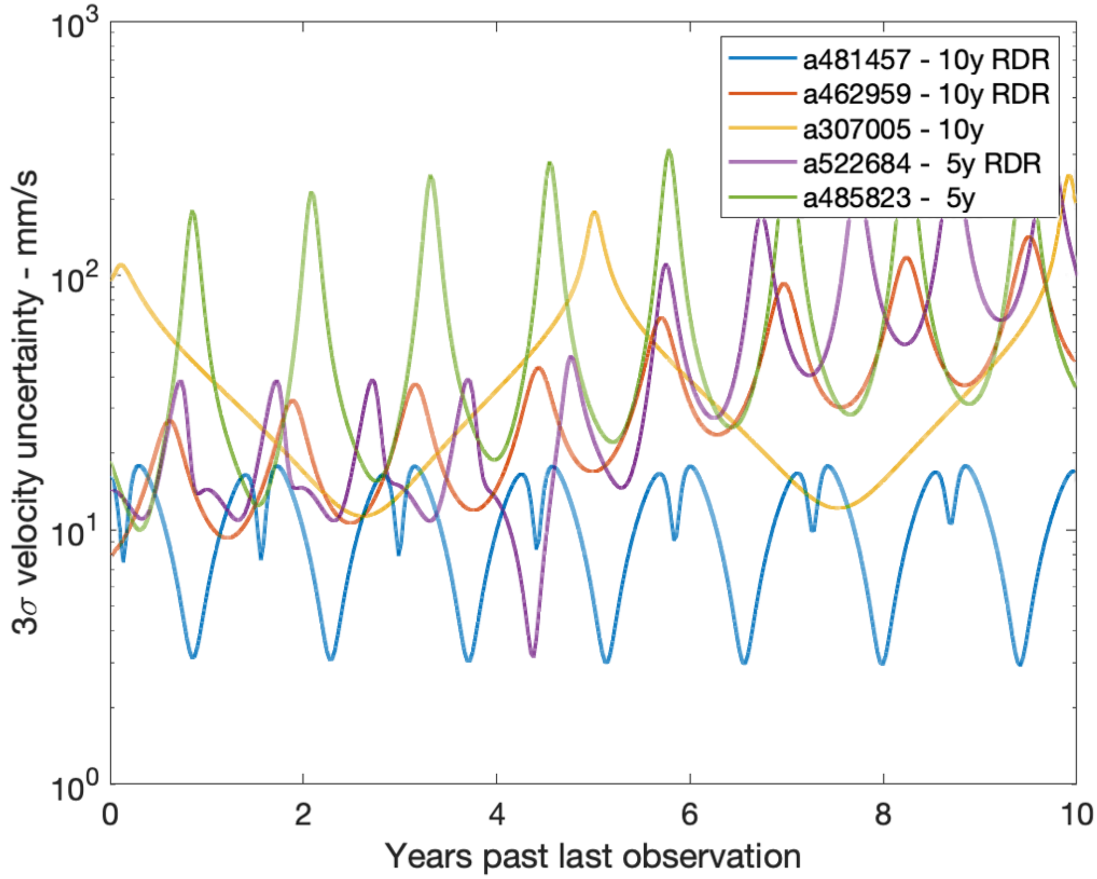

Figure D-7. Deflection observability from Earth given changes in target velocity

Note that the NASA/JPL Center for Near Earth Objects website provides an app for public users to perform their own simulations similar to those presented in JPL CNEOS (2021b).

E ARCHITECTURE TRADE STUDY

Included on the following pages is the entire presentation of JPL's Team-X, entitled *4X Planetary Decadal – Planetary Defense RMA 2021-04*, and presented on April 13, 15, 20, and 21, of 2021.

4X Planetary Decadal - Planetary Defense RMA 2021-04

Customers: **Steven E Matousek, Paul Abell, William E Frazier**

Facilitator: **Troy Hudson**

Session Dates: 13-Apr-2021 to 15-Apr-2021 and 20-Apr-2021 to 21-Apr-2021

Study ID: 387

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Jet Propulsion Laboratory
California Institute of Technology

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Team X Participants

- Troy Hudson (Facilitator)
- Mason Takidin (Cost)
- Robert Kinsey (Deputy Systems)
- Melora Larson (Instruments)
- Charles Reynerson (Mission Design)
- Ronald Hall (Power)
- Matthew Kowalkowski (Propulsion)
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Table of Contents

Executive Summary

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Executive Summary

Study Overview

- In support of the Planetary Science Decadal Survey, Team-X performed a Rapid Mission Architecture study numerous flight-element-level concepts for Near Earth Asteroid characterization and mitigation in the context of a Planetary Defense Demonstration Mission.
  - This study examined a total of 17 architectures, many with sub-options, for a total of 30 options. Additionally, all options were re-evaluated assuming launch on a low-cost small launch vehicle with a kick stage, for a total of 60 evaluated cases.
- The goal was to identify those concepts which would be technically feasible and fit within a Discovery mission class cost cap, such that those architectures could be recommended to the Decadal panel.
  - The total budget is $500M, inclusive of Phases A-D and the cost of a launch vehicle.
- The concepts investigated broadly broke into two categories:
  - Characterization: determining which asteroids are dangerous by assessing their mass, density, and structural integrity / composition
  - Mitigation: testing various methods asteroid redirection or disruption including, for example, gravity tractors and Nuclear Explosive Devices (NED; simulated)

Executive Summary

Mission Architecture and Assumptions

- Mission Architecture
  - 17 different architectures were studied; they shared the following common features:
    - 3 year design life
      - See Cost Report for assumed schedule
    - Risk class C; single-string
    - Reserve posture: 30% for Phases A-D, 15% for Phases E-F
    - Launch date: 2032-10-10
    - Cost cap of $500M for All Phases A-D, plus Launch Vehicle
      - Assumed 85% Phase A-D Costs; 15% Phase E-F Costs
    - Cost are reported in FY2025 dollars.

- Assumptions
  - Instrument resource requirements were given by analogy
  - MOS/GDS costs were estimated with rules of thumb
  - Radio science was assumed standard for all options

Executive Summary

Overview of Options

- This study examined a total of 17 architectures, many with sub-options, for a total of 30 options. Additionally, all options were re-evaluated assuming launch on a low-cost small launch vehicle with a kick stage, for a total of 60 evaluated cases.
- The following slides summarize the primary customer-supplied options (16), broken into the Characterization options (1-9, excepting 7 which was not studied) and the Mitigation options (10-16, notice two versions of option ‘13’). For full details see the Systems section of this report.

Executive Summary

Options Overview

- A total of 16 architecture options were specified by the customer. The customer-supplied options table is shown here and on the next slide.
- The table at right shows options examined in Week 1, which focused on Characterization missions (see next slide for Week 2)
- Option 7 was not examined, because at the resolution of this study it was indistinguishable from Option 1
  - It has the same instrumentation, ΔV, and data rate requirements as Option 1
  - There are programmatic distinctions (potentially longer Phase E, different navigation requirements), but these are not modeled in this study

Trade space point	Mission Purpose	Payload (s) (see columns to right)	Mission Design (s/c Δv)	Comments/ notes
First Week (Characterization)				
1	PHO/NEO Flyby Reconnaissance – response to particular threat	Vis NAC, Vis/NIR spec, Radio Science	0.25 km/sec	Get as much as you can in a single flyby? "Fast" development/deployment mission; 5 kbps w/ 'standard' antenna
2	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec	2 km/sec	See Papais Fig 13. This DV captures adequate fraction of population
3	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec; limited to SmallSats (cheaper/faster)	4 km/sec	SmallSat works on ESPA Grande. This DV captures large fraction of population.
4	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec, mono-static radar, lidar	2 km/sec	[make instruments separable due to cost concern]
5	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec	4 km/sec	This DV captures large fraction of population (Non-SmallSat Components)
6	PHO/NEO Rendezvous (two elements)	Vis WAC, Vis/NIR spec, Bi-static GPR, and LIDAR plus **Deployable assets** for surface operations hopper (if possible) to enhance geophysical characterization of targets	2km/s	instrument costing needs to be separable
7 (Not examined)	PHO/NEO Tour	Vis NAC, Radio Science, NIR spec	0.25 km/sec	This DV facilitates >100 different tours per Karimi analysis 4/8/21
8	PHO/NEO Tour (multiple uSats can provide perspective)	(Same as above but instruments "disaggregated" onto usats <100 kg) (But still need NAC on all)	0.25 km/sec	This DV facilitates >100 different tours per Karimi analysis 4/8/22 [1 'A' w/ both inst.; 1-5 'B' w/ NAC only]
9	PHO/NEO Tour (mother ship & cubesats)	Deployable cubesats for perspective to enhance characterization of targets via NACs	0.25 km/sec	1 cubesat per flyby provides perspective and mothership does DTE comm. Cubesats could impact too. ['Mother' w/ NAC + Vis/NIR Spec; 4 'daughter' cubesats w/ JCam]

Executive Summary

Options Overview

- The table at right shows options examined in Week 2, which focused on Mitigation.
- There was an additional Option 17 added by the System Engineer (SE) post-study, as well as many sub-options, for a total of 30 options exploring the dimensions of the option space
- See the Systems section of the report for more details on payload, data assumptions, and spacecraft specifications for the options and their sub-options

Trade space point	Mission Purpose	Payload (s) (see columns to right)	Mission Design (s/c Δv)	Comments/ notes
Second Week (Mitigation)				
10	Intercept PLUS Mitigation for short-warning scenario	NAC, WAC, plus NED. Radar plus ability to trigger @ high closing vel.	1 km/sec	Operational device would be ~ 200 kg; demo device is smaller.
11	Rendezvous w/Nuclear Explosive Device (NED) Simulator	NED, Vis WAC, and NIR Spec; range radar for trigger	2 km/sec	See above rendezvous cases (Option 2)
12	Rendezvous w/Nuclear Explosive Device (NED) Simulator and observer (Two elements)	Vis WAC and IR Spec ; combined with NED. two element system with observer to verify deflection result. Also mono-static radar	2 km/sec	keep characterization instruments costed separately
13-DI	Two-element Kinetic Impact with flyby (like DI). Single launch.	Vis WAC for Impactor OpNav, NAC for crater eval on Flyby s/c	0.5 km/sec	Impact >10 km/s
13-DART	Two-element Kinetic Impact with flyby (like DART). Single launch.	Vis WAC for Impactor OpNav (big S/C), JCam for crater eval on Flyby cubesat	0.5 km/sec	Impact >10 km/s
14	Two-element Kinetic Impact with rendezvous observer to eval crater. Single launch.	Vis WAC for impactor OpNav, NAC for observer crater eval	1 km/s (Impactor to intercept) 6 km/sec (Observer to rendezvous) Flight time can be long (few years)	Impact >10 km/s; observer does rendezvous first. MD requires longer cruise time and Earth GA's.
15	Characterization and Mitigation Rendezvous using Ion beam (SEP)	Vis camera WAC. Includes ability to automatically hover @ 750 m while thrusting against surface.	2 km/sec (to rendezvous only) **TBD** kg Xe for ion beam deflection	See Brophy paper. Assume 10 kW SA and 5 kW for SPT-140 thrusters like Psyche.
16	Gravity Tractor	advanced autonomous guidance and navigation, imagers, spectrometers, radar, radio science	2 km/sec (to rendezvous only) **TBD** kg Xe for GT deflection	

Executive Summary

Instruments Summary

Instruments list provided by the customer team:

	Mass kg	Power W	Most likely Mass	Most likely max Power	Cost $M (NICM 50%)	Options	Notes
Vis NAC	10	17	12.3	17	8.4	1, 7-10, 13, 14	Ave of BASIX, Condor from MSSS
Vis/NIR Spec	6	12	7.38	12	5.5	1-9, 11, 12, 16	MLPS + optics
Vis WAC	4	10	4.92	10	4.2	2-6, 10-16	Ave of BASIX and Trident
Radar 1 (HFR)	6	137	7.38	137	47	4	HERA heavy
Radar 2 (LFR)	4	50	4.92	50	24.7	6, 10, 12, 16	HERA bistatic (light)
LIDAR	13	31	15.99	31	12.6	4, 6	LOLA
Cubesat cam	3.2	14	3.9	14	4.5	9, 13 DART	Jcam

Most likely max values of mass and power used for NICM. Mass includes DP9 margins. 36 month lifetime. Note that NICM database does not include low mass RADAR analogues (European builds)

Executive Summary

Technical Findings

- Of the various options studied, **<u>most</u>** fit within the desired cost cap, regardless of the launch vehicle assumed
- The options that consistently did **<u>NOT</u>** fit within the cost target are the following:

  <u>Characterization</u>

  - **Option 4:** Rendezvous with a large instrument compliment of WAC, Vis/NIR spectrometer, mono-static radar, and lidar.
  - **Option 6:** Rendezvous with a large instrument complement Vis WAC, Vis/NIR spec, Bi-static GPR, and LIDAR plus **deployable assets** for surface operations to enhance geophysical characterization of targets.

  <u>Mitigation</u>

  - **Option 12:** Rendezvous w/ NED Simulator. Only the BiProp version of this option overran the cap.
  - **Option 14:** Two-element Kinetic Impact w/ SEP rendezvous observer to evaluate result.
  - **Option 15 (4 of 6 sub-options):** Rendezvous for characterization & mitigation using SEP-powered Ion Beam deflection

Executive Summary

Conclusions, Risks, and Recommendations

- All of the concepts studied here appear technically feasible (though there are questions about the actual deflection capabilities of the Ion Beam and Gravity Tractor demonstrations)
- Some are over the cost cap, but there are many that appear achievable within $500M.
  - See full breakdown in Cost Report section.
- It is worth noting that there is a high degree of uncertainty in these cost estimates. The bus costs are estimated almost entirely based on dry mass, and Phase E costs are not modeled explicitly. It may not be achievable within the budget of this study, but one or two Team X studies for selected point designs would help increase confidence in (or allow recalibration of) the cost estimates.

Systems Report

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Systems

Study Overview

- The goal of this study was to estimate the technical and cost feasibility of a wide range of concepts for a *Planetary Defense Demonstration* mission
- The study was conducted over two weeks and five half-day sessions, with significant additional out-of-session work. The first week focused on *characterization* missions, while the second week focused on *mitigation* missions.
- This study examined a total of 17 architectures, many with sub-options, for a total of 30 options. Additionally, all options were re-evaluated assuming launch on a low-cost small launch vehicle with a kick stage, for a total of 60 evaluated cases.
- This is an **Architecture Study**, and did not do a complete subsystem-level design. Several subsystem point designs for Power and Propulsion were used to calibrate estimates based on low-fidelity sizing models, to arrive at system-level low-fidelity estimates of total mass and cost.

- Customer Inputs
  - Definition of the Architecture Options (modified in-session)
  - Payload suite
  - Mission design information for some relevant trajectories
  - Launch vehicle assumptions for low-cost launch options
- Team X Outputs
  - Architecture-level estimates (total mass and project cost) for each option

Systems

Architecture Assessment Approach

- In a Team X Architecture Study, feasibility and cost are assessed at the mission and spacecraft level
  - There typically is no “subsystem-level” design
  - In this study, system mass and feasibility was assessed using a low-fidelity integrated model called the Tool for Architectural Tradespace Exploration and Refinement (TATER).
  - The Team X Power and Propulsion chairs did subsystem-level design work to use as “calibration points”, that were fed into the TATER model. See “Design Assumptions” slide for more details.

- Payload accommodation requirements for mass, power, volume, pointing, and data rates were used to drive the TATER sizing model

- Total mass is compared against launch capability

→ **Technical Feasibility is assessed by whether mass margins are positive**

- Costs are estimated from a combination of direct estimates and rules of thumb
  - Payload costs (WBS 5) were estimated using the NASA Instrument Cost Model (NICM), a suite of tools used to estimate the development cost of future NASA spaceflight instruments. (Link: NASA Instrument Cost Model - NICM | NASA)
  - Bus costs for modeled spacecraft (WBS 6) were estimated using a regression model (see Cost report for more)
    - Bus costs for select flight elements were taken from historical data or subject matter expert assessment
  - Launch costs were taken from a mix of current NASA pricing and approximation from publicly available data
  - All other wrap costs were estimated using Rules of Thumb (scaled off of WBS 5 and 6) based on data from previous $500M-class missions
  - Reserves were added
- **→ Cost Feasibility is assessed by whether the total mission cost is under the target cost**

- **Note that this is considered a “low-fidelity” study, with significant uncertainty in the technical and cost assessments.**

Systems

Margin Policy

- Mass and Power Maximum Expected Values (MEV) for the payload were taken from the customer input materials
  - The AIAA and JPL Design Principles 9 term for this is “Predicted Mass”
- For models that required Current Best Estimate (CBE) values, a 20% contingency value was assumed, and CBE was back-calculated as MEV / (1 + 0.2)
  - This also corresponds to the assumed 20% Mass Growth Allowance (MGA) for early-phase concepts in the up-coming JPL Design Principles v9 (DP9)
- To ensure compliance with JPL’s Design Principles (v8, “DP8”), we asserted a 30% JPL Dry Mass Margin on launch mass
  - JPL Dry Mass Margin = (Dry Capability – CBE Requirement)/(Dry Capability)
    - Dry Capability = Launch Allocation – Propellant Mass
  - 30% JPL Dry Mass Margin (DP v8) corresponds to a 43% increase over the CBE dry mass
    - Margined Dry Mass = 1.43 * CBE Dry Mass
  - This is equivalent to the JPL Design Principles 9 value of 23% (as fraction of CBE or “basic mass”) margin over Predicted Mass (or over the customer-supplied MEV value)
  - Margined dry mass (payload + spacecraft) was compared against Launch Vehicle capability
- Also for compliance with JPL’s Design Principles (both v8 and v9), payload power values were also assigned 30% JPL Margin, and compared against bus capabilities

Systems

Guidelines

- Mission Class: C
  - Single-string spacecraft is acceptable
  - The customer indicated that we should assume the use of *small spacecraft* (“SmallSat”) components wherever possible, to save mass and cost, and that this was consistent with their intended risk posture (one exception was made in Option 5, see later discussion).
- Cost Cap: $500M, including Launch Services and Phase E/F
- Fiscal Year: 2025
- Includes 30%/15% reserves on development/operations; and assumes an 85%/15% development/operations cost split.

Systems

Options Overview

- A total of 16 architecture options were specified by the customer. The customer-supplied options table is shown here and on the next slide.
- The table at right shows options examined in Week 1, which focused on Characterization missions (see next slide for Week 2)
- Option 7 was not modeled, because at the resolution of this study it was indistinguishable from Option 1
  - It has the same instrumentation, ΔV, and data rate requirements as Option 1
  - There are programmatic distinctions (potentially longer Phase E, different navigation requirements), but these are not modeled in this study

Trade space point	Mission Purpose	Payload (s) (see columns to right)	Mission Design (s/c Δv)	Comments/ notes
First Week (Characterization)				
1	PHO/NEO Flyby Reconnaissance – response to particular threat	Vis NAC, Vis/NIR spec, Radio Science	0.25 km/sec	Get as much as you can in a single flyby? "Fast" development/deployment mission; 5 kbps w/ 'standard' antenna
2	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec	2 km/sec	See Papais Fig 13. This DV captures adequate fraction of population
3	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec; limited to SmallSats (cheaper/faster)	4 km/sec	SmallSat works on ESPA Grande. This DV captures large fraction of population.
4	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec, mono-static radar, lidar	2 km/sec	[make instruments separable due to cost concern]
5	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec	4 km/sec	This DV captures large fraction of population (Non-SmallSat Components)
6	PHO/NEO Rendezvous (two elements)	Vis WAC, Vis/NIR spec, Bi-static GPR, and LIDAR plus **Deployable assets** for surface operations hopper (if possible) to enhance geophysical characterization of targets	2km/s	instrument costing needs to be separable
7 (Not examined)	PHO/NEO Tour	Vis NAC, Radio Science, NIR spec	0.25 km/sec	This DV facilitates >100 different tours per Karimi analysis 4/8/21
8	PHO/NEO Tour (multiple uSats can provide perspective)	(Same as above but instruments "disaggregated" onto usats <100 kg) (But still need NAC on all)	0.25 km/sec	This DV facilitates >100 different tours per Karimi analysis 4/8/22 [1 'A' w/ both inst.; 1-5 'B' w/ NAC only]
9	PHO/NEO Tour (mother ship & cubesats)	Deployable cubesats for perspective to enhance characterization of targets via NACs	0.25 km/sec	1 cubesat per flyby provides perspective and mothership does DTE comm. Cubesats could impact too. ['Mother' w/ NAC + Vis/NIR Spec; 4 'daughter' cubesats w/ JCam]

Systems

Options Overview

- The table at right shows options examined in Week 2, which focused on Mitigation.
- There was an additional Option 17 added by the System Engineer (SE) post-study (documented in this Systems report but which does not show up in other subsystem reports), as well as many sub-options, for a total of 30 options exploring the dimensions of the option space
- See the following slides for more details on payload, data assumptions, and spacecraft specifications for the options and their sub-options

Trade space point	Mission Purpose	Payload (s) (see columns to right)	Mission Design (s/c Δv)	Comments/ notes
Second Week (Mitigation)				
10	Intercept PLUS Mitigation for short-warning scenario	NAC, WAC, plus NED. Radar plus ability to trigger @ high closing vel.	1 km/sec	Operational device would be ~ 200 kg; demo device is smaller.
11	Rendezvous w/Nuclear Explosive Device (NED) Simulator	NED, Vis WAC, and NIR Spec; range radar for trigger	2 km/sec	See above rendezvous cases (Option 2)
12	Rendezvous w/Nuclear Explosive Device (NED) Simulator and observer (Two elements)	Vis WAC and IR Spec ; combined with NED. two element system with observer to verify deflection result. Also mono-static radar	2 km/sec	keep characterization instruments costed separately
13-DI	Two-element Kinetic Impact with flyby (like DI). Single launch.	Vis WAC for Impactor OpNav, NAC for crater eval on Flyby s/c	0.5 km/sec	Impact >10 km/s
13-DART	Two-element Kinetic Impact with flyby (like DART). Single launch.	Vis WAC for Impactor OpNav (big S/C), JCam for crater eval on Flyby cubesat	0.5 km/sec	Impact >10 km/s
14	Two-element Kinetic Impact with rendezvous observer to eval crater. Single launch.	Vis WAC for impactor OpNav, NAC for observer crater eval	1 km/s (Impactor to intercept) 6 km/sec (Observer to rendezvous) Flight time can be long (few years)	Impact >10 km/s; observer does rendezvous first. MD requires longer cruise time and Earth GA's.
15	Characterization and Mitigation Rendezvous using Ion beam (SEP)	Vis camera WAC. Includes ability to automatically hover @ 750 m while thrusting against surface.	2 km/sec (to rendezvous only) **TBD** kg Xe for ion beam deflection	See Brophy paper. Assume 10 kW SA and 5 kW for SPT-140 thrusters like Psyche.
16	Gravity Tractor	advanced autonomous guidance and navigation, imagers, spectrometers, radar, radio science	2 km/sec (to rendezvous only) **TBD** kg Xe for GT deflection	

Systems

Options Overview – Specification, Week 1 (Characterization)

- The table below shows the full list of architecture options that were examined in the study, week 1 (characterization)
- This table shows all sub-options, flight elements, and payload elements, plus ΔV and data volume

Option #	Option Name	Flight Elements		ΔV	Instruments				Total Data Volume
					1	2	3	4	
		Qty	Name	m/s	Name	Name	Name	Name	Gbit
1	Flyby Recon	1	Flyby S/C	250	Vis NAC	Vis/NIR spec			30
2	Rendezvous	1	Rendezvous S/C	2000	Vis WAC	Vis/NIR spec			400
2-E	Rendez SEP	1	Rendezvous S/C	2000	Vis WAC	Vis/NIR spec			400
3	Rendez high ΔV	1	Rendezvous S/C	4000	Vis WAC	Vis/NIR spec			400
4	Rendez Radar-Lidar	1	Rendezvous S/C	2000	Vis WAC	Vis/NIR spec	Radar (HERA Heavy)	Lidar	400
4-E	Rendez SEP Radar-Lidar	1	Rendezvous S/C	2000	Vis WAC	Vis/NIR spec	Radar (HERA Heavy)	Lidar	400
5	Rendez high ΔV BigSat	1	Rendezvous S/C	4000	Vis WAC	Vis/NIR spec			400
6	Rendez 2FE	1	Rendezvous Mother S/C	2000	Vis WAC	Vis/NIR spec	Radar (HERA Light)	Lidar	400
		1	Deployed Lander	-	Radar (HERA Light)				
6-E	Rendez SEP 2FE	1	Rendezvous Mother S/C	2000	Vis WAC	Vis/NIR spec	Radar (HERA Light)	Lidar	400
		1	Deployed Lander	-	Radar (HERA Light)				
7 (like #1)	Tour Single	1	Flyby S/C	250	Vis NAC	Vis/NIR spec			30
8-1	Tour Multiple (1xB)	1	Flyby S/C (A)	250	Vis NAC	Vis/NIR spec			30/target
		1	Flyby S/C (B)	250	Vis NAC				30/target
		1	Dual Payload Adapter	-					
8-2	Tour Multiple (2xB)	1	Flyby S/C (A)	250	Vis NAC	Vis/NIR spec			30/target
		2	Flyby S/C (B)	250	Vis NAC				30/target
		1	Multi Payload Adapter	-					
8-3	Tour Multiple (3xB)	1	Flyby S/C (A)	250	Vis NAC	Vis/NIR spec			30/target
		3	Flyby S/C (B)	250	Vis NAC				30/target
		1	Multi Payload Adapter	-					
9	Tour CubeSats	1	Flyby Mother S/C	250	Vis NAC	Vis/NIR spec			60/target
		4	Flyby CubeSat	-	JCam				

Options Overview – Specification, Week 2 (Mitigation)

- The table below shows the architecture options that were examined in the study, week 2 (mitigation)
- Note that Option 17 was added post-study by the Systems Engineer, and was not in the original study request. Documented in Systems only.

Option #	Option Name	Flight Elements		ΔV	Instruments				Total Data Volume
					1	2	3	4	
		Qty	Name	m/s	Name	Name	Name	Name	Gbit
10-MP	Intercept	1	Intercept S/C	1000	Vis NAC	Vis WAC	Radar (HERA Light)	NED Simulator	30
10-BP	Intercept	1	Intercept S/C	1000	Vis NAC	Vis WAC	Radar (HERA Light)	NED Simulator	30
11	Rendezvous wNED	1	Rendezvous S/C	2000	Vis/NIR spec	Vis WAC		NED Simulator	400
11-E	Rendezvous SEP wNED	1	Rendezvous S/C	2000	Vis/NIR spec	Vis WAC		NED Simulator	400
12	Rendezvous wNED (2E)	1	Rendezvous S/C	2000	Vis/NIR spec	Vis WAC	Radar (HERA Light)	NED Simulator (Deployable)	400
12-E	Rendezvous SEP wNED (2E)	1	Rendezvous S/C	2000	Vis/NIR spec	Vis WAC	Radar (HERA Light)	NED Simulator (Deployable)	400
13-DI	Kinetic Impact (DI)	1	Flyby Observer S/C	500	Vis NAC				30
		1	Impactor S/C	50	Vis WAC				
13-DART	Kinetic Impact (DART)	1	Mothership Impactor	500	Vis WAC				30
		1	Flyby Observer SmallSat	50	JCam				
14	Kinetic Impact (SEP obs)	1	SEP Rendez. Observer S/C	6000	Vis NAC				400
		1	Impactor S/C	1000	Vis WAC				
15-A-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	2000+32kg	Vis WAC				400
15-B-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	2000+150kg	Vis WAC				400
15-C-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	2000+615kg	Vis WAC				400
15-A-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2000+32kg	Vis WAC				400
15-B-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2000+150kg	Vis WAC				400
15-C-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2000+600kg	Vis WAC				400
16	Gravity Tractor	1	SEP Deflector S/C	2000+50kg	Vis WAC	Vis/NIR spec	Radar (HERA Light)		400
17*	Ion Beam & Gravity Tractor	1	SEP Deflector S/C	2000+82kg	Vis WAC	Vis/NIR spec	Radar (HERA Light)		400

Systems

Design Assumptions – Spacecraft Modeling (1/3)

- The spacecraft buses were primarily modeled with the Tool for Architectural Tradespace Exploration and Refinement (TATER), which uses a combination of empirical subsystem models and simple physical models to estimate the mass and technical feasibility of a spacecraft architecture
  - See (slightly outdated) paper by Hogstrom et al, “From Cocktail Napkin to Concept Feasibility: Spacecraft Design in Early Formulation with TATER” (IEEE 2018)
- In cases with multiple Flight Elements, certain flight elements were generally specified as “carrying” others, and the carrying element’s propulsion and structures were sized accordingly

- **Instruments:** Instrument masses and average power consumption were taken from customer-supplied materials

- **Attitude Control Subsystem (ACS)**
  - SmallSat components were assumed for an Inertial Measurement Unit (IMU) and Star Tracker, with the exception of Option 5 which used “standard” avionics
  - In most designs, a single 2-channel gimbal drive electronics board was assumed for controlling solar arrays (gimbaled so that it could be on-sun while communicating via a fixed HGA)
  - All designs were assumed to use reaction wheels, which were sized according to a rule of thumb that scaled off of the spacecraft wet and dry mass. For some spacecraft (esp. flyby S/C), this may not be the optimal choice; however, that trade was not performed.

- **Command and Data Subsystem (CDS):** mass and power were estimated as a fixed value
  - Most options assumed a SmallSat CDS system (5kg CBE, 4W CBE). This is similar to JPL’s “Sabertooth” avionics package.
  - For the “Flyby SmallSat” in Option 13-DART, a Sphinx CubeSat CDS board was assumed (1kg CBE, 2W CBE), as the S/C mass was consistent with a 12U CubeSat
  - Option 5 used a fixed “standard” CDS package (8.5kg CBE, 19W CBE)

- **Mechanical**: The mechanical and structures mass was estimated using a “tax” on all other spacecraft mass, with a coefficient based on a regression from past spacecraft designs (specifically for orbiters, flyby spacecraft, etc., and excluding rovers, landers, and probes)
- **Cabling**: The cabling mass was likewise estimated with a “tax” on the avionics subsystems (Instruments, CDS, ACS, Telecom, Power)

Systems

Design Assumptions – Spacecraft Modeling (2/3)

- **Telecom:**
  - Most flyby options, with 30 Gb of total data volume required (per target, in the case of the Option 8 tour), assumed a 1m deployable high-gain antenna (HGA) plus an Iris radio, which would trickle back the 30 Gb of flyby data (plus an assumed 30% overhead) at 20kb/s at an Earth-S/C range of 1AU over 3 months (or 3 months per target for Option 8)
  - All rendezvous options (with the exception of Option 5), with 400 Gb of total data volume required, assumed that the telecom subsystem could be upgraded with a 25W Solid State Power Amplifier (SSPA) rather than using the Iris radio's default 4W SSPA. With the same 1m deployable HGA, it would achieve a data rate of 100kb/s at an Earth-S/C range of 1AU, enabling it to return the 400Gb of data (plus 30% overhead) over about 7 months.
  - For options which required spacecraft-to-spacecraft relay, the Mother Spacecraft was assumed to have an Electra-Lite UHF relay radio; the Daughter Spacecraft (if modeled explicitly) was assumed to have a CubeSat UHF radio. In Option 9, where the Daughters are also flyby S/C, the Mother Spacecraft's data return requirements are assumed to double to 60Gb per target. A 10W SSPA is assumed, to double the return rate to 40kb/s, and the data return time is kept the same at 3 months per target x 4 targets = 12 months.

- **Thermal:**
  - Thermal subsystem mass was estimated using a regression-based model, which is a simple tax on non-Thermal mass; in this case, the regression is for "Spacecraft"-type vehicles (that is, not Rovers, Landers, etc.).
  - Thermal subsystem power was assumed to be zero. This is an under-estimate, and it is therefore likely that future design iterations will find that the thermal power requirements drive up the power subsystem sizing, with ripples through the whole flight element.

Design Assumptions – Spacecraft Modeling (3/3)

- **Propulsion:**
  - Propulsion systems were sized to the specified ΔV requirement, plus 50m/s for attitude control and desaturations
  - In the case of SEP designs, it was generally assumed that the Cruise + Rendezvous portion of the mission (the primary SEP trajectory, prior to proximity operations) should take less than 1 year (with the exception of Option 14), with the EP engines assumed thrusting at a duty cycle of 85% (as in the customer-supplied tour trajectory). So the number of active thrusters was kept high enough such that the duty-cycled burn duration to reach the target ΔV was less than one year. If there were a trajectory with the same ΔV but which allowed for greater than 0.85yr of total thruster “on” time, it may in some cases be possible to reduce the number of thrusters or run at a lower power level (though the latter reduces I_{sp}).
  - In some cases, the propellant quantity caused the propellant throughput to exceed the total allowable for the number of active thrusters. In these cases, additional thrusters were added, with the assumption that not all thrusters would be active at a time. Note that the propulsion chair has indicated that the MaSMi thrusters may be able to handle double their rated throughput, 200kg rather than 100kg; if this is the case, it would remove the need for inactive thrusters.
  - See next slides for more information on Propulsion sizing cases and mass estimation

- **Power Subsystem:**
  - A rough concept of operations was modeled for power sizing purposes, using a handful of steady-state power modes:
    - A “Launch” mode, lasting 2 hours
    - A “Cruise” mode, assumed continuous
    - A “Telecom” mode, assumed to last 8 hours
    - A “Prox Ops” mode, adjusted on a per-concept basis, but often assumed continuous
  - Secondary batteries were sized to one or more “battery only” power modes, usually only Launch, but adjusted on a per-concept basis
  - Solar Arrays were sized to keep the spacecraft power positive in one of the modes (usually the larger of Telecom or Prox Ops)
  - See next slide for details on solar array and power electronics mass estimation

Systems

Power and Propulsion Sizing Cases (1/4)

- For the **Power** and **Propulsion** subsystems, a series of “Sizing Cases” were provided to the Power and Propulsion Team X chairs, who then performed subsystem designs using their standard Team X design tools. Those subsystem designs were then used to inform and calibrate the bus designs for the study’s architecture Options:

- **Power**
  - **Power Electronics:** The power electronics masses were used directly in most Options (with the exception of the CubeSat-sized spacecraft)
    - Different masses were used for systems with **Chemical** propulsion vs. with **EP** propulsion
  - **Solar Arrays**: The solar array specific power (W/kg) at End of Life (EoL) for **rigid** (87 W/kg) and **low-mass** (UltraFlex or ROSA-type) arrays (132 W/kg) were used to estimate solar array mass
    - The cost/mass ($M/kg) of rigid and low-mass arrays, along with specific designs’ array masses, were used to estimate a “cost upper” for low-mass arrays
    - Despite this cost upper, in all cases the overall mass savings from using low-mass arrays resulted in a modeled cost savings; therefore low-mass arrays were assumed to be used in all designs over 1700W.
- **Propulsion**
  - The chemical propulsion designs were used as “calibration points” for a power regression model to estimate propulsion system dry mass, of the form:
    - Propulsion Dry Mass CBE = A * $(\text{propellant mass})^B$
      - B is a power coefficient that has been regressed from historical data. There is a different coefficient for Monoprop and Biprop systems.
      - A is adjusted to fit the curve through the “calibration point”
  - The EP designs were used to build a slightly more detailed mass model:
    - Per-thruster masses for the total [Thruster + PPU + Feed System] were taken from the specific Propulsion designs, for both MaSMi and SPT-140 designs
    - The remaining tank, plumbing, and miscellaneous mass was estimated using a power regression model, regressed on all of the EP propulsion designs
    - Isp and power consumption were taken from the propulsion design as well as from a MaSMi throttle curve document provided by Damon Landau
- The next two slides show tables of Propulsion and Power “sizing cases”
- The slide after that shows a table of the study Options, and maps them to the Power and Propulsion designs used

Power and Propulsion Sizing Cases (2/4)

- The table at right shows the Power subsystem sizing cases that were run by the Power chair
- See Power report for more details on each

Power Option	Solar Array Configuration	Power Electronics for…
Power A	Rigid	SEP
Power B	Rigid	SEP
Power C	Rigid	Chemical. Prop
Power D	Rigid	SEP
Power E	Rigid	SEP
Power F	UltraFlex (“Low-mass”)	SEP
Power G	UltraFlex (“Low-mass”)	SEP

Systems

Power and Propulsion Sizing Cases (3/4)

- The table at right shows a list of the propulsion "sizing cases" that were run. See Propulsion report for details on each case.
- The table on the next slide describes the propulsion design used in each flight element. In that table:
  - "Prop D" indicates that case Prop D was used as a calibration case for a chemical system
  - "Prop E-MP" indicates that the monoprop system from case Prop E was used as a calibration case for a chemical system
  - "2x MaSMi" indicates that the EP sizing model was used, assuming two MaSMi thrusters
  - "(2+2)x MaSMi" indicates that the EP sizing model was used, assuming 4 total MaSMi thrusters, of which 2 are active at a time and 2 are inactive
  - All SEP buses include both an EP system and a monoprop system

Propulsion Case	Type	Comment
Prop A	EP (1x MaSMi) + Monoprop Blowdown	Varying thruster power and Isp
Prop B		
Prop C	Monoprop Blowdown	
Prop D	Biprop (dual mode)	
Prop E	EP (2x MaSMi) + Monoprop Blowdown	Varying thruster power, Isp, and number of engines active.
Prop F		
Prop G		
Prop H		
Prop I	EP (5x MaSMi) + Monoprop Blowdown	
Prop J	Biprop (dual mode)	
Prop K	Monoprop Blowdown	
Prop O-A	EP (2x SPT-140) + Monoprop Blowdown	Varying Xe quantity
Prop O-B		
Prop O-C	EP (4x SPT-140) + Monoprop Blowdown	

Power and Propulsion Sizing Cases (4/4)

This table shows the Propulsion Case, Power Electronics design, and Solar Array type for all 30 options + sub-options.

Option #	Option Name	Flight Elements		Prop Case	Power Electronics	Solar Arrays
		Qty	Name			
1	Flyby Recon	1	Flyby S/C	Prop C	Chem	Rigid
2	Rendezvous	1	Rendezvous S/C	Prop D	Chem	Rigid
2-E	Rendez SEP	1	Rendezvous S/C	1x MaSMi + Prop E-MP	SEP	Rigid
3	Rendez high ΔV	1	Rendezvous S/C	2x MaSMi + Prop E-MP	SEP	Low-mass
4	Rendez Radar-Lidar	1	Rendezvous S/C	Prop D	Chem	Rigid
4-E	Rendez Radar-Lidar	1	Rendezvous S/C	1x MaSMi + Prop E-MP	SEP	Low-mass
5	Rendez high ΔV BigSat	1	Rendezvous S/C	1x SPT-140 + Prop E-MP	SEP	Low-mass
6	Rendez 2FE	1	Rendezvous Mother S/C	Prop D	Chem	Rigid
		1	Deployed Lander			
6-E	Rendez 2FE	1	Rendezvous Mother S/C	1x MaSMi + Prop E-MP	SEP	Rigid
		1	Deployed Lander			
8-1	Tour Multiple (1xB)	1	Flyby S/C (A)	Prop C	Chem	Rigid
		1	Flyby S/C (B)	Prop C	Chem	Rigid
		1	Dual Payload Adapter			
8-2	Tour Multiple (2xB)	1	Flyby S/C (A)	Prop C	Chem	Rigid
		2	Flyby S/C (B)	Prop C	Chem	Rigid
		1	Multi Payload Adapter			
8-3	Tour Multiple (3xB)	1	Flyby S/C (A)	Prop C	Chem	Rigid
		3	Flyby S/C (B)	Prop C	Chem	Rigid
		1	Multi Payload Adapter			
9	Tour CubeSats	1	Flyby Mother S/C	Prop C	Chem	Rigid
		4	Flyby CubeSat			

Option #	Option Name	Flight Elements		Prop Case	Power Electronics	Solar Arrays
		Qty	Name			
10-MP	Intercept	1	Intercept S/C	Prop K	Chem	Rigid
10-BP	Intercept	1	Intercept S/C	Prop J	Chem	Rigid
11	Rendezvous wNED	1	Rendezvous S/C	Prop D	Chem	Rigid
11-E	Rendezvous wNED	1	Rendezvous S/C	1x MaSMi + Prop E-MP	SEP	Rigid
12	Rendezvous wNED (2E)	1	Rendezvous S/C	Prop D	Chem	Rigid
12-E	Rendezvous wNED (2E)	1	Rendezvous S/C	1x MaSMi + Prop E-MP	SEP	Rigid
13-DI	Kinetic Impact (DI)	1	Flyby Observer S/C	Prop C	Chem	Rigid
		1	Impactor S/C	Prop E-MP	Chem	Rigid
13-DART	Kinetic Impact (DART)	1	Mothership Impactor	Prop K	Chem	Rigid
		1	Flyby Observer SmallSat	Prop E-MP	SmallSat	Rigid
14	Kinetic Impact (SEP obs)	1	SEP Rendez. Observer S/C	2x MaSMi + Prop E-MP	SEP	Low-mass
		1	Impactor S/C	Prop J	Chem	Rigid
15-A-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	2x MaSMi + Prop E-MP	SEP	Low-mass
15-B-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	(2+2)x MaSMi + Prop E-MP	SEP	Low-mass
15-C-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	(4+4)x MaSMi + Prop C	SEP	Low-mass
15-A-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2x SPT-140 + Prop E-MP	SEP	Low-mass
15-B-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2x SPT-140 + Prop C	SEP	Low-mass
15-C-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2x SPT-140 + Prop C	SEP	Low-mass
16	Gravity Tractor	1	SEP Deflector S/C	2x MaSMi + Prop E-MP	SEP	Low-mass
17	Ion Beam & Gravity Tractor	1	SEP Deflector S/C	2x MaSMi + Prop E-MP	SEP	Low-mass

Systems

Design Assumptions – Launch Vehicles

- For the initial estimate of mission costs, *all concepts* were assumed to launch on a Falcon 9
  - The Falcon 9 can provide far more mass to C3 = 2.0 km2/s2 than is needed for many of the concepts
  - Other concepts approach the launch vehicle mass capability, and some are even sized to "max out" the capability
- As a post-study exercise, all concepts were also evaluated against an assumed future low-cost, low-mass launch vehicle
  - This vehicle does not represent a single currently available vehicle; rather, it is an amalgamation of multiple existing as well as planned small launch vehicles
  - It is assumed that a kick stage could be used, which it is assumed, vs. launch to LEO without the kick stage, a) reduces the mass capability to about 30% b) doubles the cost, and c) cuts the available volume in half. These factors are already included in the table below (without a kick stage, it could be assumed to launch ~3000kg to LEO, for $22M).

Launch Vehicle Name	Orbit Description	Fairing Volume			Launch Mass Capability	Cost
		Cylinder				
		Diam	H	Volume		
		m	m	m^3	kg	$M (FY2025)
Falcon 9 C3=2.0	C3 2.0 km2/sec2	4.6	6.6	110	1595	$ 115
Future Small LV with kick stage	C3 2.0 km2/sec2	2	1.25	3.9	1000	$ 44

Sizing Spacecraft to Minimum Masses

- In some of the options examined in this study, there are *minimum mass constraints* on the design
  - In the designs that use impactors (Options 13 and 14), it is desired to keep the mass of the impactor above 300kg
  - In the gravity tractor options (Option 16 and 17), higher spacecraft mass increases the gravitational force between the spacecraft and the body, and a minimum mass of 400kg was needed to allow the design to achieve the required deflection
- In these cases, it is assumed that the *margined dry mass* must be greater than the constraint value
  - It is assumed that additional ballast mass can be added to the spacecraft if its design does not grow to consume all of its mass margin
  - It is assumed that the propellant tanks have been emptied at the time that the minimum mass constraint must be met
  - There may be un-modeled cost effects associated with adding this ballast, but this is assumed to be negligible
- If the margined dry mass of the spacecraft by itself does not meet the constraint, additional “dumb mass” is added as carried mass, in increments of 5kg, until the constraint is met.
  - This added mass is always costed at a flat $100k. However, the carried mass will also “ripple” through the structures, propulsion, and ACS designs, which increases the dry bus mass that is used for cost estimation.

Systems

Instruments Table

- The table at right shows the instrument mass and power values assumed in this study
- The MEVs highlighted yellow indicate values that were provided by the customer
- For purposes of modeling, CBE values were back-calculated assuming 20% contingency,
- 23% additional margin (over the MEV, but as fraction of CBE) was applied to arrive at a margined values. This is consistent with JPL's Design Principles (both v8 and v9).

Instrument Name	Mass				Power Consumed			
					Average while On			Margined
	CBE	Cont.	MEV	Margined	CBE	Cont.	MEV	Margined
	kg	%	kg	kg	W	%	W	W
Vis NAC	8.6	20%	10	12.3	14.2	20%	17	20.2
Vis/NIR spec	5.2	20%	6	7.4	10.0	20%	12	14.3
Vis WAC	3.4	20%	4	4.9	8.3	20%	10	11.9
Radar (HERA Heavy)	5.2	20%	6	7.4	75.0	20%	90	107.1
Lidar	11.2	20%	13	16.0	25.8	20%	31	36.9
Radar (HERA Light)	3.4	20%	4	4.9	8.3	20%	10	11.9
JCam	2.8	20%	3.2	3.9	11.7	20%	14	16.7
NED Simulator	8.6	20%	10	12.3				
NED Simulator (Deployable)	8.6	20%	10	12.3				

Systems

Spacecraft Bus Table

- The table at right describes the various spacecraft bus elements used in this study (including “dumb” masses or adapters included in WBS6)
- Most of the buses were sized specifically for a single option.
  - Only Option 8 re-used a bus, from Option 1
- See Cost report for details on the cost modeling approach

Spacecraft Bus			Spacecraft Bus Mass						Cost		
Alias	Downlink Rate	ΔV	Dry				Wet		1st Unit	Nth Unit	Learning Curve Approach
			CBE	Cont	MEV	Margined	Propellant	Margined			
	Mbit/s	m/s	kg	%	kg	kg	kg	kg	$M (FY2025)	$M (FY2025)	
Bus 1 - Flyby Monoprop [M]	0.02	250	108.8	20%	130.5	155.4	25.9	181.3	$58.0	$23.2	Nth unit discount
Bus 2 - Biprop [M]	0.1	2000	243.4	20%	292.0	347.7	378.6	726.3	$117.3	$46.9	Nth unit discount
Bus 2-E - SEP [M]	0.1	2000	199.0	20%	238.8	284.3	51.3	335.6	$98.3	$39.3	Nth unit discount
Bus 3 - Hi ΔV SEP [M]	0.1	4000	233.7	20%	280.5	333.9	119.7	453.6	$115.8	$46.3	Nth unit discount
Bus 4 - Biprop [M]	0.1	2000	320.0	20%	384.0	457.1	517.7	974.8	$149.0	$59.6	Nth unit discount
Bus 4-E - SEP [M]	0.1	2000	212.9	20%	255.4	304.1	58.7	362.8	$104.3	$41.7	Nth unit discount
Bus 5-S - High ΔV BigSat - SPT-140	0.1	4000	312.2	20%	374.6	446.0	136.0	582.0	$153.7	$61.5	Nth unit discount
Bus 6 - Biprop [M]	0.1	2000	336.8	20%	404.2	481.1	559.4	1040.6	$155.8	$62.3	Nth unit discount
Bus 6-E - SEP [M]	0.1	2000	206.6	20%	247.9	295.2	59.9	355.1	$101.6	$40.6	Nth unit discount
Deployed Lander Bus			9.5	20%	11.0	13.5	0.0	13.5	$18.9	$7.6	Nth unit discount
ESPA Ring (6-port)			133.0	5%	139.7	171.8		171.8	$6.0	$6.0	No discount
Dual Payload Adapter			92.5	20%	111.0	132.2		132.2	$6.0	$6.0	No discount
Bus 9 - Mothership [M]	0.04	250	171.5	20%	205.8	245.0	65.1	310.1	$86.3	$34.5	Nth unit discount
12U Cubesat Bus with Prop			20.0	20%	24.0	28.6		28.6	$13.8	$5.5	Nth unit discount
12U Cubesat Dispenser			8.0	20%	9.6	11.4		11.4	$0.2	$0.2	No discount
Bus 10-MP - Monoprop [M]	0.02	1000	200.9	20%	241.1	287.0	207.0	494.0	$99.2	$39.7	Nth unit discount
Bus 10-BP - Biprop [M]	0.02	1000	208.8	20%	250.5	298.3	148.0	446.2	$102.6	$41.0	Nth unit discount
Bus 11 - Biprop [M]	0.1	2000	301.7	20%	362.0	430.9	478.4	909.3	$141.5	$56.6	Nth unit discount
Bus 11-E - SEP [M]	0.1	2000	215.8	20%	259.0	308.3	57.5	365.8	$105.6	$42.2	Nth unit discount
Bus 12 - Biprop [M]	0.1	2000	321.1	20%	385.3	458.7	512.6	971.3	$149.5	$59.8	Nth unit discount
Bus 12-E - SEP [M]	0.1	2000	209.4	20%	251.3	299.2	56.9	356.1	$102.8	$41.1	Nth unit discount
Bus 13-DI - Monoprop [M]	0.02	500	170.5	20%	204.7	243.6	83.6	327.2	$85.9	$34.4	Nth unit discount
Impactor 13-DI - Monoprop [M]		50	75.1	20%	90.1	107.3	7.0	114.3	$41.9	$16.8	Nth unit discount
Dumb Mass 13-DI			190.0	0%	190.0	190.0		190.0	$0.1	$0.1	No discount
13-DART - Impactor Mothership	0.02	500	179.9	20%	215.9	257.0	100.8	357.8	$90.0	$36.0	Nth unit discount
13-DART - Flyby SmallSat	0.02	50	18.2	20%	21.9	26.0	0.7	26.7	$13.8	$5.5	Nth unit discount
13-DART - Dumb Mass			45.0	20%	120.0	120.0		120.0	$0.1	$0.0	Nth unit discount
Bus 14 - SEP [M]	0.1	6000	241.6	20%	289.9	345.1	194.6	539.7	$119.1	$47.7	Nth unit discount
Impactor 14 - Biprop [M]		1000	204.1	20%	244.9	291.5	154.1	445.7	$100.5	$40.2	Nth unit discount
Dumb Mass 14			50.0	0%	50.0	50.0		50.0	$0.1	$0.1	No discount
Bus 15-A-M - SEP [M] - UltraFlex	0.1	2000	230.7	20%	276.8	329.6	88.4	418.0	$115.6	$46.2	Nth unit discount
Bus 15-B-M [M] - UltraFlex, 2M	0.1	2000	288.6	20%	346.3	412.2	240.8	653.0	$139.8	$55.9	Nth unit discount
Bus 15-C-M [M] - UltraFlex, 6M	0.1	2000	474.2	20%	569.0	677.4	832.0	1509.4	$217.7	$87.1	Nth unit discount
Bus 15-A-S - SEP [M] - UltraFlex	0.1	2000	376.9	20%	452.3	538.5	120.4	658.9	$188.1	$75.2	Nth unit discount
Bus 15-B-S - SEP [M] - UltraFlex	0.1	2000	403.5	20%	484.2	576.4	266.1	842.5	$198.6	$79.4	Nth unit discount
Bus 15-C-S - SEP [M] - UltraFlex	0.1	2000	530.9	20%	637.1	758.4	827.6	1586.0	$248.2	$99.3	Nth unit discount
Bus 16 - SEP [M] - UltraFlex, 2M	0.1	2000	260.4	20%	312.5	372.0	190.7	562.7	$127.0	$50.8	Nth unit discount
16 - Dumb Mass			15.0	0%	15.0	15.0		15.0	$0.1	$0.1	No discount
Bus 17 - SEP [M] - UltraFlex, 2M	0.1	2000	272.3	20%	326.8	389.1	217.8	606.8	$133.1	$53.2	Nth unit discount

Systems

Options Overview – Option Results, Week 1 (Characterization)

- The table below shows the architecture options in more detail, including a summary of the results, for week 1 (characterization)
- *Note that Option 7 was not explicitly studied; to the resolution of this study, it is not distinguishable from #1, but would in reality likely have a higher cost due to a longer Phase E and more complex navigation requirements.

Option #	Option Name	Flight Elements		ΔV	Instruments Mass (marg)	Total Data Volume	Spacecraft Bus: Primary		Spacecraft Bus: 2		Spacecraft Bus Mass (Margined): Dry	Propellant	Bus Total	Stack Mass	LV Cap.	Mission Cost
		Qty	Name	m/s	kg	Gbit	Qty	Name	Qty	Name	kg	kg	kg	kg	kg	$M FY2025
1	Flyby Recon	1	Flyby S/C	250	19.7	30	1	Bus 1 - Flyby Monoprop [M]			155.4	25.9	181.3	201.0	1595.0	277.1
2	Rendezvous	1	Rendezvous S/C	2000	12.3	400	1	Bus 2 - Biprop [M]			347.7	378.6	726.3	738.6	1595.0	401.4
2-E	Rendez SEP	1	Rendezvous S/C	2000	12.3	400	1	Bus 2-E - SEP [M]			284.3	51.3	335.6	347.9	1595.0	358.7
3	Rendez high ΔV	1	Rendezvous S/C	4000	12.3	400	1	Bus 3 - Hi ΔV SEP [M]			333.9	119.7	453.6	465.9	1595.0	398.1
4	Rendez Radar-Lidar	1	Rendezvous S/C	2000	35.7	400	1	Bus 4 - Biprop [M]			457.1	517.7	974.8	1010.5	1595.0	615.2
4-E	Rendez SEP Radar-Lidar	1	Rendezvous S/C	2000	35.7	400	1	Bus 4-E - SEP [M]			304.1	58.7	362.8	398.5	1595.0	514.4
5	Rendez high ΔV BigSat	1	Rendezvous S/C	4000	12.3	400	1	Bus 5-S - High ΔV BigSat - SPT-140			446.0	136.0	582.0	594.3	1595.0	483.6
6	Rendez 2FE	1	Rendez Mother S/C	2000	33.2	400	1	Bus 6 - Biprop [M]			481.1	559.4	1040.6	1092.2	1595.0	643.9
		1	Deployed Lander	-	4.9	0	1	Deployed Lander Bus			13.5	0.0	13.5	-	-	
6-E	Rendez SEP 2FE	1	Rendez Mother S/C	2000	33.2	400	1	Bus 6-E - SEP [M]			295.2	59.9	355.1	406.8	1595.0	521.7
		1	Deployed Lander	-	4.9	0	1	Deployed Lander Bus			13.5	0.0	13.5	-	-	
7 (like #1)	Tour Single	1	Flyby S/C	250	19.7	30	1	Bus 1 - Flyby Monoprop [M]			155.4	25.9	181.3	201.0	1595.0	277.1*
8-1	Tour Multiple (1xB)	1	Flyby S/C (A)	250	19.7	30	1	Bus 1 - Flyby Monoprop [M]			155.4	25.9	181.3	526.7	1595.0	350.5
		1	Flyby S/C (B)	250	12.3	30	1	Bus 1 - Flyby Monoprop [M]			155.4	25.9	181.3	-	-	
		1	Dual Payload Adapter	-	0.0	0	1	Dual Payload Adapter			132.2	0.0	132.2	-	-	
8-2	Tour Multiple (2xB)	1	Flyby S/C (A)	250	19.7	30	1	Bus 1 - Flyby Monoprop [M]			155.4	25.9	181.3	759.9	1595.0	410.4
		2	Flyby S/C (B)	250	12.3	30	1	Bus 1 - Flyby Monoprop [M]			155.4	25.9	181.3	-	-	
		1	Multi Payload Adapter	-	0.0	0	1	ESPA Ring (6-port)			171.8	0.0	171.8	-	-	
8-3	Tour Multiple (3xB)	1	Flyby S/C (A)	250	19.7	30	1	Bus 1 - Flyby Monoprop [M]			155.4	25.9	181.3	953.5	1595.0	470.3
		3	Flyby S/C (B)	250	12.3	30	1	Bus 1 - Flyby Monoprop [M]			155.4	25.9	181.3	-	-	
		1	Multi Payload Adapter	-	0.0	0	1	ESPA Ring (6-port)			171.8	0.0	171.8	-	-	
9	Tour CubeSats	1	Flyby Mother S/C	250	19.7	30	1	Bus 9 - Mothership [M]	4	12U Cubesat Dispenser	290.7	65.1	355.8	505.5	1595.0	434.0
		4	Flyby CubeSat	-	3.9	30	1	12U Cubesat Bus with Prop			28.6	0.0	28.6	-	-	

Green: Under cap | Orange: < 10% over | Red: >10% over

Options Overview – Option Results, Week 2 (Mitigation)

- The table below shows the architecture options in more detail, including a summary of the results, for week 2 (mitigation)

Option #	Option Name	Flight Elements Qty	Flight Elements Name	ΔV m/s	Instruments Mass (marg) kg	Total Data Volume Gbit	Spacecraft Bus Primary Qty	Spacecraft Bus Primary Name	Spacecraft Bus 2 Qty	Spacecraft Bus 2 Name	Spacecraft Bus Mass (Margined) Dry kg	Propellant kg	Bus Total kg	Stack Mass kg	LV Cap. kg	Mission Cost $M FY2025
10-MP	Intercept (Monoprop)	1	Intercept S/C	1000	34.4	30	1	Bus 10-MP - Monoprop [M]			287.0	207.0	494.0	528.5	1595.0	423.0
10-BP	Intercept (Biprop)	1	Intercept S/C	1000	34.4	30	1	Bus 10-BP - Biprop [M]			298.3	148.0	446.2	480.7	1595.0	430.7
11	Rendezvous wNED	1	Rendezvous S/C	2000	24.6	400	1	Bus 11 - Biprop [M]			430.9	478.4	909.3	933.9	1595.0	456.3
11-E	Rendezvous SEP wNED	1	Rendezvous S/C	2000	24.6	400	1	Bus 11-E - SEP [M]			308.3	57.5	365.8	390.4	1595.0	375.2
12	Rendezvous wNED (2E)	1	Rendezvous S/C	2000	29.5	400	1	Bus 12 - Biprop [M]			458.7	512.6	971.3	1000.9	1595.0	529.9
12-E	Rendezvous SEP wNED (2E)	1	Rendezvous S/C	2000	29.5	400	1	Bus 12-E - SEP [M]			299.2	56.9	356.1	385.6	1595.0	424.8
13-DI	Kinetic Impact (DI)	1	Flyby Observer S/C	500	12.3	30	1	Bus 13-DI - Monoprop [M]			243.6	83.6	327.2	648.8	1595.0	432.0
		1	Impactor S/C	50	4.9	0	1	Impactor 13-DI - Monoprop [M]	1	Dumb Mass 13-DI	297.3	7.0	304.3	-	-	
13-DART	Kinetic Impact (DART)	1	Mothership Impactor	500	4.9	0	1	13-DART - Impactor Mothership	1	13-DART - Dumb Mass	388.4	100.8	489.2	524.8	1595.0	369.6
		1	Flyby Observer SmallSat	50	3.9	30	1	13-DART - Flyby SmallSat			26.0	0.7	26.7	-	-	
14	Kinetic Impact (SEP obs)	1	SEP Rendez. Observer S/C	6000	12.3	400	1	Bus 14 - SEP [M]			345.1	194.6	539.7	1052.6	1595.0	639.1
		1	Impactor S/C	1000	4.9	0	1	Impactor 14 - Biprop [M]	1	Dumb Mass 14	341.5	154.1	495.7	-	-	
15-A-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	2000	4.9	400	1	Bus 15-A-M - SEP [M] - UltraFlex			329.6	88.4	418.0	422.9	1595.0	385.1
15-B-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	2000	4.9	400	1	Bus 15-B-M [M] - UltraFlex, 2M			412.2	240.8	653.0	658.0	1595.0	439.8
15-C-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	2000	4.9	400	1	Bus 15-C-M [M] - UltraFlex, 6M			677.4	832.0	1509.4	1514.4	1595.0	615.6
15-A-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2000	4.9	400	1	Bus 15-A-S - SEP [M] - UltraFlex			538.5	120.4	658.9	663.8	1595.0	548.6
15-B-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2000	4.9	400	1	Bus 15-B-S - SEP [M] - UltraFlex			576.4	266.1	842.5	847.4	1595.0	572.5
15-C-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	2000	4.9	400	1	Bus 15-C-S - SEP [M] - UltraFlex			758.4	827.6	1586.0	1590.9	1595.0	684.2
16	Gravity Tractor	1	SEP Deflector S/C	2000	17.2	400	1	Bus 16 - SEP [M] - UltraFlex, 2M	1	16 - Dumb Mass	387.0	190.7	577.7	594.9	1595.0	479.4
17	Ion Beam & Gravity Tractor	1	SEP Deflector S/C	2000	17.2	400	1	Bus 17 - SEP [M] - UltraFlex, 2M			389.1	217.8	606.8	624.1	1595.0	492.7

Green: Under cap | Orange: < 10% over | Red: >10% over

Systems

Options Overview – Low-Cost Launch Vehicle Options

- As a post-study exercise, all Launch Vehicle options were switched from a Falcon 9 to a "Future small Launch Vehicle" with a kick stage, which is assumed to have a mass capability of 1000kg to C3=2.0 and a cost of $44M. See Launch Vehicle Assumptions slide.
- Four options which were previously over the cost cap (4-E, 6-E, 12, and 15-A-S) are now under it (highlighted gold below). However, Option 12 is *just barely* over the mass constraint.
- Six options are too massive to fit on the small LV (4, 6, 12, 14, 15-C-M, and 15-C-S), and their stack masses are highlighted red below. However, all but one (#12) are still over the cost cap.
- Other missions may not fit due to volumetric constraints (2m fairing assumed). It is beyond the scope of this study to properly estimate S/C volume; however, some which appear immediately problematic are highlighted. Options 8-2 and 8-3 assume the use of an ESPA ring for a multi-payload adapter, which would not fit, though a 3-S/C stack (8-2) may be possible with other adapters, and a larger fairing may be possible with future "small plus" launchers. 13-DI assumes the impactor could be ESPA-like, though it need not specifically be an ESPA.

Option #	Option Name	Flight Elements		Stack Mass	LV Cap.	Mission Cost
		Qty	Name	kg	kg	$M FY2025
1	Flyby Recon	1	Flyby S/C	201.0	1000.0	205.7
2	Rendezvous	1	Rendezvous S/C	738.6	1000.0	330.1
2-E	Rendez SEP	1	Rendezvous S/C	347.9	1000.0	287.3
3	Rendez high ΔV	1	Rendezvous S/C	465.9	1000.0	326.8
4	Rendez Radar-Lidar	1	Rendezvous S/C	1010.5	1000.0	543.8
4-E	Rendez Radar-Lidar	1	Rendezvous S/C	398.5	1000.0	443.0
5	Rendez high ΔV BigSat	1	Rendezvous S/C	594.3	1000.0	412.2
6	Rendez 2FE	1	Rendezvous Mother S/C	1092.2	1000.0	572.6
		1	Deployed Lander	-	-	
6-E	Rendez 2FE	1	Rendezvous Mother S/C	406.8	1000.0	450.3
		1	Deployed Lander	-	-	
8-1	Tour Multiple (1xB)	1	Flyby S/C (A)	526.7	1000.0	279.2
		1	Flyby S/C (B)	-	-	
		1	Dual Payload Adapter	-	-	
8-2	Tour Multiple (2xB)	1	Flyby S/C (A)	759.9	1000.0	339.0
		2	Flyby S/C (B)	-	-	
		1	Multi Payload Adapter	-	-	
8-3	Tour Multiple (3xB)	1	Flyby S/C (A)	953.5	1000.0	398.9
		3	Flyby S/C (B)	-	-	
		1	Multi Payload Adapter	-	-	
9	Tour CubeSats	1	Flyby Mother S/C	505.5	1000.0	362.6
		4	Flyby CubeSat	-	-	

Option #	Option Name	Flight Elements		Stack Mass	LV Cap.	Mission Cost
		Qty	Name	kg	kg	$M FY2025
10-MP	Intercept	1	Intercept S/C	528.5	1000.0	351.7
10-BP	Intercept	1	Intercept S/C	480.7	1000.0	359.3
11	Rendezvous wNED	1	Rendezvous S/C	933.9	1000.0	384.9
11-E	Rendezvous wNED	1	Rendezvous S/C	390.4	1000.0	303.9
12	Rendezvous wNED (2E)	1	Rendezvous S/C	1000.9	1000.0	458.6
12-E	Rendezvous wNED (2E)	1	Rendezvous S/C	385.6	1000.0	353.4
13-DI	Kinetic Impact (DI)	1	Flyby Observer S/C	648.8	1000.0	360.7
		1	Impactor S/C	-	-	
13-DART	Kinetic Impact (DART)	1	Mothership Impactor	524.8	1000.0	298.2
		1	Flyby Observer SmallSat	-	-	
14	Kinetic Impact (SEP obs)	1	SEP Rendez. Observer S/C	1052.6	1000.0	567.7
		1	Impactor S/C	-	-	
15-A-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	422.9	1000.0	313.8
15-B-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	658.0	1000.0	368.4
15-C-M	Ion Beam (MaSMi)	1	SEP Deflector S/C	1514.5	1000.0	544.3
15-A-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	663.8	1000.0	477.3
15-B-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	847.4	1000.0	501.1
15-C-S	Ion Beam (SPT-140)	1	SEP Deflector S/C	1590.9	1000.0	612.8
16	Gravity Tractor	1	SEP Deflector S/C	594.9	1000.0	408.0
17	Ion Beam & Gravity Tractor	1	SEP Deflector S/C	624.1	1000.0	421.3

Green: Under cap | Orange: < 10% over | Red: >10% over | Gold: was over, now under

Systems

Options Discussion - Characterization (1/3)

- **Option 1: Flyby Recon:** Simple recon mission, with just a Narrow Angle Camera and a Vis/NIR spectrometer. This mission should be achievable with SmallSat class spacecraft. We assumed that all avionics were single-string, and used COTS SmallSat components wherever possible. The concept requires only 250m/s of ΔV, which is easily achievable with a blowdown monoprop system. For telecom, we assumed a 1m deployable high-gain antenna (HGA) plus an Iris radio, which would trickle back the 30Gb of flyby data at 20kb/s at an Earth-S/C range of 1AU over about 3 months. As modeled, this came in well under the cost cap ($277M). A large chunk of this was the Launch Vehicle cost of a Falcon 9 ($115M), which can launch far more mass than is necessary for this mission.
  - **Note that these avionics assumptions (single-string, SmallSat) were applied to almost all other options (excluding Option 5)**
  - **The telecom design (1m deployable HGA + Iris → 20kb/s @ 1AU range) was assumed in almost all *flyby* options (excluding Option 9)**
  - Note also that if faster data return is required (due to planetary defense data latency needs), the Telecom design could be switched to be the same as in Option 2, with a mass and cost increase (~20kg wet margined mass increase, ~$13M total mission cost increase), to get all the data back in 2 weeks.

- **Option 2: Rendezvous**: Rendezvous mission, using a biprop chemical propulsion bus to provide 2km/s of ΔV. This option uses the same simple payload suite as Option 1 (NAC + Vis/NIR spec), but returns a larger data volume (400Gb) due to a longer time spent at the body. We assumed that the telecom subsystem from Option 1 could be upgraded with a 25W Solid State Power Amplifier (SSPA) rather than using the Iris radio's default 4W SSPA. With the same 1m deployable HGA, it would achieve a data rate of 100kb/s at an Earth-S/C range of 1AU, enabling it to return the 400Gb of data over about 7 months. As modeled, this concept came in under the cost cap ($401M). Total margined launch mass was a bit under half of the Falcon 9's capability, at 739kg.
  - **Note that this telecom design (1m deployable HGA + Iris + 25W RF SSPA → 100kb/s @ 1AU range) was assumed in all *rendezvous* options (excluding Option 5)**

- **Option 2-E: Rendez SEP:** This was a SEP variant on Option 2. A rendezvous mission with a NAC + Vis/NIR spec, using a SEP spacecraft, providing 2km/s of ΔV. Comes in as lower mass (and therefore lower estimated cost, $359M) than the biprop Option 2. Assumes use of a single MaSMi Hall thruster, thrusting for 0.6y at 85% duty cycle, consuming 779 W CBE and with an I_{sp} of 1542s. This S/C used rigid solar arrays, assuming 87W/kg at end of life (EoL) at 1AU, rated at 1414W, and sized for spacecraft power requirements while simultaneously thrusting with the EP system and transmitting with the telecom system.

Options Discussion – Characterization (2/3)

- **Option 3: Rendez high ΔV**: A rendezvous mission, with the same simple NAC + Vis/NIR spec payload as Options 1 and 2, but assuming a higher ΔV of 4km/s. Uses a SEP bus, and assumes two MaSMi Hall thrusters operating simultaneously to keep the burn duration under 1 year. Total burn duration of 0.8yr, at a duty cycle of 85%. Still comes in under the cost cap, at $398M.

- **Option 4: Rendez Radar-Lidar**: A rendezvous mission, with a biprop system providing 2km/s ΔV like in Option 2, but where the payload includes a Radar and a Lidar. The cost of the payload (and to some extent the ripple of the added mass and power through the S/C bus) pushes the cost over the cap to $615M.

- **Option 4-E: Rendez SEP Radar-Lidar**: A SEP variant of Option 4. Using SEP reduces the mass, and therefore the estimated cost, but it is still a bit over the cost cap at $514M. Like in Option 2-E, uses a single MaSMi thruster, at 779W CBE power, and has rigid arrays sized to 1414W. Burn time is 0.7 years at a duty cycle of 85%.

- **Option 5: Rendez high ΔV BigSat**: Like Option 3 (high ΔV of 4km/s, simple NAC + Vis/NIR spec payload, SEP design), but does not use SmallSat components, and uses a single SPT-140 thruster rather than a MaSMi. Note that this was the only option to use “traditional” footprint avionics rather than SmallSat components. All avionics were still single-string. Since the SPT-140 requires more power than a MaSMi, the arrays were larger at 7205W, and we therefore used low-mass arrays to save mass and ultimately cost (see Design Assumptions section). The higher thrust of the SPT-140 means it only needs to burn for 0.3yr to achieve 4km/s of ΔV. This bus was higher mass than in Option 3, but the mission still came in under the cost cap at $484M. Note that using MaSMi rather than an SPT-140 would bring the est. mission cost down to $422M.

Systems

Options Discussion – Characterization (3/3)

- **Option 6: Rendez 2FE:** Like Option 4 (rendezvous with Lidar and Radar), but now with an additional small deployed lander as well. The deployed lander was assumed to have a mass of 15kg and cost of $44M including its instruments. The instrument was assumed to be equivalent to the HERA Light radar. Option 4 was already over the cost cap, so this pushes it further over, to $644M.

- **Option 6-E: Rendez SEP 2FE:** A SEP variant on Option 6. Using SEP brings down the estimate mass and therefore cost, to $521M. It is still a bit over the cost cap. As with Option 2-E and 4-E, the design used a single MaSMi thruster running at 779W CBE, with arrays sized to 1414W EoL. Total burn time to reach 2km/s of ΔV was 0.8 years at a duty cycle of 85%.

- (Note that there originally was a customer-specified Option 7, which would have involved a tour of multiple bodies, with the same payload suite and ΔV as Option 1. Compared to Option 1, in theory this would involve a longer Phase E and greater Mission Design and navigation cost. However, those effects were not modeled in this study, so to the resolution of this study Option 7 was indistinguishable from Option 1, and it was not examined separately.)

- **Option 8-1: Tour Multiple (1xB):** Launches two flyby spacecraft together, on a single launch; both spacecraft follow the same multi-body tour trajectory (like the un-explored Option &), and nominally fly by 4 distinct bodies. At each body, they each follow very slightly different trajectories to provide alternate perspectives (e.g. on different sides of the body). Spacecraft A has a visible NAC as well as a Vis/NIR spectrometer. Spacecraft B has only a NAC. Both S/C are assumed to launch together with a Dual Payload Adapter, though equally feasible using an ESPA ring. The data volume is assumed to be 30Gbit per spacecraft per flyby, and it is assumed that there is time between and after the flybys to relay it all back, which would take (3 mo/target) x (4 targets) = 12mo total. Well under the cost cap at $351M.
- **Option 8-2: Tour Multiple (2xB):** Like 8-1, but increases the quantity of Spacecraft B to 2, for a total of 3 Spacecraft. All are assumed to launch together on an ESPA ring. Still under the cost cap at $410M.
- **Option 8-3: Tour Multiple (3xB):** Again increasing the number of copies of Spacecraft B, now to 3, for a total of 4 S/C. All are assumed to launch together on an ESPA ring. Still under the cost cap at $470M.
  - Note that Increasing the number of B again to 4 would have brought the total just over the cap to $530M.
- **Option 9: Tour CubeSats:** A multi-target "tour" concept like Option 8. A Mothership carries 4x 12U CubeSats, and launches one during each of 4 body flybys, to give an extra vantage point for the flyby. The CubeSats relay data back to the mothership via a UHF link. The CubeSats were assumed to have a margined wet mass of 29kg and a first-unit cost of $14M. The data return requirement for the Mothership would approximately double, so a $10W_{RF}$ SSPA was assumed, to bring the data rate to 40kb/s @ 1AU range and keep the downlink time to 3mo/target. Each CubeSat was carried in an 11kg (margined) and ~$200k dispenser. Under the cost cap at $434M.

Options Discussion – Mitigation (1/5)

- **Option 10-MP: Intercept (monoprop):** A demonstration for a Nuclear Explosive Device (NED) intercept mission. The S/C would carry a non-deployable NED simulator with a trigger, and would do a very close flyby of the body, with a small radar to measure distance. It would demonstrate "detonating" with the trigger, though it would not actually contain any explosives. This sub-option uses a monoprop propulsion system to provide 1km/s of ΔV, with a margined launch wet mass of 529kg. Under the cost cap at $423M.
- **Option 10-BP: Intercept (biprop):** A sub-option, like Option 10-MP but using a biprop propulsion system to provide the same 1km/s of ΔV. The modeled total wet mss was lower (for a margined launch wet mass of 481kg), but the dry mass was very slightly higher due to the higher complexity of the biprop system, and the modeled total cost was therefore slightly higher at $431M. There is an un-modeled propulsion cost upper that could push the cost up even further in a grass-roots estimate.

- **Option 11: Rendezvous wNED:** Carries a non-deployable NED simulator, as in Option 10; but performs a rendezvous with the body, rather than a flyby. Would make a very close approach to the body, and activate the NED simulator's trigger. Uses a biprop propulsion system, providing 2km/s of ΔV. Under the cost cap at $456M. Note that as modeled here, a version with a deployed (rather than body-fixed) NED simulator (as in option 12) would not change the cost estimate, though there could be un-modeled cost uppers in that case.
- **Option 11-E: Rendezvous SEP wNED:** A SEP variant on Option 11. Comes in at lower mass and therefore lower estimated cost, at $375M. As in Option 2-E, 4-E, and 6-E, uses a single MaSMi thruster running at 779W CBE and I_{sp}=1542s, with rigid solar arrays sized to 1414W. The total burn time is 0.7 years at a duty cycle of 85% to reach the total 2km/s of ΔV.

- **Option 12: Rendezvous wNED (2E):** A rendezvous concept which includes a deployed NED simulator. The spacecraft would release the NED simulator on a trajectory towards the asteroid, and the simulator would trigger when in close proximity or contact. More expensive than Option 11 because of the addition of a radar, which was for characterization of the body (and not for triggering). Over the cost cap at $529M.
- **Option 12-E: Rendezvous SEP wNED (2E):** A SEP variant of Option 12. Switching to a SEP design brings the estimated mass down enough that the estimated cost is now under the cap, at $425M. SEP design is still a single MaSMi @ 779W CBE, 1414W arrays. Total burn time of 0.7 years for 2 km/s of ΔV.

Options Discussion – Mitigation (2/5)

Note that all 3 concepts on this page are sized to *minimum masses.* See note in Design Assumptions about minimum mass S/C.

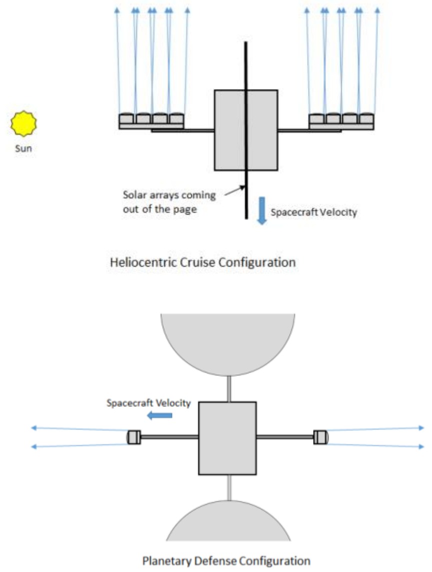
- **Option 13-DI: Kinetic Impact (DI):** A two-element kinetic impact demo, architecturally similar to Deep Impact. A monoprop mothership S/C (500m/s ΔV capability) deploys a kinetic impact S/C one day prior to close approach, and observes the impact. The kinetic impact vehicle carries 190kg of additional "dumb" mass to bring its total margined mass to at least 300kg. Note that this “dumb” mass was assigned a fixed $100k cost, and was not fed to the bus cost model; but there are additional “taxes” on that mass, because the impactor was sized to carry it (affecting structures, propulsion, thermal, ACS); and the mothership was sized to carry that impactor. Under the cost cap at $432M.
- **Option 13-DART: Kinetic Impact (DART):** A two-element kinetic impact demo, architecturally similar to the DART mission. A monoprop mothership serves as the impact vehicle, but releases a small observer spacecraft before impact. The observer is estimated to be similar in mass to a 12U CubeSat, and it is assumed that it could fit in a 12U form factor, and released from a 12U dispenser; therefore a CubeSat CDS board was used (rather than the SmallSat box assumed in other concepts). The mothership impactor carries 45kg of additional "dumb" mass to bring the impacting mass above 300kg. This option is more mass-efficient than Option 13-DI, and therefore lower estimated cost, because the naturally heavier S/C (mothership with propulsion) is the impactor. The total amount of “dumb” mass can be reduced vs. 13-DI, and therefore the “taxes” (structure, propulsion, thermal, ACS) to carry that extra mass are reduced, for a reduction in costed dry mass. This concept is enabled by the ability to observe and send back data from the small observer spacecraft, which requires capable pointing (using SmallSat ACS components) and a deep-space comm system in a small form factor (the same Iris + 1m deployable HGA as in other flyby concepts). Comes in under the cap at $370M.

- **Option 14: Kinetic Impact (SEP obs):** This concept is a kinetic impact demonstration that uses two spacecraft that separate from each other immediately after release from the Launch Vehicle. A SEP observer spacecraft, carrying only a Narrow Angle Camera for observing the impact, takes a 6km/s low-thrust trajectory to rendezvous with the body. It uses 2x MaSMi engines, running simultaneously at 779W CBE each, with low-mass (132W/kg) solar arrays sized at 2639W EoL, and takes 1.3 years of thrusting (at a duty cycle of 85%) to achieve 6km/s of ΔV. Meanwhile, a monoprop impactor S/C uses an impulsive (1km/s) trajectory to target the body, and needs 25kg of additional “dumb mass” to bring its total margined mass to over 300kg. The “dumb mass” was increased further to 50kg, such that it plus the predicted bus mechanical and structure mass (85kg) would exceed the 133kg mass of a 6-port ESPA ring, It was then assumed that the impactor spacecraft could use an ESPA ring (or similar tube structure) for the bulk of its primary structure, and could carry the load of the SEP spacecraft above it on the launch vehicle, obviating the need for a dual payload adapter (DPA). The use of two relatively high ΔV spacecraft pushes this concept over the cost cap ($639M), despite modest camera-only payloads. **Note that there is additional mission design work needed to show that such a concept is feasible.**

Systems

Options Discussion – Mitigation (3/5)

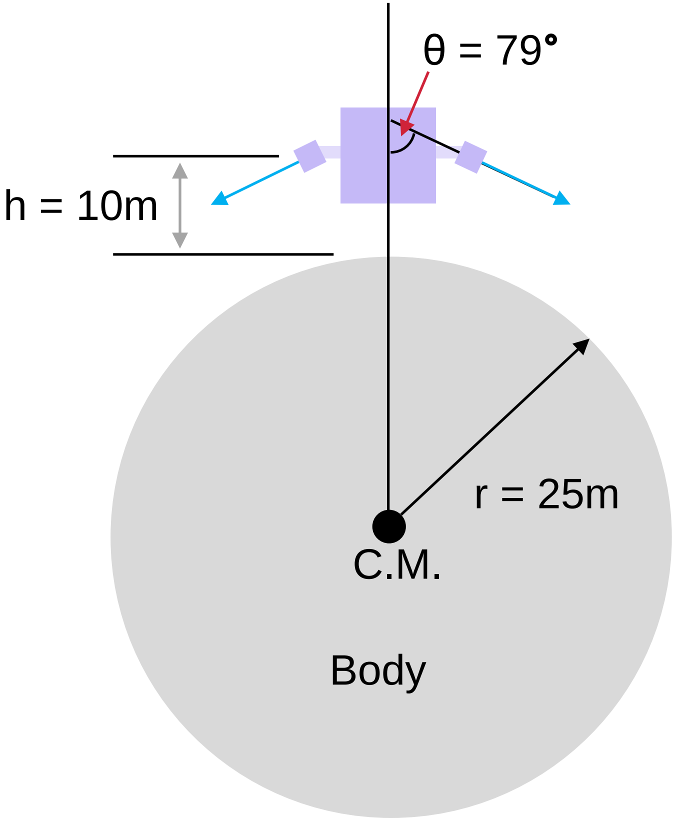
- **Option 15: Ion Beam:** (with 6 sub-options) is for an Ion Beam demonstration. It involves a SEP spacecraft, with at least two engines that can be operated simultaneously. The engines are assumed to be on outriggers, such that they can be used simultaneously for both propulsion and deflection (see diagram at right, from Brophy et al paper). Ion Beam deflection is best performed with an *Ion engine*, with as tight an exhaust ion beam as possible; however, it was assumed that it can also be demonstrated (at lower effectiveness, 50% assumed) with Hall thrusters (which have wider exhaust spread). This assumption perhaps merits additional scrutiny. It was also assumed that 1 mm/s of change to the body's velocity would be detectable, and preliminary analysis (see Mission Design report) indicated that this would require only 32kg of Xenon propellant from a Hall thruster (SPT-140, or MaSMi at a high throttle setting) for a demonstration on a 50m body. Because there is uncertainty in this figure, we ran three primary sub-cases, with varying Xe quantities for the demonstration: A) 32kg B) 150kg C) max out the launch allocation (~600kg). Further, sub-cases were run with both MaSMi engines (M) and SPT-140 engines (S), for a total of 6 sub-options. In all cases it was assumed that the ΔV budget prior to the start of the deflection demonstration was 2km/s.
- All options use low-mass arrays (132W/kg EoL)
- The second two MaSMi options (15-B-M and 15-C-M) add extra inactive thrusters to avoid exceeding the MaSMi's 100kg rated throughput limit; however, if this limit is revised upwards, the dry mass and cost can come down. **Note that the propulsion chair indicates that the throughput could possibly be increased to 200kg, which would eliminate all need for extra thrusters.**



Option #	Xe for demo	EP Thruster	Thruster I_{sp}	Array power (EoL)	Burn time for 2km/s ΔV @ 85% D.C.	Burn time for deflection @ 100% D.C.	Mission Cost ($M, FY25)	Comments
15-A-M	32 kg	2x MaSMi	1790 s	3536 W	115 days	70 days	385.1	
15-B-M	150 kg	4x MaSMi (2 active)	1790 s	3536 W	178 days	328 days	439.8	Requires 2 extra engines to keep MaSMi under rated 100kg throughput. Just 3 could handle throughput, but must be an even number of them.
15-C-M	615 kg	8x MaSMi (4 active)	1790 s	6883 W	205 days	1.8 years	615.6	Requires 4 extra engines for throughput. Propellant increased to max out Falcon 9 mass allocation without exceeding engine throughput.
15-A-S	32 kg	2x SPT-140	1780 s	14.3 kW	29 days	11 days	548.6	A "Ferrari"! Very high thrust, short burn durations.
15-B-S	150 kg	2x SPT-140	1780 s	14.3 kW	37 days	52 days	572.5	
15-C-S	600 kg	2x SPT-140	1780 s	14.3 kW	69 days	210 days	684.2	Xe increased to max out Falcon 9 mass allocation. Throughput is 79% allowable.

Options Discussion – Mitigation (4/5)

- **Option 16: Gravity Tractor:** This is a Gravity Tractor demonstration, with 100kg of Xe for the deflection. The spacecraft configuration is assumed to be the same as in Option 15, with two MaSMi engines on outriggers. In addition to a WAC, it carries a Vis/NIR spec for characterizing the body, and a small radar to maintain spacing with the body. It is assumed that 1mm/s of ΔV in the body is measurable; to achieve this in 1.5 years, with a low-mass (and therefore low-cost) spacecraft, the spacecraft must fly very close to the body. For the point design in this study, it must fly as close as 10m from the surface of a 50m diameter body with a 400kg spacecraft (see Mission Design report for analysis description and body assumptions). In all cases it was assumed that the ΔV budget prior to the start of the deflection demonstration was 2km/s.
- To reduce the thrust (to keep the S/C from just flying away from the body), and to avoid plume impingement on the body, the two MaSMi engines must be throttled down to their lowest level (~0.01N each) and canted off to the side by 79°. This angle is 46° to clear the limb, plus an assumed 10° to clear the Hall thruster plume spread, and 26° additional to reduce thrust further so that the spacecraft mass can be kept low. This canting reduces the efficiency of the system, and most of the impulse is lost; but it still closes with a reasonable propellant budget of 100kg.
- Whether flying this close is actually achievable may be questionable, especially since that body will not actually be a sphere and may have protrusions beyond 10m from its “mean” surface.
  - The concept is actually *not* sensitive to the asteroid’s density or mass; it can simply change the cant angle, and achieve the same ΔV in the same time.
  - However, it is sensitive to the asteroid’s size, and 1mm/s will not be achievable from anything larger than 50m. As the asteroid gets smaller, the achievable ΔV goes up, but the required distance gets small (it stays around 20% of the diameter) and the required precision on the thrust vectoring gets very fine as the thrusters point “nearly straight out”.
  - Flying so close to the asteroid, only half of all possible “tug” vectors are available, due to the asteroid blocking the Sun, affecting array power.
- With both MaSMi thrusters on, each at 779W CBE, it achieves 2km/s of ΔV for cruise and rendezvous in 200 days. The gravity tractor demonstration is then conducted at minimum throttle, with an assumed duty cycle of 100%, and takes 1.5 years.
- Note that higher S/C mass helps. We added 15kg to make this concept close, and there is plenty of room on the LV to add additional “dumb” mass, but the ripple effects on the spacecraft will quickly drive it over the cost cap, as the concept already stands at $479M.
- It is recommended to evaluate a wide population of known asteroids, and to find the distribution of achievable deflection ΔV’s.



Options Discussion – Mitigation (5/5)

- **Option 17*: Ion Beam & Gravity Tractor:** This option was added after the study was over, as a “systems-only bonus option”, and is not documented in most other subsystem reports. This option combines Options 15 and 16 into a single demo of both Ion Beam (32kg Xe) and a gravity tractor (100kg of Xe), since both propellant amounts were fairly low. Uses 2x MaSMi thrusters, in the same outrigger configuration as Options 15 and 16, and the payload (with radar) of Option 16.
- In cruise and rendezvous, it takes 169 days to achieve a ΔV of 2km/s
- The Ion Beam deflection takes 70 days, at maximum power (1064W CBE per engine, Isp of 1790s), to achieve 1mm/s deflection using 32kg of Xe.
- The Gravity Tractor deflection takes 1.5 years, at minimum power (226W CBE per engine, Isp of 947s), 10m from the surface of the 50m asteroid, to achieve 1mm/s deflection using 100kg of Xe.
- The concept closes without any added “dumb mass”, almost exactly at the cap at $492M.
  - Note that this was run with the calculated propellant loads for the ion beam and gravity tractor concepts, both of which have a good deal of uncertainty.
  - It is therefore quite likely that a technically feasible “double demo” would be over the cost cap; or even that there is no technically feasible concept (especially the gravity tractor portion, see previous slide)

Systems

Conclusions, Risks, and Recommendations

- All of the concepts studied here appear technically feasible (though there are questions about the actual deflection capabilities of the Ion Beam and Gravity Tractor demonstrations)
- Some are over the cost cap, but there are many that appear achievable within $500M.
- It is worth noting that there is a high degree of uncertainty in these cost estimates. The bus costs are estimated almost entirely based on dry mass, and Phase E costs are not modeled explicitly. It may not be achievable within the budget of this study, but one or two Team X studies for selected point designs would help increase confidence in (or allow recalibration of) the cost estimates. Or this could be useful for architectures which are close to the cost cap or have technical uncertainties that warrant greater scrutiny. Examples:
  - A single Rendezvous case (ex. Option 1), and a single Flyby case (ex. Option 2), could be used as “calibration points” to adjust all other flyby and rendezvous architecture cost estimates.
  - Options 3 (“Rendez high ΔV”) and 5 (“Rendez high ΔV BigSat”) are somewhat close to the cap, and differ only in their choice of smallsat vs. traditional avionics. A useful exercise might be to a) examine the mission design more closely, and b) determine the maximum reasonable extent to which smallsat avionics might be used in such a mission.
  - Options 15, 16, and 17 are all subject to significant uncertainty with respect to the ion beam and gravity tractor assumptions. If these models and assumptions can be clarified, a design session could then identify a feasible spacecraft implementation.

Instruments Report

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Instruments

Design Requirements

- Mission:
  - Multiple Mission options studied, all to small bodies (PHO/NEO)
  - Class C instruments
  - FY25
- Constraints
  - Variety: Fly-by, impact, rendezvous
- Measurement
  - Spectra, imaging, penetrating radar, LIDAR topography, distance

Instruments

Overview

Instruments list from the customer team (MEV mass and power provided):

Name	Description	Mass kg	Power W	Options	Instrument Analogues/Equivalents
Vis NAC	Visible Narrow Angle Camera	10	17	1, 7-10, 13, 14	Ave of BASIX, Condor from MSSS
Vis/NIR Spec	Visible Near Infrared Spectrometer	6	12	1-9, 11, 12, 16	MLPS + optics
Vis WAC	Visible Wide Angle Camera	4	10	2-6, 10-16	Ave of BASIX and Trident
Radar 1 (HFR)	High Frequency Radar	6	137	4	HERA heavy
Radar 2 (LFR)	Low Frequency Radar	4	50	6, 10, 12, 16	HERA bistatic (light)
LIDAR	Laser topography	13	31	4, 6	LOLA
Cubesat cam	Simple color camera	3.2	14	9, 13 DART	Jcam

Instruments

Cost Summary

	Mass kg	Power W	Most likely Mass	Most likely max Power	Cost $M (NICM 50%)	Options	Notes
Vis NAC	10	17	12.3	17	8.4	1, 7-10, 13, 14	Ave of BASIX, Condor from MSSS
Vis/NIR Spec	6	12	7.38	12	5.5	1-9, 11, 12, 16	MLPS + optics
Vis WAC	4	10	4.92	10	4.2	2-6, 10-16	Ave of BASIX and Trident
Radar 1 (HFR)	6	137	7.38	137	47	4	HERA heavy
Radar 2 (LFR)	4	50	4.92	50	24.7	6, 10, 12, 16	HERA bistatic (light)
LIDAR	13	31	15.99	31	12.6	4, 6	LOLA
Cubesat cam	3.2	14	3.9	14	4.5	9, 13 DART	Jcam

- Most likely max values of mass and power used for NICM. Mass includes DP9 margins.
- Assumed 36 month lifetime.
- Note that NICM database does not include low mass RADAR analogues, so those cost numbers are not well anchored (European builds)

Instruments

Cost

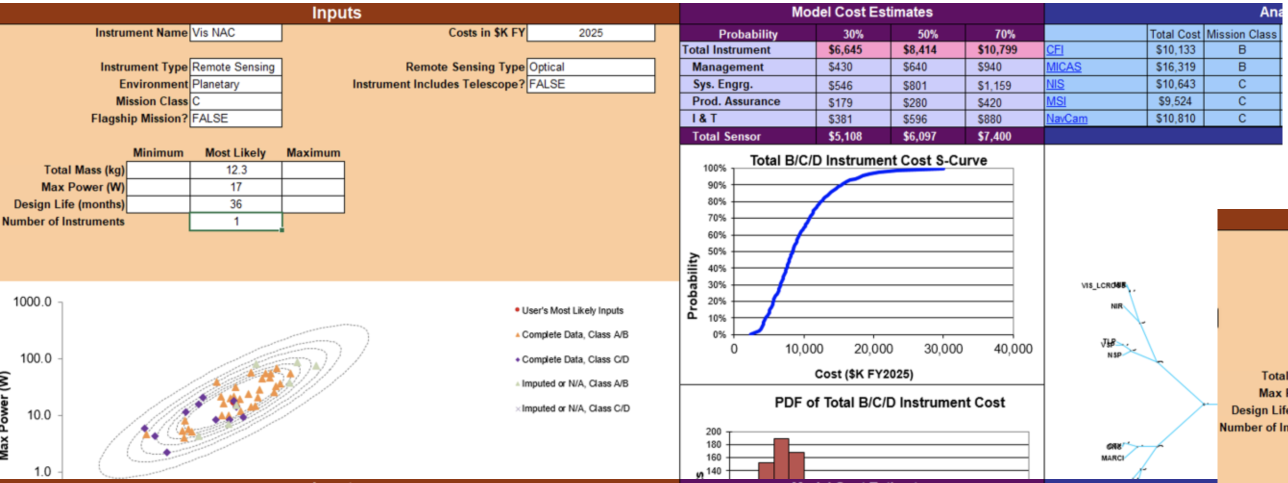
- Cost Assumptions
  - Assume new builds
  - If multiple copies in a given architecture, additional units will cost 40% of the initial unit

- Cost Method
  - NICM System run performed
  - Used the customer mass numbers plus DP 9 required margin
  - Maximum power used in NICM run
  - 50% probability numbers used since reserves will be applied at a higher level

- See next sheets for the NICM runs

Instruments

NICM Runs



Inputs

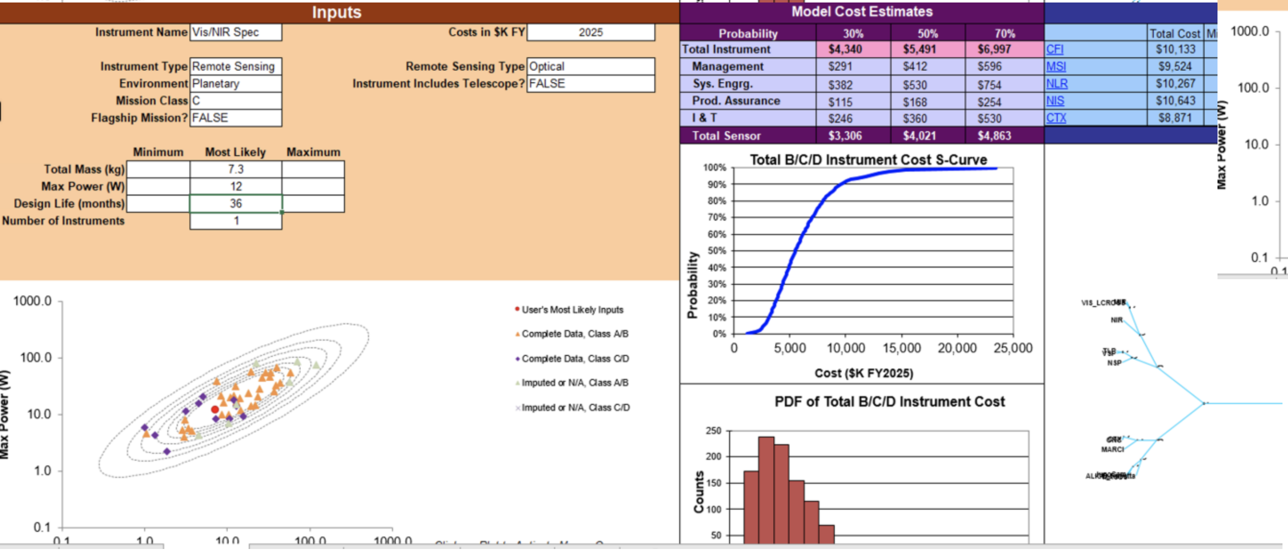
Instrument Name	Vis NAC	Costs in $K FY	2025
Instrument Type	Remote Sensing	Remote Sensing Type	Optical
Environment	Planetary	Instrument Includes Telescope?	FALSE
Mission Class	C		
Flagship Mission?	FALSE		

	Minimum	Most Likely	Maximum
Total Mass (kg)		12.3	
Max Power (W)		17	
Design Life (months)		36	
Number of Instruments		1	

Model Cost Estimates

Probability	30%	50%	70%
Total Instrument	$6,645	$8,414	$10,799
Management	$430	$640	$940
Sys. Engrg.	$546	$801	$1,159
Prod. Assurance	$179	$280	$420
I & T	$381	$596	$880
Total Sensor	$5,108	$6,097	$7,400

	Total Cost	Mission Class
CFI	$10,133	B
MICAS	$16,319	B
NIS	$10,643	C
MSI	$9,524	C
NavCam	$10,810	C



Inputs

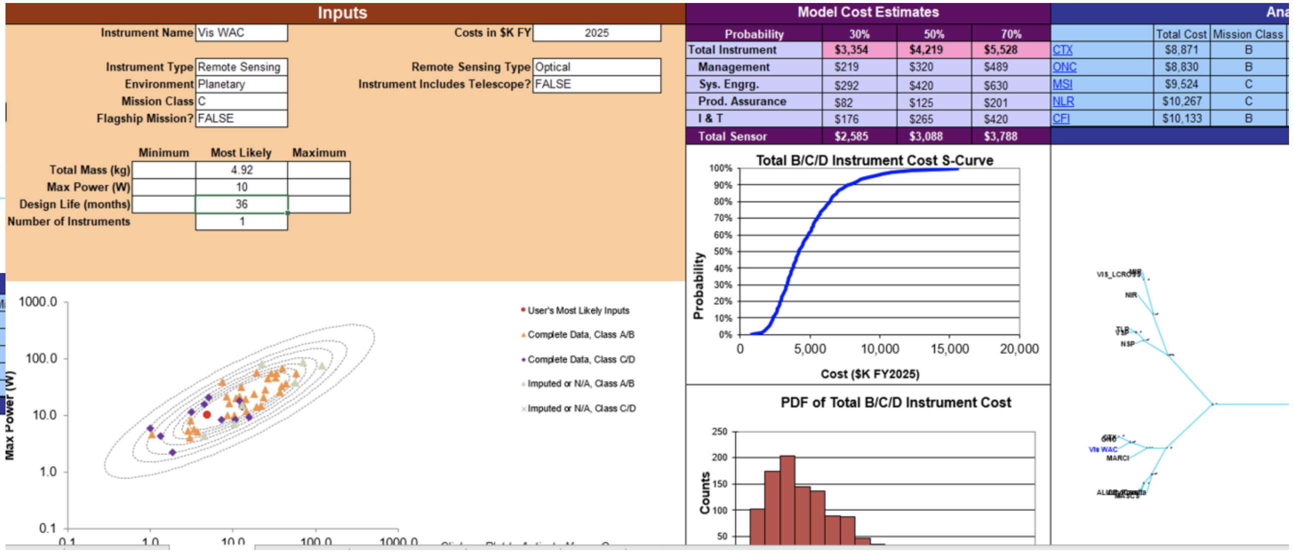
Instrument Name	Vis/NIR Spec	Costs in $K FY	2025
Instrument Type	Remote Sensing	Remote Sensing Type	Optical
Environment	Planetary	Instrument Includes Telescope?	FALSE
Mission Class	C		
Flagship Mission?	FALSE		

	Minimum	Most Likely	Maximum
Total Mass (kg)		7.3	
Max Power (W)		12	
Design Life (months)		36	
Number of Instruments		1	

Model Cost Estimates

Probability	30%	50%	70%
Total Instrument	$4,340	$5,491	$6,997
Management	$291	$412	$596
Sys. Engrg.	$382	$530	$754
Prod. Assurance	$115	$168	$254
I & T	$246	$360	$530
Total Sensor	$3,306	$4,021	$4,863

	Total Cost
CFI	$10,133
MSI	$9,524
NLR	$10,267
NIS	$10,643
CTX	$8,871



Inputs

Instrument Name	Vis WAC	Costs in $K FY	2025
Instrument Type	Remote Sensing	Remote Sensing Type	Optical
Environment	Planetary	Instrument Includes Telescope?	FALSE
Mission Class	C		
Flagship Mission?	FALSE		

	Minimum	Most Likely	Maximum
Total Mass (kg)		4.92	
Max Power (W)		10	
Design Life (months)		36	
Number of Instruments		1	

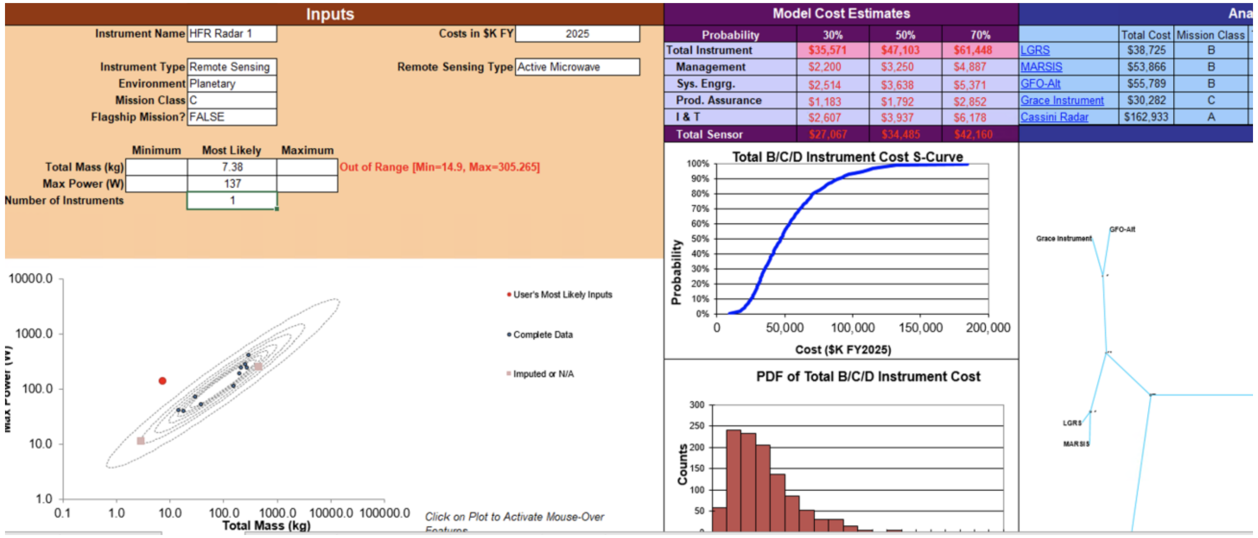
Model Cost Estimates

Probability	30%	50%	70%
Total Instrument	$3,354	$4,219	$5,528
Management	$219	$320	$489
Sys. Engrg.	$292	$420	$630
Prod. Assurance	$82	$125	$201
I & T	$176	$265	$420
Total Sensor	$2,585	$3,088	$3,788

	Total Cost	Mission Class
CTX	$8,871	B
ONC	$8,830	B
MSI	$9,524	C
NLR	$10,267	C
CFI	$10,133	B

Instruments

NICM Runs



Inputs

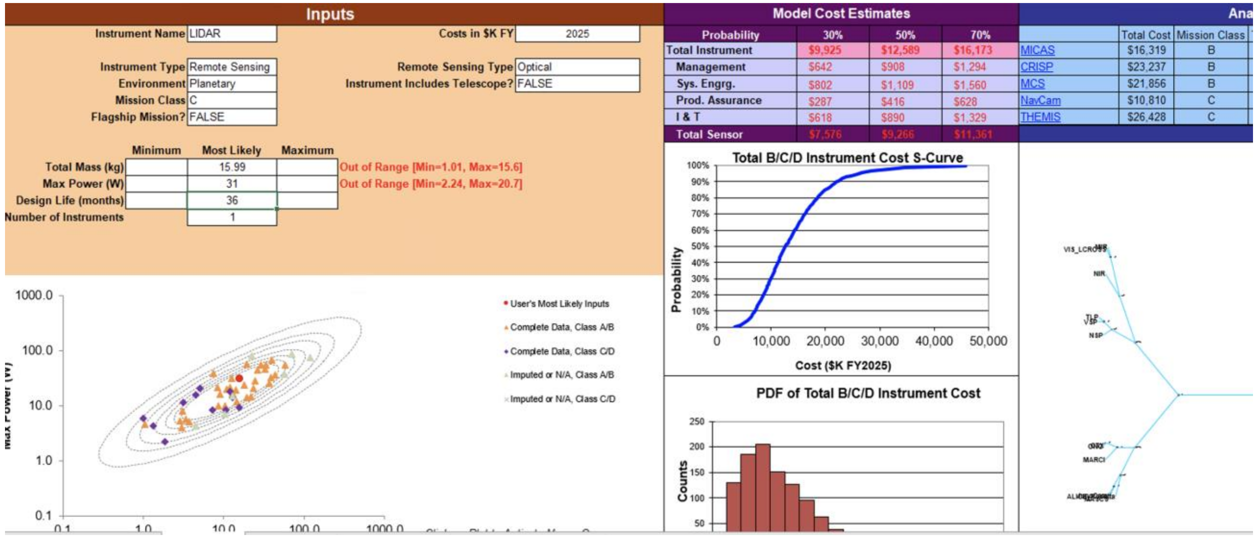
Instrument Name	HFR Radar 1
Costs in $K FY	2025
Instrument Type	Remote Sensing
Remote Sensing Type	Active Microwave
Environment	Planetary
Mission Class	C
Flagship Mission?	FALSE

	Minimum	Most Likely	Maximum	
Total Mass (kg)		7.38		Out of Range [Min=14.9, Max=305.265]
Max Power (W)		137		
Number of Instruments		1		

Model Cost Estimates

Probability	30%	50%	70%
Total Instrument	$35,571	$47,103	$61,448
Management	$2,200	$3,250	$4,887
Sys. Engrg.	$2,514	$3,638	$5,371
Prod. Assurance	$1,183	$1,792	$2,852
I & T	$2,607	$3,937	$6,178
Total Sensor	$27,067	$34,485	$42,160

	Total Cost	Mission Class
LGRS	$38,725	B
MARSIS	$53,866	B
GFO-Alt	$55,789	B
Grace Instrument	$30,282	C
Cassini Radar	$162,933	A



Inputs

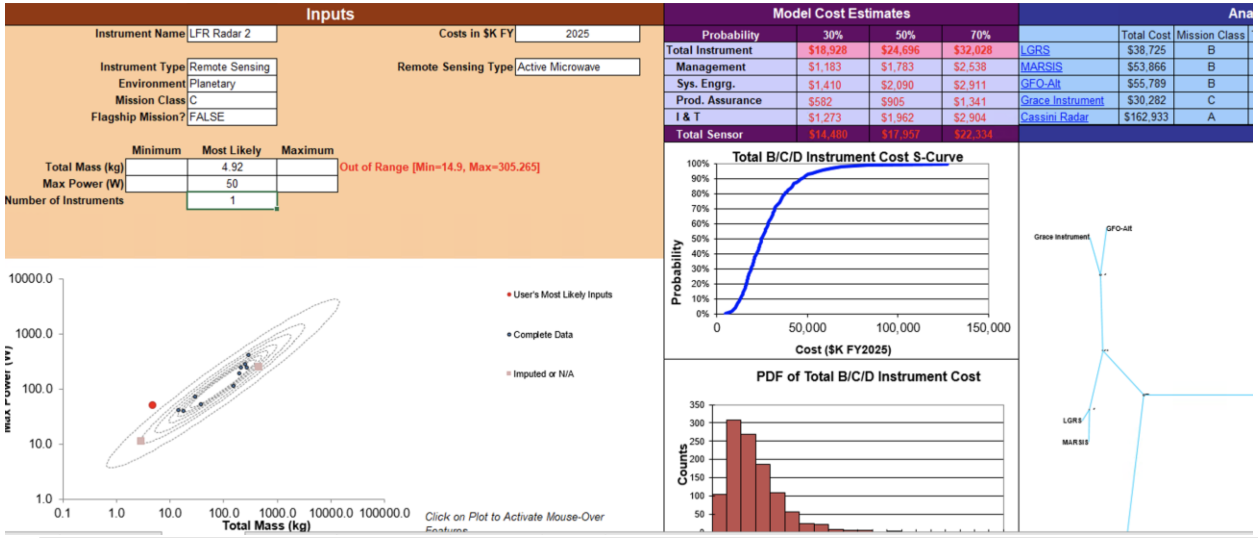
Instrument Name	LIDAR
Costs in $K FY	2025
Instrument Type	Remote Sensing
Remote Sensing Type	Optical
Environment	Planetary
Instrument Includes Telescope?	FALSE
Mission Class	C
Flagship Mission?	FALSE

	Minimum	Most Likely	Maximum	
Total Mass (kg)		15.99		Out of Range [Min=1.01, Max=15.6]
Max Power (W)		31		Out of Range [Min=2.24, Max=20.7]
Design Life (months)		36		
Number of Instruments		1		

Model Cost Estimates

Probability	30%	50%	70%
Total Instrument	$9,925	$12,589	$16,173
Management	$642	$908	$1,294
Sys. Engrg.	$802	$1,109	$1,560
Prod. Assurance	$287	$416	$628
I & T	$618	$890	$1,329
Total Sensor	$7,576	$9,266	$11,361

	Total Cost	Mission Class
MICAS	$16,319	B
CRISP	$23,237	B
MCS	$21,856	B
NavCam	$10,810	C
THEMIS	$26,428	C



Inputs

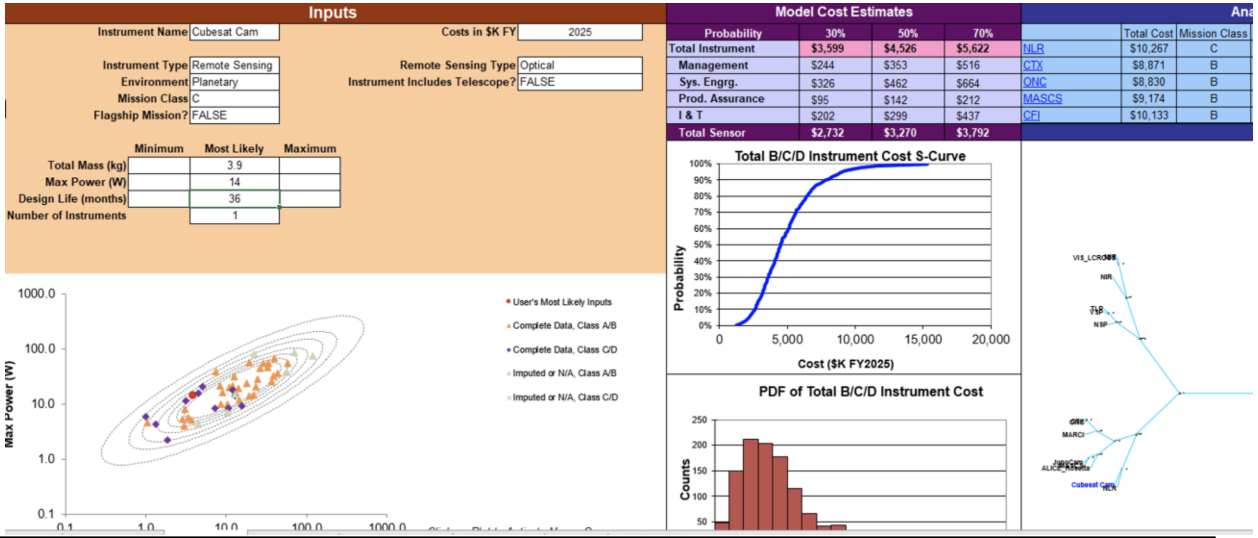
Instrument Name	LFR Radar 2
Costs in $K FY	2025
Instrument Type	Remote Sensing
Remote Sensing Type	Active Microwave
Environment	Planetary
Mission Class	C
Flagship Mission?	FALSE

	Minimum	Most Likely	Maximum	
Total Mass (kg)		4.92		Out of Range [Min=14.9, Max=305.265]
Max Power (W)		50		
Number of Instruments		1		

Model Cost Estimates

Probability	30%	50%	70%
Total Instrument	$18,928	$24,696	$32,028
Management	$1,183	$1,783	$2,538
Sys. Engrg.	$1,410	$2,090	$2,911
Prod. Assurance	$582	$905	$1,341
I & T	$1,273	$1,962	$2,904
Total Sensor	$14,480	$17,957	$22,334

	Total Cost	Mission Class
LGRS	$38,725	B
MARSIS	$53,866	B
GFO-Alt	$55,789	B
Grace Instrument	$30,282	C
Cassini Radar	$162,933	A



Inputs

Instrument Name	Cubesat Cam
Costs in $K FY	2025
Instrument Type	Remote Sensing
Remote Sensing Type	Optical
Environment	Planetary
Instrument Includes Telescope?	FALSE
Mission Class	C
Flagship Mission?	FALSE

	Minimum	Most Likely	Maximum
Total Mass (kg)		3.9	
Max Power (W)		14	
Design Life (months)		36	
Number of Instruments		1	

Model Cost Estimates

Probability	30%	50%	70%
Total Instrument	$3,599	$4,526	$5,622
Management	$244	$353	$516
Sys. Engrg.	$326	$462	$664
Prod. Assurance	$95	$142	$212
I & T	$202	$299	$437
Total Sensor	$2,732	$3,270	$3,792

	Total Cost	Mission Class
NLR	$10,267	C
CTX	$8,871	B
ONC	$8,830	B
MASCS	$9,174	B
CFI	$10,133	B

Instruments

Cost

- Cost Drivers
  - Radar costs are driven by power of the radar based on the NICM database.
- Potential Cost Savings
  - Use commercially available instruments
    - COTs if possible
- Potential Cost Uppers
  - Avoid any technology maturation by using well established instrument product lines

Mission Design Report

Title: 387 4X Planetary Decadal –Planetary Defense Architecture Study

Date: 13-21 Apr 2021

Author: Charles Reynerson

Email: reynerso@jpl.nasa.gov

Mission Design

Design Requirements

- Mission:
  - Heliocentric Orbit, Targets of interest: Near Earth Orbiting (NEO) Asteroid or Potentially Hazardous Object (PHO
  - Demonstration mission is the goal, not an actual Planetary Defense operational mission.
  - 16 Architecture options examined each with Multiple Payload variations, some with multiple vehicles/elements, and a delta V was specified for each option. No specific trajectory to analyze.
  - Mission variations: Flyby, Rendezvous, Impactors, Asteroid Tour.
  - Mission Classes: Characterization mission options (1 - 9); Mitigation mission options (10 – 16)
  - Propulsion variants: Chemical (impulsive) vs Solar Electric Propulsion (SEP) (Low Thrust)
  - Target Launch Date: 2032

- Mission Design
  - 1 - 4 year nominal mission life (longest for the Tour option)
  - Sizing delta-V prop: assume dV as directed by customer, see dVs for each option in later slides
  - Trajectory constraints: No constraints other than to perform various mission options.

- Launch Vehicle
  - LV is TBD, Need to examine potential options
  - Initial LV C2 = 2.0 km^2/s^s
  - Max satellite mass: 1595 kg (Falcon 9 initial assumption)

Mission Design

Design Assumptions, Characterization (Options 1 thru 9)

- LV C3 = 2 km^2/s^2
- Delta – V variants:
  - 0.25 km/s
  - 2 km/s
  - 4 km/s

Trade space point	Mission Purpose	Payload (s) (see columns to right)	Mission Design (s/c Δv) (Assume C3 = 2 km2/s2 for all)	Comments/ notes	Telecom
First Week (Characterization)					
1A	PHO/NEO Flyby Reconnaissance – response to particular threat	Vis NAC, Vis/NIR spec, Radio Science, ~~Multi-spectral imagers?~~	0.25 km/sec	Get as much as you can in a single flyby? "Fast" development/deployment mission; 5 kbps w/ 'standard' antenna	30 Gb Total 5 kbps @ 1AU
~~1B (?)~~				~~Same as 1A but w/ 1m deployable HGA~~	~~30 Gb Total 20 kbps @ 1AU~~
2	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec	2 km/sec	See Papais Fig 13. This DV captures adequate fraction of population	400 Gb Total 100 kbps (TBC) @ 1 AU
3	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec; limited to SmallSats (cheaper/faster)	4 km/sec	SmallSat works on ESPA Grande. This DV captures large fraction of population.	400 Gb Total 100 kbps (TBC) @ 1 AU
4	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec, mono-static radar, lidar	2 km/sec	<->	400 Gb Total 100 kbps (TBC) @ 1 AU
5	PHO/NEO Rendezvous	Vis WAC, Vis/NIR Spec	4 km/sec	This DV captures large fraction of population	400 Gb Total 100 kbps (TBC) @ 1 AU
6	PHO/NEO Rendezvous (two elements)	Vis WAC, Vis/NIR spec, Bi-static GPR, and LIDAR plus **Deployable assets** for surface operations hopper (if possible) to enhance geophysical characterization of targets	2km/s	instrument costing needs to be separable	400 Gb Total 100 kbps (TBC) @ 1 AU
7	PHO/NEO Tour	Vis NAC, Radio Science, ~~Multi-spectral imagers (?)~~, NIR spec	0.25 km/sec	This DV facilitates >100 different tours per Karimi analysis 4/8/21	30 Gb Total 5 kbps @ 1AU
8	PHO/NEO Tour (multiple uSats can provide perspective)	(Same as above but instruments "disaggregated" onto usats <100 kg) (But still need NAC on all)	0.25 km/sec	This DV facilitates >100 different tours per Karimi analysis 4/8/22	30 Gb Total 5 kbps @ 1AU
9	PHO/NEO Tour (mother ship & cubesats)	Deployable cubesats for perspective to enhance characterization of targets via NACs	0.25 km/sec	1 cubesat per flyby provides perspective and mothership does DTE comm. Cubesats could impact too.	30 Gb Total 5 kbps @ 1AU

Mission Design

Design Assumptions, Mitigation (Options 10 thru 16)

- LV C3 = 2 km^2/s^2
- Delta – V variants:
  - 0.25 km/s
  - 1 km/s
  - 2 km/s
  - 6 km/s

Trade space point	Mission Purpose	Payload (s) (see columns to right)	Mission Design (s/c Δv) (Assume C3 = 2 km2/s2 for all)	Comments/ notes
First Week (Characterization)				
Second Week (Mitigation)				
10	Intercept PLUS Mitigation for short-warning scenario	NAC, WAC, plus NED. Radar plus ability to trigger @ high closing vel.	1 km/sec	Operational device would be ~ 200 kg. ~~Should this be part of the demo?~~ [assume NO]
11	Rendezvous w/Nuclear Explosive Device (NED) Simulator	NED, Vis WAC, and NIR Spec ; range radar for trigger	2 km/sec	See above re rendezvous cases
12	Rendezvous w/Nuclear Explosive Device (NED) Simulator and observer (Two elements)	Vis WAC and IR Spec ; combined with NED. two element system with observer to verify deflection result. Also Bistatic radar	2 km/s	keep characterization instruments costed separately
13	Two-element Kinetic Impact with flyby (like DI). Single launch.	Vis WAC for Impactor OpNav, NAC for crater eval on Flyby s/c	0.5 km/sec	Impact >10 km/s
14	Two-element Kinetic Impact with rendezvous observer to eval crater. Single launch.	Vis WAC for impactor OpNav, NAC for observer crater eval	1 km/s (Impactor to intercept) 6 km/sec (Observer to rendezvous)	Impact >10 km/s; observer does rendezvous first. MD requires longer cruise time and Earth GA's.
15	Characterization and Mitigation Rendezvous using Ion beam (SEP)	Vis camera WAC. Includes ability to automatically hover @ 750 m while thrusting against surface.	2 km/sec (to rendezvous only) TBD kg Xe for ion beam deflection	See Brophy paper. Assume 10 kW SA and 5 kW for SPT-140 thrusters like Psyche.
16	Gravity Tractor	advanced autonomous guidance and navigation, imagers, spectrometers, radar, radio science	2 km/sec (to rendezvous only) TBD kg Xe for GT deflection	

Mission Design

Design – Initial Heliocentric Trajectory

- Mission trajectory and orbit parameters after Launch (shown in the table)
  - LV C3 = 2 km^2/s^2
  - Delta-Vs applied after LV separation as needed for specific mission option.

Parameter	Value	Unit
Target or Destination	NEO asteroid / PHO	
Mission Type (flyby, rendezvous, lander/cruise stage, etc.)	Heliocentric Orbit, Flyby and Rendezvous missions considered, Asteroid tour considered (up to 4 targets)	
Cruise Duration	Varies with option	0.1 – 4 years
Delivery Trajectory Type	Direct Inject to Escape	
Semi-major axis	Varies with option	km
Eccentricity	Varies with option	
Periapsis Altitude	Varies with option	km
Apoapsis Altitude	Varies with option	km
Orbit Period	Varies with option	year
Eclipse Time	Varies with option	Min.
Max S/C – Sun Distance	Varies with option	AU
Max Earth – S/C Range	Varies with option	AU

Mission Design

Design: Gravity Tractor Deflection Method

- Documenting equations used in spreadsheet (passed to the Systems Chair)
- Using F=ma, F = mMG/r^3 :masses of spacecraft (m) and asteroid (M), grav. Const (G), separation distance (r). Required s/c mass: m = Fr^3/(MG)
- Mass estimate: sphere, V=4/3*pi*r^3, m = rho*V , V = pi/6* D^3
- Densities (rho): Class – density (in g/cm^3): C - 1.38, S - 2.71, M - 5.32. An S class was chosen for calculations; a value of 2.0 was used by the Systems Chair for consistency
- A ”noticeable” dV is on the order of 1 m/s
- Applied force is as the engine is designed (Isp, power, efficiency)
- Two thrusters must be used and directed such as not to impinge on the asteroid and create a reaction force from the thrust beams. Therefore the angle of thrust direction is dictated by the separation distance, the asteroid diameter, and the beam width of the thruster plume.
  - Tan(theta) = D/r, theta is the angle between r and the edge of asteroid.
  - Sep angle is theta = 0.5 * beamwidth
- **Systems Note: See Systems report for specific numerical results**

Mission Design

Design: Ion Beam Deflection Method

- Documenting equations used in spreadsheet (passed to the Systems Chair)
- Using F=ma, dV = F*dt/m = I/m (mass(m), force(F), change in time(dt), impulse(I))
- Mass estimate: sphere, V=4/3*pi*r^3, m = rho*V , V = pi/6* D^3
- Densities: Class – density (in g/cm^3): C - 1.38, S - 2.71, M - 5.32. An S class was chosen for calculations; a value of 2.0 was used by the Systems Chair for consistency
- A ”noticeable” dV is on the order of 1 m/s
- Applied force is as the engine is designed (Isp, power (P), efficiency (eta)) – see Prop report for engine specifications assumed.
- Two opposing engines are required to maintain distance
- Thrust: T = 2*eta*P*1000/(g*Isp). Where g = 9.81 m/s^2
- Mass flow rate for fuel estimates: mdot = T/(g*Isp)
- Seconds to flow = t = required time to cause asteroid to change velocity by 1 m/s
- Prop mass = mdot * t
- Note: a factor of 0.5 was used for force application efficiency (how effective the thruster force is at creating a force in the desired direction). Side components of force cancel and are not effective at moving the asteroid.
- **Systems Note: See Systems report for specific numerical results**

Mission Design

Design Rationale

- Orbit specified by customer, not part of this study
- Trajectory
  - Heliocentric
- Delta V
  - See options table
- Launch Vehicle
  - TBD

Power Report

387 4X Planetary Decadal - Planetary Defense RMA 2021-04

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Power

Design Assumptions

- Single string, class C
- Single string power electronics
- Minimal eclipses
- If SEP or high power operating modes are NOT operated during eclipse
- For the power subsystem the options are designated A through F based on power demand for array sizing as provided by the Systems Team X chair
    - Systems will assign these power subsystem options to any flight system options as appropriate

Power

Design Requirements

- Mission:
  - Near Earth Object characterization
- Power Options involve solar array sizing trades ONLY
- All options sized for near 1 AU operating distance from the sun, 2 wings
- Options C is a “standard” 30V power subsystem architecture
- All other options are have solar electric propulsion (SEP) requiring 100V inputs to the prop. system’s PPU.

Power

Design – Array

Option	Configuration	Power (W)	Active Area (m^2)	Mechanical Area (m^2)	Mass (kg)	Other
A	Rigid	600	1.97	2.32	6.88	SEP
B	Rigid	1,000	3.29	3.87	11.45	SEP
C	Rigid	270	0.89	1.04	3.11	Chemical. Prop
D	Rigid	3,000	9.87	11.61	52.86	SEP
E	Rigid	10,000	32.89	38.70	176.15	SEP
F	UltraFlex	10,000	32.89	42.12	75.50	SEP
G	UltraFlex	33,000	108.55	135.93	246.96	SEP

Power

Design – **Batteries**

- Same battery suite for all power options… battery sizing not explicitly analyzed because operating modes were not provided and couldn't be evaluated.

Flight Batteries	
Chemistry	Li-ION
Capacity (Ah)	10
Cells / Battery	10
Prime Flight Batteries	1

Power

Design – Electronics, Options A, B, D - F

Equipment Type or Level of Effort Cost Type Hover cursor here for details	Specific Equipment Type or Specific Level of Effort Hover cursor here for details	Design Level Hover cursor here for details	Board Name Alias Hover cursor here for details	Usage Information	FMs	FSs				Protos					
					FM Units	FS Units	FS Parts Kits	FS Units & Kits	FS Built Units	PT Units	EM Units	EM Sub Units Hover cursor here for details	EM Units	Total Units Cards Slices Assys Arrays RPSs Batts	Board Slice or Assy Level BTE Units
Level of Effort	none														
Level of Effort	Subsystem Engineering	Exotic													
Solar Array	Solar Array	Standard			1			0	0				0	1	
Battery	Secondary Battery	B-to-P			1		1	1	0		1		1	3	
Diodes	Diodes Assembly	B-to-P assy			1		1	1	0	1			0	3	
Chassis	4-slot power chassis	Easy			1		1	1	0	1			0	3	
Backplane	CPCI backplane (4 slots)	Easy			1		1	1	0	1			0	3	
DC-DC Converters	SMAP Housekeeping Power Converter Unit (HPCU)	B-to-P board			1		1	1	0	1			0	3	
Propulsion I/F	SMAP Guidance Interface Driver Card (GID)	B-to-P board			1		1	1	0	1			0	3	
Load Switches	SMAP Power Switch Slice - High Side (MPSS-HS)	B-to-P slice			2		1	1	0	1			0	4	
Pyro Switches	SMAP Pyro Firing Slice (PFS)	B-to-P slice			2		1	1	0	1			0	4	
HV Down Converter	High Voltage Down Converter (aka High Voltage Electronics Assy (HVEA)	New assy			1		1	1	0	1	1		1	4	1

Power

Design – **Electronics, Options C**

					FMs	FSs				Protos				
Equipment Type or Level of Effort Cost Type **Hover cursor here for details**	**Specific Equipment Type or Specific Level of Effort** **Hover cursor here for details**	**Design Level** **Hover cursor here for details**	**Board Name Alias** **Hover cursor here for details**	**Usage Information**	**FM Units**	**FS Units**	**FS Parts Kits**	**FS Units & Kits**	**FS Built Units**	**PT Units**	**EM Units**	**EM Sub Units** **Hover cursor here for details**	**EM Units**	**Total Units** **Cards Slices Assys Arrays RPSs Batts**
Level of Effort	none													
Level of Effort	Subsystem Engineering	Minimal	Includes subsystem menagement											
Solar Array	Solar Array	Standard			1			0	0				0	1
Battery	Secondary Battery	B-to-P			1		1	1	0		1		1	3
Array + ABSL Battery I/F	SMAP Array I/F & Power Slice (AIPS)	B-to-P slice			1		1	1	0	1			0	3
Power Control	SMAP Power Bus Controller slice (PBC)	B-to-P slice			1		1	1	0	1			0	3
Diodes	Diodes Assembly	B-to-P assy			1		1	1	0	1			0	3
Chassis	4-slot power chassis	Easy			1		1	1	0	1			0	3
Backplane	CPCI backplane (4 slots)	Easy			1		1	1	0	1			0	3
DC-DC Converters	SMAP Housekeeping Power Converter Unit (HPCU)	B-to-P board			1		1	1	0	1			0	3
Propulsion I/F	SMAP Guidance Interface Driver Card (GID)	B-to-P board			1		1	1	0	1			0	3
Load Switches	SMAP Power Switch Slice - High Side (MPSS-HS)	B-to-P slice			2		1	1	0	1			0	4
Pyro Switches	SMAP Pyro Firing Slice (PFS)	B-to-P slice			2		1	1	0	1			0	4

Power

Design Rationale

- Array
  - Each option sized for the system engineer recommendation for power sizing
- Batteries
  - Not sized to meet any specific mission discharge requirements: 10Ah battery used for all options
- Electronics
  - Single string system based on SMAP
    - Option C includes Power Bus Controller and Array and Battery Interface Slice
    - Options A, B, D-F replace the PBC and ABIS with a high voltage downconverter having PPU and battery I/F capabilities
  - Option C is chemical propulsion and has no high voltage requirement
  - All other options are Solar Electric Propulsions and therefor have high voltage electronics assemblies to feed ~100V to the PPUs

Power

Cost Assumptions

- The Power Electronics slides include cost assumptions for power generation, energy storage, and power electronics for the mission, including
  - Number of Flight, flight spare parts, prototypes, and engineering models
  - Complexity level (aka “design level”) used for labor costs
- Costed as a Class C mission with selective redundancy and heritage, as the vast majority of power electronics come from SMAP
- Board level and subsystem level board/slice and subsystem test equipment is inherited from SMAP or Dawn (for the SEP options HVEA (High Voltage Electronics Assembly)

Power

Cost

- Option A: $17,673 K
- Option B: $18,119 K
- Option C: $10,692 K
- Option D: $20,350 K
- Option E: $28,160 K
- Option F: $32,942 K
- Option G: $69,600 K

Power

Cost

- Cost Drivers
  - Array Size
  - 100V power input to prop. system PPU

- Potential Cost Savings
  - None

- Potential Cost Uppers
  - Because this was a quick architecture trade study the cost assumptions upon which trades were costed could become untrue and drive up costs

- Cost Uncertainty
  - Given that there were a large number of options that were not analyzed in detail, the cost uncertainty is high for all options
  - Power subsystem cost uncertainty is higher for electric propulsion options than it is for the chemical propulsion option (Prop C). JPL has delivered one high voltage power bus architecture for the DAWN mission launched in 2007, so our cost estimates are based on hardware information that is quite old.
  - Presumably the customer would use a commercial vendor such as MAXAR, Boeing, etc., that have “off the shelf” 100V power bus architectures for an EP mission

Power

Risks

- Cost risks are the common risks for any mission: requirements creep, incorrect assumptions
- These would be mitigated with cost reserve

Power

Option Comparison

Option	Array Configuration	Power (W)	Subsystem CBE Mass (kg)	Other
A	Rigid	600	41.5	SEP
B	Rigid	1,000	46.0	SEP
C	Rigid	270	22.6	Chemical. Prop
D	Rigid	3,000	87.5	SEP
E	Rigid	10,000	210.7	SEP
F	UltraFlex	10,000	110.1	SEP
G	UltraFlex	33,000	281.6	SEP

Power

Additional Comments

- For SEP options there is generally additional labor to analyze solar array performance during operations to make sure as much power is pulled from the solar array as possible

Propulsion Report

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Propulsion

Design Assumptions

- Missions are for planetary defense, to assess near earth objects
- 16 different scenarios reviewed
  - 300 m/s to over 4 km/sec of total required delta V
  - Spacecraft mass ranging from ESPA class to full size spacecraft
- Prop system will provide ACS capability, reaction wheel desaturation, and delta V maneuver capabilities
- EP systems will utilize the JPL MaSMI design
  - Power will be 500 W or 1000 W depending on the mission
- Chem Prop systems will be assessed for the low delta V missions
  - Monoprop and biprop will be assessed when such a trade makes sense
- Assume the following for duration of the various phases
  - A = 9 months, B = 9 months, C = 15 months, D = 16 months
- Provide all prop dry mass numbers to systems with zero contingency, which will be added by systems utilizing the TATER tool
  - This avoids double booking the mass contingency
- Propellant contingency and residuals are included in the prop calculations
- All cost numbers assume FY 2025 dollars

Propulsion

Design – Prop A (EP with Monoprop ACS – 1 MaSMI @ 500W)

- NOTE: Bipropellant system was assessed, and the deltaV is too high for the system to close
- Electric Propulsion System with blowdown monopropellant ACS
  - 303 m/s total delta V capability
  - 50 m/s ACS capability
  - 500 kg spacecraft wet mass
  - Isp = 1231 sec, thrust = 34mN
- Hardware - EP
  - Qty (1) JPL MaSMI thrusters (operating at 500W input)
  - Qty (1) Cobham 7161 Xenon Tank
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80389-1 Diaphragm tank
- Propellant
  - 11.3 kg of Hydrazine
  - 14.3 kg Xenon
  - 0.1 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 10.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 15.5 kg with 0 contingency
- Cost
  - System Cost $23.9M
    - Non-Recurring $11.1M
    - Recurring $12.8M

Propulsion

Design – Prop B (EP with Monoprop ACS – 1 MaSMI @ 1000W)

- NOTE: Bipropellant system was assessed, and the deltaV is too high for the system to close
- Electric Propulsion System with blowdown monopropellant ACS
  - 305 m/s total delta V capability
  - 50 m/s ACS capability
  - 500 kg spacecraft wet mass
  - Isp = 1542 sec, thrust = 68mN
- Hardware - EP
  - Qty (1) JPL MaSMI thrusters (operating at 1000W input)
  - Qty (1) Cobham 7161 Xenon Tank
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80389-1 Diaphragm tank
- Propellant
  - 11.3 kg of Hydrazine
  - 11.5 kg Xenon
  - 0.1 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 10.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 15.5 kg with 0 contingency
- Cost
  - System Cost $23.8M
    - Non-Recurring $11.1M
    - Recurring $12.7M

Propulsion

Design – Prop C (Option 1)

- Monopropellant Blowdown system
  - 303 m/s total delta V capability
  - 186 kg spacecraft wet mass
- Hardware
  - Qty (8) Aerojet/Rocketdyne MR-111C 5N thrusters
  - Qty (1) NGIS 80486-1 diaphragm tank
- Propellant
  - 28.19 kg of Hydrazine Propellant and 0.06 kg of Helium Pressurant
- Mass
  - Propulsion System Dry Mass 12.9 kg with 0 contingency
- Cost
  - System Cost $9.9M
    - Non-Recurring $5.1M
    - Recurring $4.8M

Propulsion

Design – Prop D (Option 2 – Chemical)

- NOTE: Monopropellant system was assessed, and the deltaV is too high for the system to close
- Bipropellant system
  - 2000 m/s total delta V capability
  - 50 m/s ACS capability
  - 644 kg spacecraft wet mass
- Hardware
  - Qty (4) MOOG DST-11H 22N thrusters
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80505-1 diaphragm tank – oxidizer
  - Qty (2) NGIS 80447-1 diaphragm tank – fuel
  - Qty (1) NGIS 80386-1 COPV pressurant tank – oxidizer side
  - Qty (1) NGIS 80412-1 COPV pressurant tank – fuel side
- Propellant
  - 181.2 kg of Hydrazine
  - 147.5 kg of NTO
  - 1.7 kg of Helium Pressurant
- Mass
  - Propulsion System Dry Mass 75.2 kg with 0 contingency
- Cost
  - System Cost $26.7M
    - Non-Recurring $11.1M
    - Recurring $15.6M

Propulsion

Design – Prop E (Option 3 – EP with Monoprop ACS – 1 MaSMI @ 500W)

- NOTE: Bipropellant system was assessed, and the deltaV is too high for the system to close
- Electric Propulsion System with blowdown monopropellant ACS
  - 4000 m/s total delta V capability
  - 50 m/s ACS capability
  - 700 kg spacecraft wet mass
  - Isp = 1231 sec, thrust = 34mN
- Hardware - EP
  - Qty (2) JPL MaSMI thrusters (operating at 500W input, only one burning at a time)
  - Qty (1) NGIS 80458-1 Xenon Tank
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80389-1 Diaphragm tank
- Propellant
  - 11.6 kg of Hydrazine
  - 227 kg Xenon
  - 0.2 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 10.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 42.4 kg with 0 contingency (note: 2 EP strings required due to throughput required)
- Cost
  - System Cost $31.8M
    - Non-Recurring $11.4M
    - Recurring $20.4M
- Potential Issue: The burn duration is on the order of 811 days, which may be well in excess of the allowable duration

Propulsion

Design – Prop F (Option 3 – EP with Monoprop ACS – 1 MaSMI @ 1000W)

- NOTE: Bipropellant system was assessed, and the deltaV is too high for the system to close
- Electric Propulsion System with blowdown monopropellant ACS
  - 4000 m/s total delta V capability
  - 50 m/s ACS capability
  - 700 kg spacecraft wet mass
  - Isp = 1542 sec, thrust = 68mN
- Hardware - EP
  - Qty (2) JPL MaSMI thrusters (operating at 1000W input, one burning at a time)
  - Qty (1) NGIS 80458-1 Xenon Tank
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80389-1 Diaphragm tank
- Propellant
  - 11.6 kg of Hydrazine
  - 187.1 kg Xenon
  - 0.2 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 10.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 42.4 kg with 0 contingency (note: 2 EP strings required due to throughput required)
- Cost
  - System Cost $31.7M
    - Non-Recurring $11.3M
    - Recurring $20.4M
- Potential Issue: The burn duration is on the order of 418 days, which may be well in excess of the allowable duration

Propulsion

Design – Prop G (Option 3 – EP with Monoprop ACS – 2 MaSMI @ 1000W each)

- NOTE: Bipropellant system was assessed, and the deltaV is too high for the system to close
- Electric Propulsion System with blowdown monopropellant ACS
  - 4000 m/s total delta V capability
  - 50 m/s ACS capability
  - 700 kg spacecraft wet mass
  - Isp = 1542 sec, thrust = 68mN per thruster
- Hardware - EP
  - Qty (2) JPL MaSMI thrusters (operating at 1000W input, both burning at the same time)
  - Qty (1) NGIS 80458-1 Xenon Tank
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80389-1 Diaphragm tank
- Propellant
  - 11.6 kg of Hydrazine
  - 187.1 kg Xenon
  - 0.2 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 10.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 42.4 kg with 0 contingency (note: 2 EP strings required due to throughput required)
- Cost
  - System Cost $31.7M
    - Non-Recurring $11.3M
    - Recurring $20.4M
- Potential Issue: The burn duration is on the order of 210 days

Propulsion

Design – Prop H (Option 3 – EP with Monoprop ACS – 2 MaSMI @ 500W each)

- NOTE: Bipropellant system was assessed, and the deltaV is too high for the system to close
- Electric Propulsion System with blowdown monopropellant ACS
  - 4000 m/s total delta V capability
  - 50 m/s ACS capability
  - 700 kg spacecraft wet mass
  - Isp = 1231 sec, thrust = 68 mN
- Hardware - EP
  - Qty (2) JPL MaSMI thrusters (operating at 500W input, both burning at the same time)
  - Qty (1) NGIS 80458-1 Xenon Tank
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80389-1 Diaphragm tank
- Propellant
  - 11.6 kg of Hydrazine
  - 227 kg Xenon
  - 0.2 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 10.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 42.4 kg with 0 contingency (note: 2 EP strings required due to throughput required)
- Cost
  - System Cost $31.8M
    - Non-Recurring $11.4M
    - Recurring $20.4M
- Potential Issue: The burn duration is on the order of 406 days, which may be well in excess of the allowable duration

Propulsion

Design – Prop I (EP with Monoprop ACS – 5 MaSMI @ 1000W each)

- NOTE: Bipropellant system was assessed, and the deltaV is too high for the system to close
- Electric Propulsion System with blowdown monopropellant ACS
  - 4000 m/s total delta V capability
  - 50 m/s ACS capability
  - 1260 kg spacecraft wet mass
  - Isp = 1542 sec, thrust = 68mN per thruster
- Hardware - EP
  - Qty (5) JPL MaSMI thrusters (operating at 1000W input, both burning at the same time)
  - Qty (3) General Dynamics 220142-1 Xenon Tanks
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80275-1 Diaphragm tank
- Propellant
  - 22.3 kg of Hydrazine
  - 336.8 kg Xenon
  - 0.1 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 12.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 97.2 kg with 0 contingency (note: 2 EP strings required due to throughput required)
- Cost
  - System Cost $61.0M
    - Non-Recurring $17.2M
    - Recurring $43.8M
- Potential Issue: The burn duration is on the order of 150 days

Propulsion

Design – Prop J (Chemical Bipropellant)

- NOTE: Bipropellant system selected as anchor point for TATER model
- Bipropellant system
  - 1000 m/s total delta V capability
  - 50 m/s ACS capability
  - 290 kg spacecraft wet mass
- Hardware
  - Qty (4) MOOG DST-11H 22N thrusters
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80275-1 diaphragm tank – oxidizer
  - Qty (1) NGIS 80259-1 diaphragm tank – fuel
  - Qty (1) NGIS 80386-1 COPV pressurant tank – oxidizer side
  - Qty (1) NGIS 80386-1 COPV pressurant tank – fuel side
- Propellant
  - 50.3 kg of Hydrazine
  - 38.7 kg of NTO
  - 0.5 kg of Helium Pressurant
- Mass
  - Propulsion System Dry Mass 44.4 kg with 0 contingency
- Cost
  - System Cost $24.6M
    - Non-Recurring $10.9M
    - Recurring $13.7M

Propulsion

Design – Prop K (Chemical Monopropellant)

- Note: Monopropellant Blowdown system selected to anchor TATER model
- Monopropellant Blowdown system
  - 1000 m/s total delta V capability
  - 50 m/s ACS capability
  - 360 kg spacecraft wet mass
- Hardware
  - Qty (8) Aerojet/Rocketdyne MR-111C 5N thrusters
  - Qty (2) NGIS 80447-1 diaphragm tank
- Propellant
  - 140.2 kg of Hydrazine Propellant and 0.3 kg of Helium Pressurant
- Mass
  - Propulsion System Dry Mass 33.3 kg with 0 contingency
- Cost
  - System Cost $11.5M
    - Non-Recurring $5.1M
    - Recurring $6.4M

Propulsion

Design – Prop O-A (EP with Monoprop ACS – 2 SPT-140)

- Electric Propulsion System with blowdown monopropellant ACS
  - 2000 m/s total delta V capability
  - 50 m/s ACS capability
  - 15 kg of Xenon for deflection
  - 490 kg spacecraft wet mass
  - Isp = 1780 sec, thrust = 289 mN per thruster
- Hardware - EP
  - Qty (2) SPT-140 thrusters (both burning at the same time)
  - Qty (1) NGIS 80412-1 Xenon Tanks
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80216-1 Diaphragm tank
- Propellant
  - 10.1 kg of Hydrazine
  - 78.5 kg Xenon
  - 0.1 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 9.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 76.3 kg with 0 contingency
- Cost
  - System Cost $29.8M
    - Non-Recurring $13.6M
    - Recurring $16.2M

Propulsion

Design – Prop O-B (EP with Monoprop ACS – 2 SPT-140)

- Electric Propulsion System with blowdown monopropellant ACS
  - 2000 m/s total delta V capability
  - 50 m/s ACS capability
  - 150 kg of Xenon for deflection
  - 700 kg spacecraft wet mass
  - Isp = 1780 sec, thrust = 289 mN per thruster
- Hardware - EP
  - Qty (2) SPT-140 thrusters (both burning at the same time)
  - Qty (2) General Dynamics 220615-1 Xenon Tanks
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80389-1 Diaphragm tank
- Propellant
  - 14.4 kg of Hydrazine
  - 260.6 kg Xenon
  - 0.1 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 10.7 kg with 0 contingency
  - EP Propulsion System Dry Mass 91.8 kg with 0 contingency
- Cost
  - System Cost $31.1M
    - Non-Recurring $13.7M
    - Recurring $17.5M

Propulsion

Design – Prop O-C (EP with Monoprop ACS – 4 SPT-140)

- Electric Propulsion System with blowdown monopropellant ACS
  - 2000 m/s total delta V capability
  - 50 m/s ACS capability
  - 600 kg of Xenon for deflection
  - 1595 kg spacecraft wet mass
  - Isp = 1780 sec, thrust = 289 mN per thruster
- Hardware - EP
  - Qty (4) SPT-140 thrusters (all burning at the same time)
  - Qty (6) General Dynamics 220615-1 Xenon Tanks
- Hardware - Monoprop
  - Qty (8) Aerojet/Rocketdyne MR-103G 1N thrusters
  - Qty (1) NGIS 80486-1 Diaphragm tank
- Propellant
  - 32.8 kg of Hydrazine
  - 887.6 kg Xenon
  - 0.1 kg of Helium Pressurant
- Mass
  - Monoprop Propulsion System Dry Mass 12.9 kg with 0 contingency
  - EP Propulsion System Dry Mass 213.8 kg with 0 contingency
- Cost
  - System Cost $49.8M
    - Non-Recurring $17.0M
    - Recurring $32.8M

Propulsion

Cost Assumptions

- There is not a good cost model for MaSMI in the Team X costing tool
  - Ryan Conversano previously provided a cost estimate of $3.5M for the MaSMI thruster, PPU, gimbal, and XFC
  - With this in mind, the Team X tool reflects that estimated cost for each string, plus tank and tubing costs added in based on Team X tool standard practices
- No spares were taken into account for the costing of hardware
- Phase A-D durations (in months) as follows:
  - A = 9 months, B = 9 months, C = 15 months, D = 16 months
- SPT-140 costing was a bit difficult
  - No cost data available for the SPT XFC, so went with a generic Moog PMA
    - In general the Moog PMA is costly but likely an effective choice
  - No cost data available for the SPT-140 PPU, so went with the cost estimate for the SPT-100 PPU and the mass of the SPT-140 PPU
    - The same is true of the SPT-140 thruster
  - Some tanks have only a recurring cost listed, so the non-recurring cost was assumed to be consistent with those tanks from the same vendor that did list cost
- All other costing is consistent with the Team X cost model

Propulsion

Risks

- The primary risk on many of the SEP missions is the total burn duration
  - Several of the trajectories provided were estimating 1 year of burn duration, but with the SEP thruster at only 500W, the actual duration was over 2 years
  - To provide some additional insight, higher power levels and additional thrusters were reviewed to assess burn duration as well as power demand and impact to propulsion
  - Careful assessment of burn duration is recommended when selecting a specific mission
- Propulsion assessed a series of point designs to provide calibration data points to TATER
  - These points allow for interpolation and extrapolation to some degree, but deviating significantly could violate the capability of the thruster (throughput), the tank sizing (forcing the user to a significantly larger and heavier tank), etc…
  - Care should be taken in interpreting the results, and if a specific point design is desired, propulsion can provide additional TATER calibration points

Propulsion

Additional Comments

- MaSMI has improved Isp and thrust levels as the power is increased (in general)
  - There is an advantage to utilizing a single thruster at higher power from an Isp standpoint, as the propellant load decreases slightly
  - The advantage of utilizing multiple thrusters at lower power levels is the potential for redundancy should an engine failure occur
  - Either of these scenarios could be more advantageous, depending on the mission and risk posture
- The SEP systems included a 50 m/s monopropellant system for ACS and desaturation
  - It is possible for a gimballed SEP thruster to provide desaturation capability, but the monoprop provides some capabilities in terms of relatively rapid reaction times
  - The 50 m/s number is purely an estimate and ACS should be consulted for any point design
- Per Ryan Conversano, MaSMI may be capable of 200 kg of Xenon throughput, though it is rated at 100kg
  - For a few of the cases, two strings were maintained even though one would have been sufficient (specifically, 4 km/s and 700 kg wet mass at 1000W input power). It may be possible to remove the second string to save on dry mass here
  - There are other thruster options on the market and it may be a good trade to explore (e.g. BHT-600, which has a throughput limit similar to MaSMi's rated 100kg)
- In general, for the low mass, low delta V missions, the monopropellant system will be sufficient.
- The low mass, 2000 m/s deltaV missions break the monopropellant system, but a bipropellant system will work
- The low mass, 4000 m/s deltaV missions break chemical propulsion and require EP

Cost Report

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Cost

Disclaimer

The costs presented in this report are ROM estimates, not point estimates or cost commitments.
It is likely that each estimate could range from as much as 20% percent higher to 10% lower.
The costs presented are based on Pre-Phase A design information, which is subject to change.

Cost

Cost Requirements

- Constant/Real Year Dollars: FY 2025
- Cost Target: $500M
- Class C
- Schedule

Phase A	Phase B	Phase C	Phase D	Phase E	Phase F
9 mo.	9 mo.	20 mo.	16 mo.	36 mo.	4 mo.

- 30% Phase A-D reserves
- 15% Phase E-F reserves
- Assumed 85% Phase A-D Costs
- Assumed 15% Phase E-F Costs
- No EPO costs included
- No reserves added for LV

Cost

Spacecraft Cost Options

Spacecraft Bus	Cost		
Alias	1st Unit $M (FY2025)	Nth Unit $M (FY2025)	Learning Curve Approach
Bus 1 - Flyby Monoprop [M]	$58.0	$23.2	Nth unit discount
Bus 2 - Biprop [M]	$117.3	$46.9	Nth unit discount
Bus 2-E - SEP [M]	$98.3	$39.3	Nth unit discount
Bus 3 - Hi ΔV SEP [M]	$115.8	$46.3	Nth unit discount
Bus 4 - Biprop [M]	$149.0	$59.6	Nth unit discount
Bus 4-E - SEP [M]	$104.3	$41.7	Nth unit discount
Bus 5-S - High ΔV BigSat - SPT-140	$153.7	$61.5	Nth unit discount
Bus 6 - Biprop [M]	$155.8	$62.3	Nth unit discount
Bus 6-E - SEP [M]	$101.6	$40.6	Nth unit discount
Deployed Lander Bus	$18.9	$7.6	Nth unit discount
ESPA Ring (6-port)	$6.0	$6.0	No discount
Dual Payload Adapter	$6.0	$6.0	No discount
Bus 9 - Mothership [M]	$86.3	$34.5	Nth unit discount
12U Cubesat Bus with Prop	$13.8	$5.5	Nth unit discount
12U Cubesat Dispenser	$0.2	$0.2	No discount

Systems Note: The Option 8 spacecraft were all “Bus 1 – Flyby Monoprop [M]”, since the requirements were the same as in Option 1, and they are all assumed to be identical, so all S/C after the first are costed at the Nth unit cost.

Spacecraft Bus	Cost		
Alias	1st Unit $M (FY2025)	Nth Unit $M (FY2025)	Learning Curve Approach
Bus 10-MP - Monoprop [M]	$99.2	$39.7	Nth unit discount
Bus 10-BP - Biprop [M]	$102.6	$41.0	Nth unit discount
Bus 11 - Biprop [M]	$141.5	$56.6	Nth unit discount
Bus 11-E - SEP [M]	$105.6	$42.2	Nth unit discount
Bus 12 - Biprop [M]	$149.5	$59.8	Nth unit discount
Bus 12-E - SEP [M]	$102.8	$41.1	Nth unit discount
Bus 13-DI - Monoprop [M]	$85.9	$34.4	Nth unit discount
Impactor 13-DI - Monoprop [M]	$41.9	$16.8	Nth unit discount
Dumb Mass 13-DI	$0.1	$0.1	No discount
13-DART - Impactor Mothership	$90.0	$36.0	Nth unit discount
13-DART - Flyby SmallSat	$13.8	$5.5	Nth unit discount
13-DART - Dumb Mass	$0.1	$0.0	Nth unit discount
Bus 14 - SEP [M]	$119.1	$47.7	Nth unit discount
Impactor 14 - Biprop [M]	$100.5	$40.2	Nth unit discount
Dumb Mass 14	$0.1	$0.1	No discount
Bus 15-A-M - SEP [M] - UltraFlex	$115.6	$46.2	Nth unit discount
Bus 15-B-M [M] - UltraFlex, 2M	$139.8	$55.9	Nth unit discount
Bus 15-C-M [M] - UltraFlex, 6M	$217.7	$87.1	Nth unit discount
Bus 15-A-S - SEP [M] - UltraFlex	$188.1	$75.2	Nth unit discount
Bus 15-B-S - SEP [M] - UltraFlex	$198.6	$79.4	Nth unit discount
Bus 15-C-S - SEP [M] - UltraFlex	$245.7	$98.3	Nth unit discount
Bus 16 - SEP [M] - UltraFlex, 2M	$127.0	$50.8	Nth unit discount
16 - Dumb Mass	$0.1	$0.1	No discount
Bus 17 - SEP [M] - UltraFlex, 2M	$133.1	$53.2	Nth unit discount

Cost

Cost Assumptions

- WBS05 (Payload) taken from NICM
  - A 5% WBS 5.01 and 5.02 “tax” is added only in cases where the total number of instrument types (across all flight elements) is > qty 4 or the total instrument cost > $40M

- WBS07 (MOS), and WBS09 (GDS) were estimated during this study.
- Remaining WBS wrap factors
  - WBS01: Project Management: 2.9%
  - WBS02: Systems Engineering: 4.7%
  - WBS03: S&MA: 2.8%
  - WBS04: Science/Technology: 3.3%
  - WBS07: Mission Operations: 10.2%
  - WBS 09: Ground Data Systems: 4.2%
  - WBS10: ATLO: 3.9%

WBS 06 (S/C Bus)

- SC Bus cost model is a regression (N=15) on small/medium class, planetary and astrophysics missions (excluding landers and rovers).
  - R^2-adj: 51%, p-value for both coefficients < 0.05, $\mathrm{P}(F\ \mathrm{Stat}) < 0.0001$):

  $$\mathrm{LN}(\mathrm{SC\ Cost}) = -0.303 + \mathrm{LN}(0.875 \cdot (\mathrm{CBE\ Dry\ Bus\ Mass} + 20\%\ \mathrm{contingency}))$$

  - 20% contingency (Mass growth allowance) was selected to better predict actual mass.
  - Bootstrap Error – 51%
  - Leave One Out (LOO) Mean Absolute Error – 43%
- For all options, with low mass solar arrays (UltraFlex or ROSA type) we applied a cost upper.
  - The bus cost regression source data did not include any designs with low-mass (UltraFlex or ROSA-type) arrays, so we added a cost upper for the low-mass arrays. The cost upper is computed specifically for the mass of the low-mass arrays in the design, and is based on mass/cost relationships derived from the power subsystem designs (for both low-mass and traditional rigid arrays) performed in this study.

  - Since the cost data in the bus cost regression included a diversity of propulsion types (SEP, monoprop, biprop), it was assumed that the cost model already captures the result of each mission's propulsion trades, which are driven by both mass and cost considerations. We therefore cannot use our cost model to actually estimate the cost effect of propulsion trades. However, it should be noted that, looking at a plot of propulsion mass and cost, the actual variance in propulsion system designs off of their "expected" cost (based on propulsion dry mass alone) is generally only a few million dollars (at most $8M), whereas overall bus cost changes due to changes in the propulsion system are usually higher (~10's of millions). Therefore it should generally be the case that the lowest-mass S/C design is the lowest-cost as well, but the magnitude of the savings is subject to uncertainty in the propulsion cost differences.

Cost

Cost Assumptions – Cost Model

- SC Bus cost model is a regression (N=15) on small/medium class, planetary and astrophysics missions (excluding landers and rovers).

Contour
Dawn
Deep Impact
DS1
Galex
Genesis
GRAIL
Kepler
LADEE
Mars Odyssey
Nustar
Stardust
Study 1110 Bolton Option 2
Study 1179 Sotin
WISE

Cost

Similar Mission Costs Comparisons

	$K FY25	
Mission	WBS 06	Total Mission
DART	$110,039	$339,948
Dawn	$234,779	$692,620
Deep Impact	$194,730	$527,028
Hayabusa2	$127,000	
Janus	$59,565	
Osiris Rex	$388,542	$1,256,383
Psyche	$350,307	$846,984

Cost

Total Cost

All Costs in $M FY 2025

WBS	Description	Option 1 Flyby Recon	Option 2 Rendezvous	Option 2-E Rendez SEP	Option 3 Rendez high ΔV	Option 4 Rendez Radar-Lidar	Option 4-E Rendez SEP Radar-Lidar	Option 5 Rendez high ΔV BigSat
1	Project Management	4.9	8.7	7.4	8.6	15.2	12.1	11.2
2	Systems Engineering	8.1	14.2	12.1	14.1	24.9	19.9	18.3
3	Safety & Mission Assurance	4.8	8.5	7.2	8.4	14.8	11.8	10.9
4	Science/Technology	5.8	10.2	8.7	10.1	17.8	14.2	13.1
5	Payload	13.9	9.7	9.7	9.7	72.8	72.8	9.7
6	Flight System	58.0	117.3	98.3	115.8	149.0	104.3	153.7
7	MOS	17.6	31.1	26.5	30.8	54.4	43.4	40.1
8	Launch Vehicle Services	115.0	115.0	115.0	115.0	115.0	115.0	115.0
9	GDS	7.2	12.7	10.8	12.5	22.1	17.7	16.3
10	ATLO	6.7	11.8	10.1	11.7	20.6	16.5	15.2
	Total (w/o Reserves)	**241.9**	**339.2**	**305.8**	**336.6**	**506.5**	**427.6**	**403.5**
	Reserve Cost	**35.2**	**62.2**	**52.9**	**61.5**	**108.7**	**86.8**	**80.1**
	Total (w/ Reserves)	**277.1**	**401.4**	**358.7**	**398.1**	**615.2**	**514.4**	**483.6**

Cost

Total Cost

All Costs in $M FY 2025

WBS	Description	Option 6 Rendez 2FE	Option 6-E Rendez SEP 2FE	Option 8-1 Tour Multiple (1xB)	Option 8-2 Tour Multiple (2xB)	Option 8-3 Tour Multiple (3xB)	Option 9 Tour CubeSats
1	Project Management	16.1	12.3	7.1	9.0	10.8	9.7
2	Systems Engineering	26.3	20.2	11.7	14.7	17.7	15.9
3	Safety & Mission Assurance	15.6	12.0	7.0	8.7	10.5	9.4
4	Science/Technology	18.8	14.5	8.4	10.5	12.6	11.3
5	Payload	59.7	59.7	17.3	20.6	24.0	23.8
6	Flight System	174.8	120.6	87.2	110.3	133.5	117.6
7	MOS	57.5	44.2	25.6	32.1	38.6	34.7
8	Launch Vehicle Services	115.0	115.0	115.0	115.0	115.0	115.0
9	GDS	23.4	18.0	10.4	13.1	15.7	14.1
10	ATLO	21.8	16.8	9.7	12.2	14.7	13.2
	Total (w/o Reserves)	**529.0**	**433.3**	**299.4**	**346.2**	**393.1**	**364.7**
	Reserve Cost	**114.9**	**88.3**	**51.2**	**64.2**	**77.2**	**69.3**
	Total (w/ Reserves)	**643.9**	**521.7**	**350.5**	**410.4**	**470.3**	**434.0**

Cost

Total Cost

All Costs in $M FY 2025

WBS	Description	Option 10-MP Intercept Monoprop	Option 10-BP Intercept Biprop	Option 11 Rendezvous wNED	Option 11-E Rendezvous SEP wNED	Option 12 Rendezvous wNED (2E)	Option 12-E Rendezvous SEP wNED (2E)	Option 13-DI Kinetic Impact (DI)	Option 13-DART Kinetic Impact (DART)
1	Project Management	9.3	9.6	10.4	7.9	12.6	9.4	9.6	7.7
2	Systems Engineering	15.3	15.7	17.0	12.9	20.6	15.4	15.8	12.7
3	Safety & Mission Assurance	9.1	9.3	10.1	7.7	12.3	9.2	9.4	7.5
4	Science/Technology	11.0	11.2	12.1	9.3	14.8	11.0	11.3	9.1
5	Payload	37.4	37.4	9.8	9.8	34.5	34.5	12.6	8.7
6	Flight System	99.2	102.6	141.5	105.6	149.5	102.8	128.0	104.2
7	MOS	33.5	34.3	37.1	28.3	45.1	33.7	34.5	27.7
8	Launch Vehicle Services	115.0	115.0	115.0	115.0	115.0	115.0	115.0	115.0
9	GDS	13.6	14.0	15.1	11.5	18.4	13.7	14.0	11.3
10	ATLO	12.7	13.0	14.1	10.7	17.1	12.8	13.1	10.5
	Total (w/o Reserves)	**356.1**	**362.1**	**382.2**	**318.7**	**439.8**	**357.5**	**363.2**	**314.3**
	Reserve Cost	**66.9**	**68.6**	**74.1**	**56.5**	**90.1**	**67.3**	**68.9**	**55.3**
	Total (w/ Reserves)	**423.0**	**430.7**	**456.3**	**375.2**	**529.9**	**424.8**	**432.0**	**369.6**

Cost

Total Cost

TEAM X Jet Propulsion Laboratory

All Costs in $M FY 2025

WBS	Description	Option 14 Kinetic Impact (SEP obs)	Option 15-A-M Ion Beam (MaSMi)	Option 15-B-M Ion Beam (MaSMi)	Option 15-C-M Ion Beam (MaSMi)	Option 15-A-S Ion Beam (SPT-140)	Option 15-B-S Ion Beam (SPT-140)	Option 15-C-S Ion Beam (SPT-140)
1	Project Management	15.9	8.2	9.9	15.2	13.2	13.9	17.1
2	Systems Engineering	26.1	13.4	16.1	24.9	21.6	22.7	28.0
3	Safety & Mission Assurance	15.5	8.0	9.6	14.8	12.8	13.5	16.7
4	Science/Technology	18.6	9.6	11.5	17.8	15.4	16.3	20.0
5	Payload	12.6	4.2	4.2	4.2	4.2	4.2	4.2
6	Flight System	219.8	115.6	139.8	217.7	188.1	198.6	245.7
7	MOS	57.0	29.4	35.3	54.4	47.2	49.7	61.3
8	Launch Vehicle Services	115.0	115.0	115.0	115.0	115.0	115.0	115.0
9	GDS	23.2	12.0	14.4	22.2	19.2	20.2	24.9
10	ATLO	21.6	11.1	13.4	20.7	17.9	18.9	23.3
	Total (w/o Reserves)	**525.3**	**326.5**	**369.2**	**506.9**	**454.4**	**473.1**	**556.2**
	Reserve Cost	**113.8**	**58.7**	**70.5**	**108.7**	**94.2**	**99.4**	**122.4**
	Total (w/ Reserves)	**639.1**	**385.1**	**439.8**	**615.6**	**548.6**	**572.5**	**678.7**

Cost

Total Cost

All Costs in $M FY 2025

WBS	Description	Option 16 Gravity Tractor	Option 17 Ion Beam & Gravity Tractor
1	Project Management	11.1	11.5
2	Systems Engineering	18.1	18.8
3	Safety & Mission Assurance	10.8	11.2
4	Science/Technology	13.0	13.4
5	Payload	34.4	34.4
6	Flight System	127.1	133.1
7	MOS	39.6	41.1
8	Launch Vehicle Services	115.0	115.0
9	GDS	16.1	16.7
10	ATLO	15.0	15.6
	Total (w/o Reserves)	400.2	410.7
	Reserve Cost	79.1	82.0
	Total (w/ Reserves)	479.4	492.7

F TIME-PHASED COST PROFILES

Time-phased costs were estimated by using the typical phase durations shown in Table F-1 and tabulated in Table F-2. Project cost estimates from the estimates in Appendix E were modified for the best LV choice and the rendezvous missions phase E costs were stretched by a year and added one year's worth of cost ($22M) to better reflect the complexity of these missions, leading to the adjusted costs in Table 4-4. These costs were then tabulated on a monthly basis and used to create the graphics shown in Figure F-1 and Figure F-2. Figure F-1 shows the cumulative cost of the least and most-expensive rendezvous missions studied, while Figure F-2 shows the cumulative least and most-expensive Flyby missions, which tend to be less expensive than the rendezvous missions. Note that the mapping of these mission types to the PD characterization/mitigation objectives is described in §4.2.

Table F-2. Relative cost distribution by project phase

	Cost Profiles					
	Phase A	Phase B	Phase C	Phase D	Phase E/F	Totals
Flyby missions:						
Durations (months)	6	12	20	16	36	90
Fraction of cost	5.00%	20.00%	30.00%	35.00%	10.00%	100.00%
Rendezvous Missions:						
Durations (months)	6	**15**	20	16	**48**	105
Fraction of cost	5.00%	20.00%	25.00%	30.00%	20.00%	100.00%

Table F-3. Time-phased costs for each option

Trade Study Mission Number	Mission Description	Mission Type	Est Cost, Best LV ($M FY 2025)	Mission $M per phase A	B	C	D	E/F
1	Flyby Recon	Flyby	199	9.95	39.8	59.7	69.65	19.9
2	Rendezvous Low ΔV (2 km/s)	Rendezvous	352	17.5	70	87.5	105	70
2-E	Rendezvous Solar Electric Propulsion (SEP) Low ΔV	Rendezvous	310	15.4	61.6	77	92.4	61.6
3	Rendezvous High ΔV (4 km/s)	Rendezvous	349	17.35	69.4	86.75	104.1	69.4
4	Rendezvous with Expanded Payload	Rendezvous	597	29.75	119	148.75	178.5	119
4-E	Rendezvous SEP with Expanded Payload	Rendezvous	465	23.15	92.6	115.75	138.9	92.6
5	Rendezvous High ΔV with Big Sat Components	Rendezvous	434	21.6	86.4	108	129.6	86.4
6	Rendezvous with Deployed Lander/Rover	Rendezvous	626	31.2	124.8	156	187.2	124.8
6-E	Rendezvous SEP with Deployed Lander/Rover	Rendezvous	472	23.5	94	117.5	141	94
7	Flyby Tour with a single spacecraft to 4 NEOs	Flyby	206	10.3	41.2	61.8	72.1	20.6
8-1	Flyby Tour with 2 spacecraft to 4 NEOs	Flyby	279	13.95	55.8	83.7	97.65	27.9
8-2	Flyby Tour with 3 spacecraft to 4 NEOs	Flyby	339	16.95	67.8	101.7	118.65	33.9
8-3	Flyby Tour with 4 spacecraft to 4 NEOs	Flyby	430	21.5	86	129	150.5	43
9	Flyby Tour Mothership with 4 CubeSats (1/per target)	Flyby	363	18.15	72.6	108.9	127.05	36.3
10-MP	Intercept (Mono-Prop) w/ NED Simulator	Flyby	352	17.6	70.4	105.6	123.2	35.2
10-BP	Intercept (Bi-Prop) w/ NED Simulator	Flyby	359	17.95	71.8	107.7	125.65	35.9
11	Rendezvous with NED Simulator	Rendezvous	438	21.8	87.2	109	130.8	87.2
11-E	Rendezvous SEP with NED Simulator	Rendezvous	326	16.2	64.8	81	97.2	64.8
12	Rendezvous with Deployed NED Simulator (2 elements)	Rendezvous	512	25.5	102	127.5	153	102
12-E	Rendezvous SEP with Deployed NED Simulator (2 elements)	Rendezvous	376	18.7	74.8	93.5	112.2	74.8
13	Kinetic Impact w/ 2 elements (like Deep Impact)	Flyby	361	18.05	72.2	108.3	126.35	36.1
13-DART	Kinetic Impact w/ 2 elements (like DART w/CubeSat)	Flyby	298	14.9	59.6	89.4	104.3	29.8
14	Kinetic Impact w/ 2 elements (Rendezvous SC + Impactor SC)	Rendezvous	621	30.95	123.8	154.75	185.7	123.8
15 A-M	Ion Beam w/MaSMi Thruster (32 kg Xe)	Rendezvous	336	16.7	66.8	83.5	100.2	66.8
15 B-M	Ion Beam w/MaSMi Thruster (150 kg Xe)	Rendezvous	390	19.4	77.6	97	116.4	77.6
15 C-M	Ion Beam w/MaSMi Thruster (600 kg Xe)	Rendezvous	598	29.8	119.2	149	178.8	119.2
15 A-S	Ion Beam w/SPT-140 Thruster (32 kg Xe)	Rendezvous	499	24.85	99.4	124.25	149.1	99.4
15 B-S	Ion Beam w/SPT-140 Thruster (150 kg Xe)	Rendezvous	555	27.65	110.6	138.25	165.9	110.6
15 C-S	Ion Beam w/SPT-140 Thruster (570 kg Xe)	Rendezvous	666	33.2	132.8	166	199.2	132.8
16	Gravity Tractor	Rendezvous	430	22.4	89.6	112	134.4	89.6
17	Ion Beam & Gravity Tractor	Rendezvous	443	22.05	88.2	110.25	132.3	88.2
	Rendezvous Hybrid	ave of mid-size cases	472.5	24.975	99.9	124.875	149.85	99.9
		add LIDAR	13					
		add IR spec	6					
		other GN&C upgrades	10	Median Phase E cost for rendezvous				89.6
		Total:	499.5	Cost per year				22.4

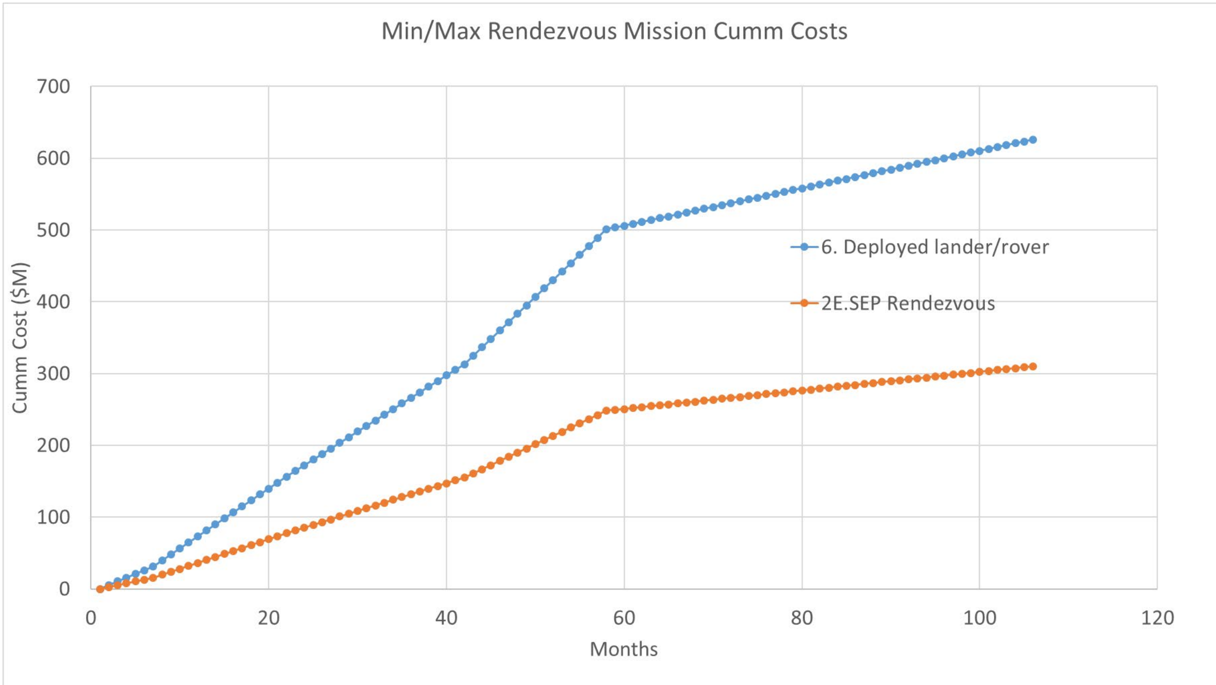

Figure F-8. Range of rendezvous missions cumulative cost profiles

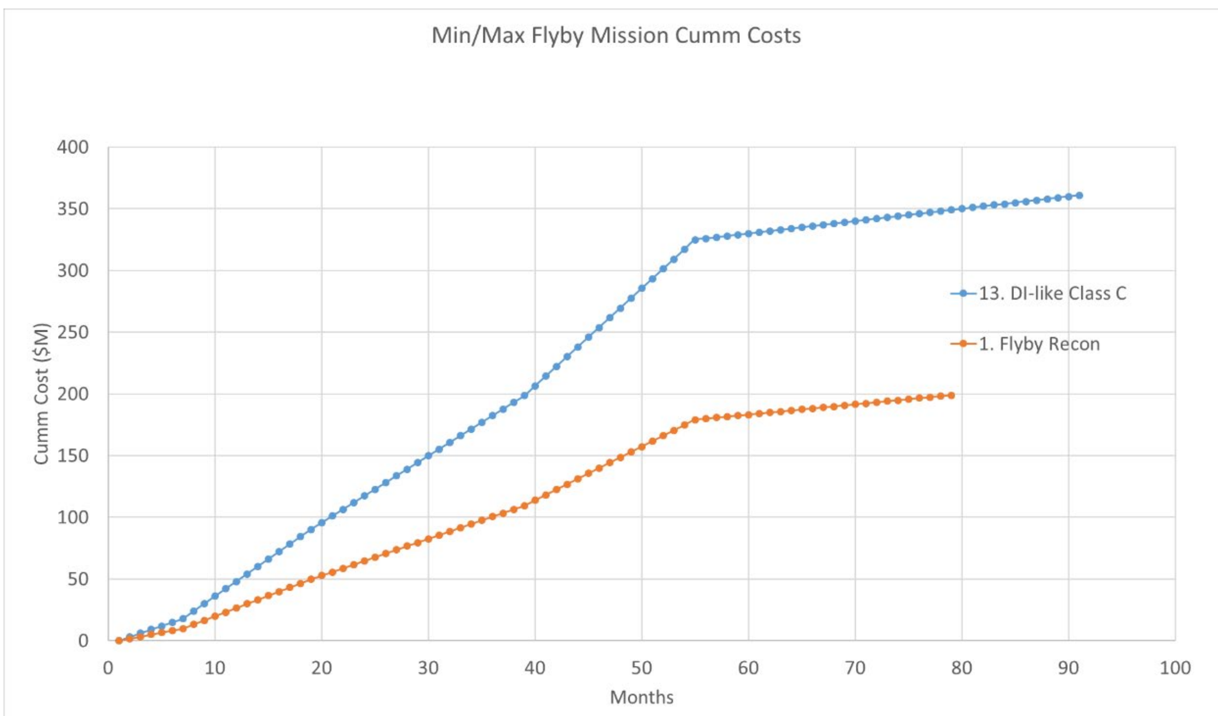

Figure F-9. Range of flyby missions cumulative cost profiles.

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