Multi-Organization – Multi-Discipline Effort Developing a Mitigation Concept for Planetary Defense

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Abstract—**There have been significant recent efforts in addressing mitigation approaches to neutralize Potentially Hazardous Asteroids (PHA). One such research effort was performed in 2015 by an integrated, inter-disciplinary team of asteroid scientists, energy deposition modeling scientists, payload engineers, orbital dynamicist engineers, spacecraft discipline engineers, and systems / architecture engineers from NASA's Goddard Space Flight Center (GSFC) and the Department of Energy (DoE) / National Nuclear Security Administration (NNSA) laboratories (Los Alamos National Laboratory (LANL), Lawrence Livermore National Laboratories (LLNL) and Sandia National Laboratories). The study team collaborated with GSFC's Integrated Design Center's Mission Design Lab (MDL) which engaged a team of GSFC flight hardware discipline engineers to work with GSFC, LANL, and LLNL Near-Earth Asteroid (NEA)-related subject matter experts during a one-week intensive concept formulation study in an integrated concurrent engineering environment. This team has analyzed the first of several distinct study cases for a multi-year NASA research grant. This Case 1 study references the NEA named Bennu as the notional target due to the availability of a very detailed Design Reference Asteroid (DRA) model for its orbit and physical characteristics (courtesy of the Origins, Spectral Interpretation, Resource Identification, Security-Regolith Explorer [OSIRIS-REx] mission team). The research involved the formulation and optimization of spacecraft trajectories to intercept Bennu, overall mission and architecture concepts, and high-fidelity modeling of both kinetic impact (spacecraft collision to change a NEA's momentum and orbit) and nuclear detonation effects on Bennu, for purposes of deflecting Bennu.**

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1. INTRODUCTION

This paper is intended to be one of a set of papers to be produced by an integrated study team comprised of NASA/ GSFC, DoE/NNSA LANL, and LLNL, of work conducted in the October 2015 timeframe. We refer to results of a parallel paper on the uncertainty caused by physical properties of asteroids and how they respond to a deflection impulse.

The focus of this paper is on the Delivery Segment and Space Segment portions of a larger NEA Mitigation Architecture as depicted in Figure 1. This activity produced a detailed concept of a multi-purpose spacecraft to carry out planetary defense mission objectives. While the Bennu scenario was utilized as a point of departure for analysis purposes, the spacecraft concept is intended to be applicable to a broader range of possible hazardous NEA scenarios. The MDL systems concept development study objectives included formulating a spacecraft concept capable of intercepting an NEA, functioning as either a Kinetic Impactor (KI) or a Nuclear Energy Device (NED) delivery system. The assumed target for this study is the Potentially Hazardous [to Earth] Asteroid (PHA) known as 101955 Bennu (1999 RQ36), which is the destination for NASA's OSIRIS-REx asteroid sample return mission (launched in September 2016). [1]

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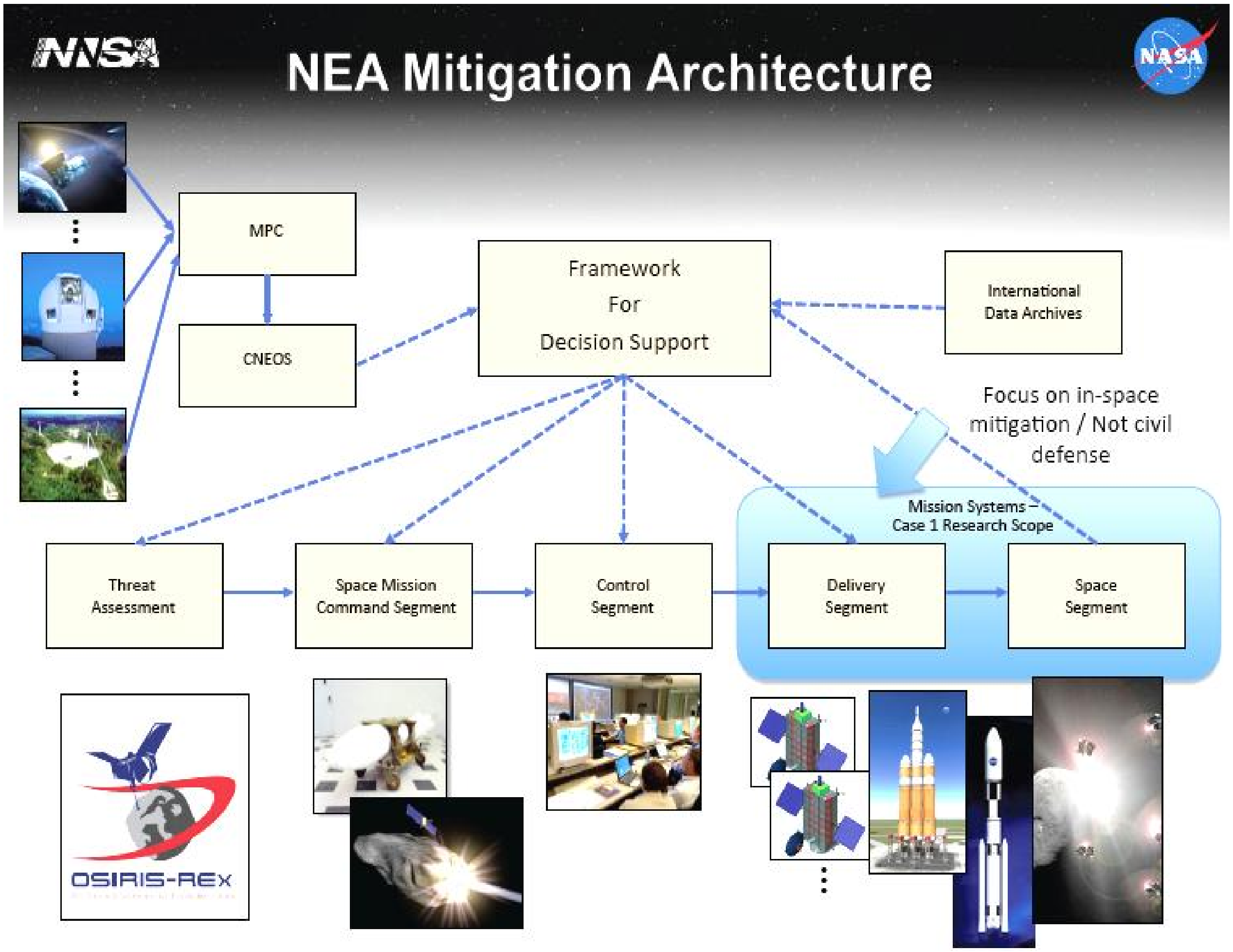


Figure 1. NEA Mitigation Architecture

The spacecraft's performance as a KI was modeled in the MDL, with the goal that it also could function as a NED delivery system (with the NED installed in the spacecraft). Accordingly, the spacecraft would be capable of housing the NED and transporting it to the targeted NEA with the goal of little or no changes to the spacecraft. Thus, the resulting higher-level objective for the MDL study was to develop a concept for a multi-functional spacecraft (operating alone or as part of a campaign including multiple spacecraft), deliverable to a target NEA by a variety of current and planned US launch vehicles or other launch delivery systems. [2]

The spacecraft concept and corresponding mission profile developed during the MDL study was named HAMMER (Hypervelocity Asteroid Mitigation Mission for Emergency Response).

2. Concept Formulation Constraints and Drivers

Class A missions are extremely critical operational systems where all practical measures are taken to ensure mission success. They have the highest cost, are of high complexity, and the longest mission life with tight launch constraints. Contract types for these systems are typically cost plus because of the substantial development risk and resultant oversight activities. These missions are achieved by strict implementation of mission assurance processes derived through proven best practices to achieve mission success over the desired life of the system. All practical measures, to include full incorporation of all specifications/standards contract requirements with little to no tailoring, are taken to achieve mission success for such missions. Class A missions are long life, (nominally greater than 5 years) and represent large national investments for critical applications.[3]

NASA Class A missions are represented by flagship missions such as the Hubble Space Telescope, Cassini, and the Jupiter Icy Moon Orbiter (JIMO). National Security Space (NSS) Class A missions include the Global Positioning System satellite and military communication satellite systems to include Milstar. [4]

Additionally, the selection of the largest commercially available launch vehicle was included as part of the mission objectives, as shown in Table 1.

Table 1. Case Study 1 Summary of Mission Objectives and Mission Goals / Requirements

Mission Objectives	Mission Goals / Requirements	Comments
Track, intercept, divert a PHA using a KI as the first option if viable	Deliver as much mass as possible, imparting max KE, at closing velocity up to 10 km/s	May challenge spacecraft control authority
Utilize largest commercially available US launch vehicle	Absolute navigation to ±5 km	
Consider use of depleted U238 ~ density 19.1 gm / cm3, in order to use full capability (GLOW [Gross Liftoff Weight]) of the Delta IV heavy class launch vehicle of ~8870 kg payload	Transition to relative nav ~I-1 day	Overall response time due to round trip time delay becomes a limiting factor
Maintain Operational readiness	Maintain guidance until impact with max telemetry	Overall mission duration as well as mission lead time becomes a limiting factor
Robust and resilient architecture	Provide diagnostic telemetry for off-nominal events	
Class A+ reliability for deployed system	Fail operational during mission critical phases	Overall mitigation FOM (figure of merit) needs consideration
Ready accommodation of KI or NED payload	Fail safe during non-critical phases	
Max Range: 2.4 AU from Earth; 1.4 AU from Sun		

3. TOP LEVEL MITIGATION FUNCTIONS (POST DETECTION)

The major functional elements required for the timely mitigation of PHA employ a feedback loop control system. Once the object has been detected on a hazardous course with our planet then actionable object information will be needed both in sufficient detail and in a timely manner in order for a control system to effect the appropriate mitigation action(s). This control system will utilize the necessary in place segments previously depicted as part of the NEA Mitigation Architecture to include a ground segment, launch segment, space segment, and communications segment. For simplification purposes, only the key segments are depicted in Figure 2. There are a number of characteristics which become drivers for this particular control loop: a) the total round trip delay / propagation time of the communications segment (ranging from 1 to possible 3 AU); b) the availability (including any delays such as the quantity and availability) of the ground-based delivery system; c) the end-to-end control loop processing time (or loop response time).

This system is likely to include potential manufacturing, launch vehicles, launch processing, spacecraft, mitigation payloads (to be paired with spacecraft), spacecraft command and communications systems, payload command and communications systems, as well as PHA detection, tracking, navigation and guidance functions. As shown, all of these functions and elements will need to work in coordination with each other in a timely, reliable, and robust manner.

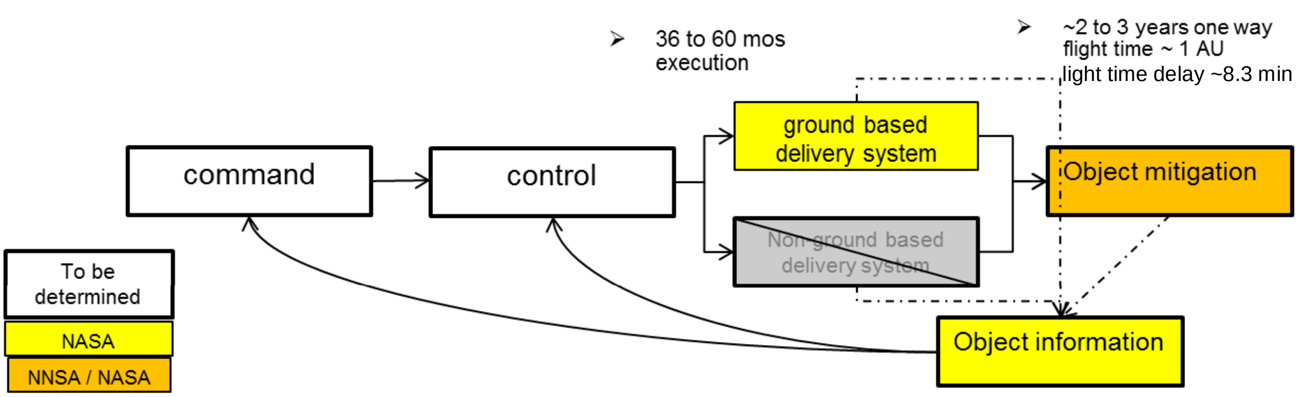


Figure 2. PHA Mitigation Control Loop Concept[5]

4. Mission Timeline

The reference timeline, see Figure 3, is a launch on January 1, 2023, with a 1-week on orbit check-out and an expected cruise phase of 740 days. The mitigation / encounter or impact phase commences roughly one day before the January 10, 2025 impact (I). Autonomous operations begin at that point. Target acquisition engages at around 9,000 kilometers from the encounter point with on-board targeting acquisition strategies such as infrared or mass centroid detection. At I-1 hour (around 36,000 kilometers) commit target processing and on-board optical imaging commences; at I-10 minutes (approximately 6,000 kilometers from engagement) velocity is approximately 4.48 km/s with formulation concept upper bound goal established at 10 km/s.

5. Autonomous Navigation System

Prior to the terminal navigation timeline phase the autonomous navigation system (ANS) is in update mode with ground segment systems. It is then corrected and refreshed with information to allow autonomous navigation to take place on board within spacecraft avionics. This function guides and navigates the spacecraft in the final terminal impact sequence towards the asteroid. Figure 3 shows the closed-loop functional block diagram of this ANS process.

Figure 4 depicts the ANS subsystem of the terminal approach phase beginning at impact minus 2 hours (I-120 minutes). It depicts the linear covariance analysis, the Monte Carlo error analysis, and utilizes a sequential Kalman filter with observations derived from the asteroid centroid location sensor. It solves for the initial position and velocity of the spacecraft with respect to the target asteroid.

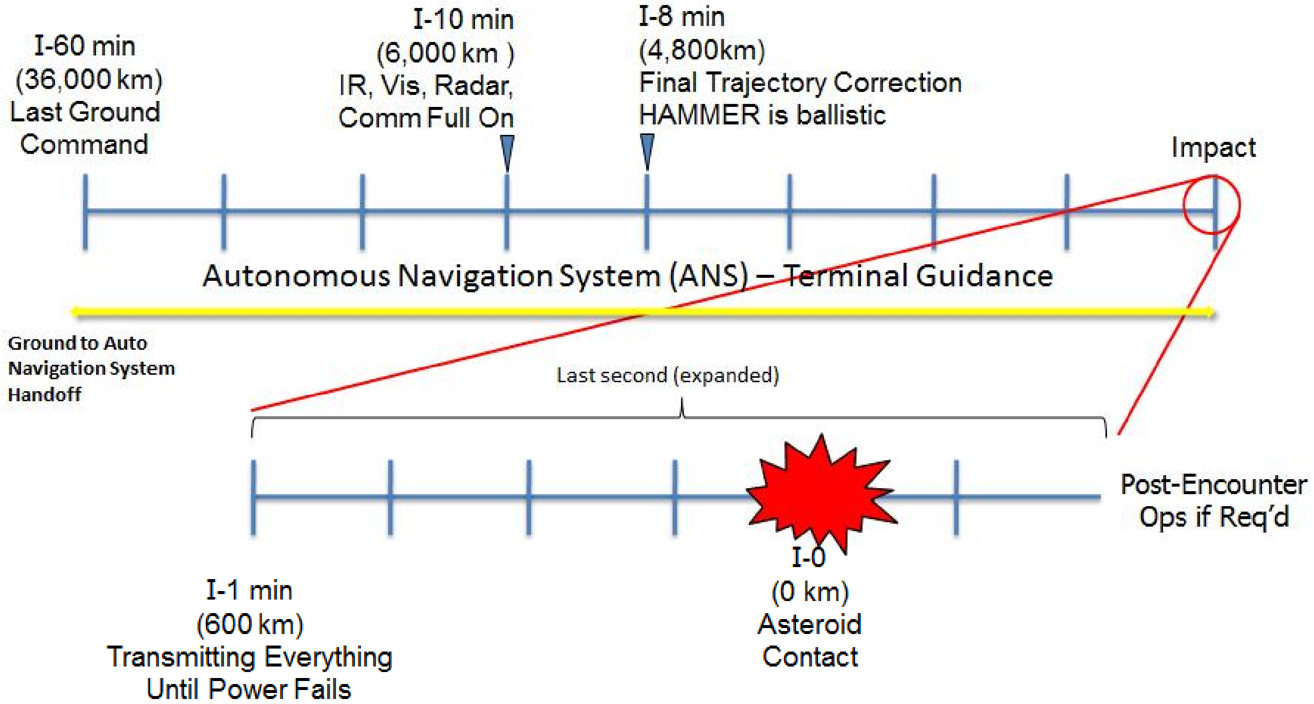


Figure 3. Case 1 Study Mission and Expanded Impact Timelines

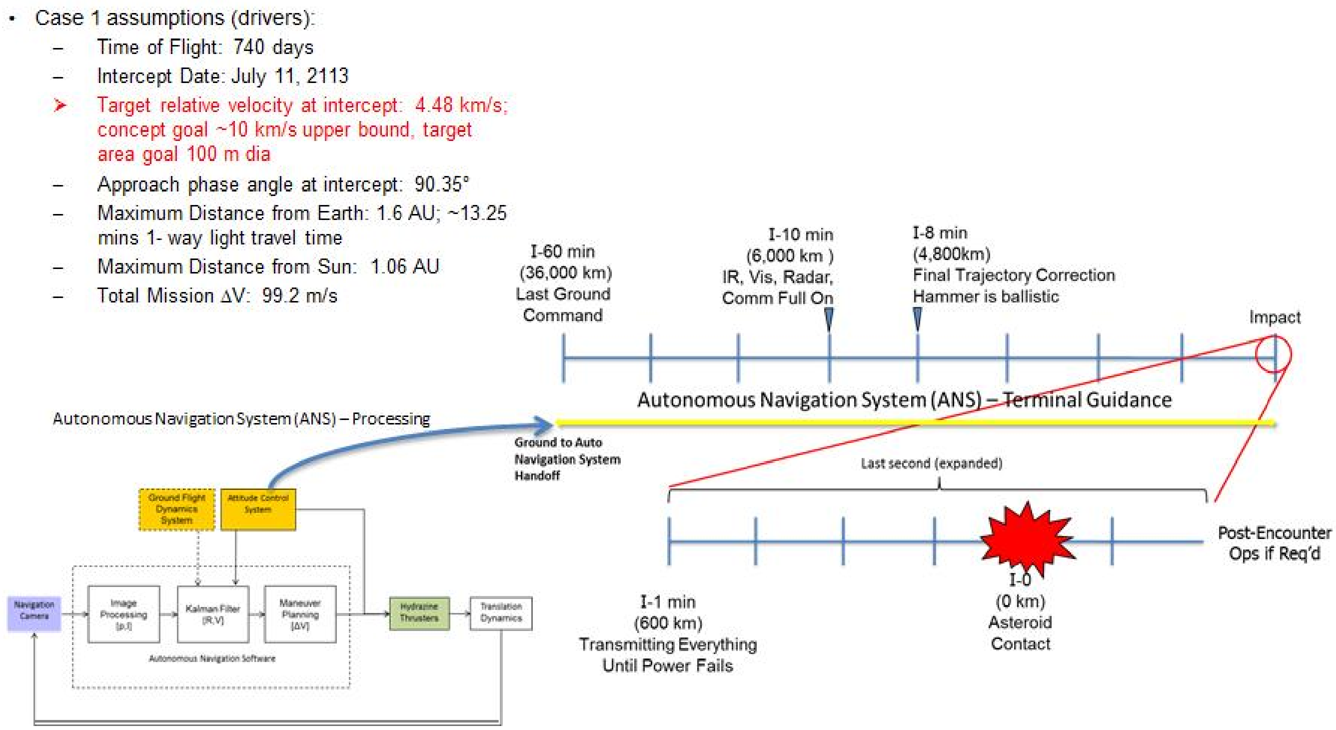


Figure 4. Autonomous Navigation Processing Subsystem

6. ORBITAL PHYSICS

The next three figures (Figures 5, 6 and 7) depict the heliocentric orbit trajectories for the departure, arrival, and final encounter terminal phases. These trajectories determine the environments in which the spacecraft and payload must operate during the launch, transit, and terminal mitigation encounter phases with the asteroid.

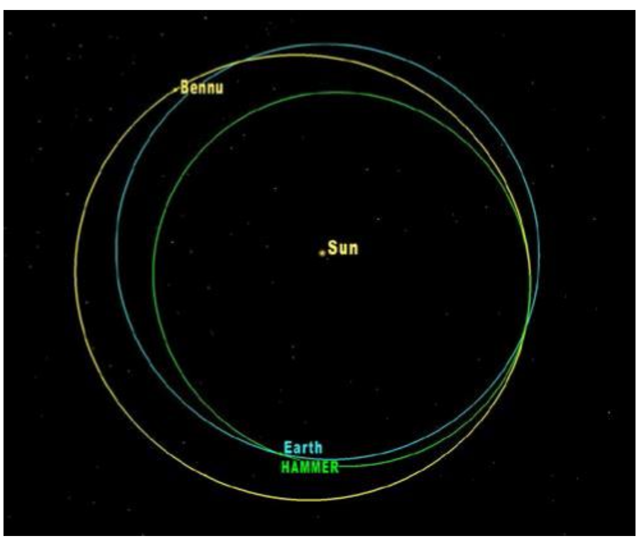


Figure 5. Heliocentric Trajectory for the Earth Departure Phase

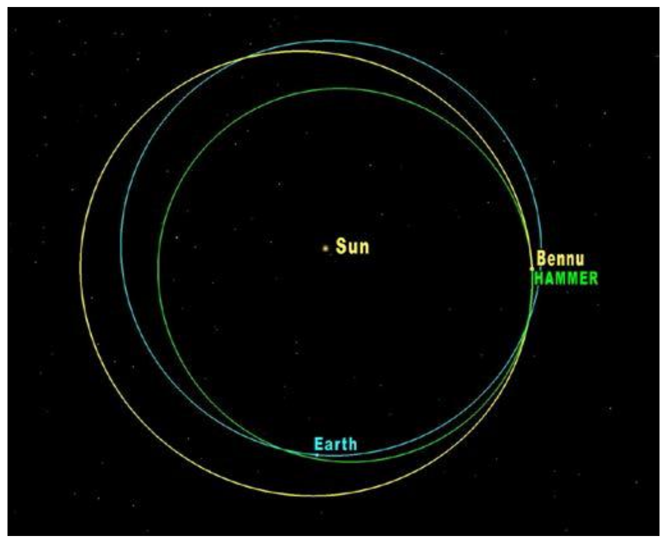


Figure 6. Heliocentric Trajectory for the Arrival Phase at 101995 Bennu

NOTE: In Figure 6, the asteroid is only in partial view with respect to the spacecraft in the foreground.

These three figures were all simulated using the orbit determination toolbox (ODTBX) software analysis which is an advanced navigation and mission simulation and analysis tool used for concept exploration, early concept formulation, and/or rapid design center formulation environments. This tool was developed by the Navigation and Mission Design Branch at the NASA GSFC.

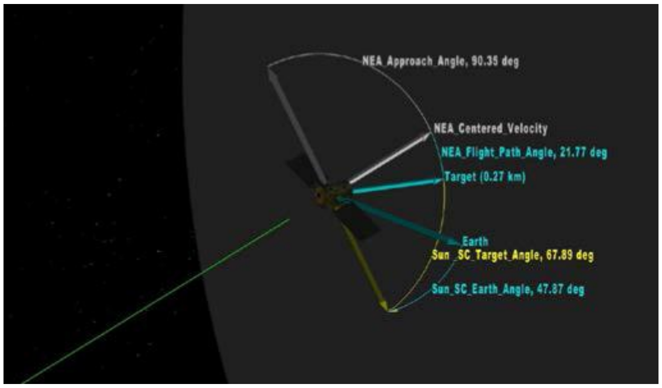


Figure 7. Final Approach Angles, Geometry and Spacecraft Orientation with the Target

7. FUNCTIONAL CONCEPT

As part of the formulation strategy, a functional concept diagram was utilized. This functional block diagram (Figure 8) was design approach agnostic. Salient top-level functions of HAMMER are depicted in this figure. The major elements were used as concept building blocks. Kinetic payloads were depicted in simpler forms and building blocks. Multiple or modular kinetic or NED, payloads along with partitioning strategies, were conveyed to the concept development team. The separation and simplification of payload to spacecraft interfaces were depicted. This allowed for the partitioning of functions and the trading of these functions between the various elements of payload blocks and spacecraft subsystems.

The top portion contains the kinetic payload which is in the conical section. The center cylindrical section contains simpler / traditional spacecraft avionics such as Attitude Control System (ACS), thermal control system, Command and Data Handling (C&DH), secondary power, separation system and perhaps 8,000 kg of another mass slug. The bottom section contains more traditional spacecraft functions such as communications, ACS, C&DH, thermal, propulsion, solar arrays, and power.

The goal was to achieve modular, separable, loosely coupled functional allocations of the payload and spacecraft precision delivery functions. Additionally, the two-part payload concept would allow for change out of one of these as possible NED or hybrid combination payload(s). It is noted that a number of functions are shown in each section. This is to depict both the function(s) potentially needed within each section as well as denoting future allocation trade-offs such as performance or redundancy needs. Spacecraft housekeeping and other delivery functions were to be grouped together to allow for both simpler payload to spacecraft interfaces and to facilitate partitioning of functions within HAMMER, as well as inter-organizational/ inter-agency roles and responsibilities. It is anticipated that this functional block diagram will remain a work in progress and will continue to evolve and further develop as additional studies are continued and cases / point-of-departure concepts are considered.

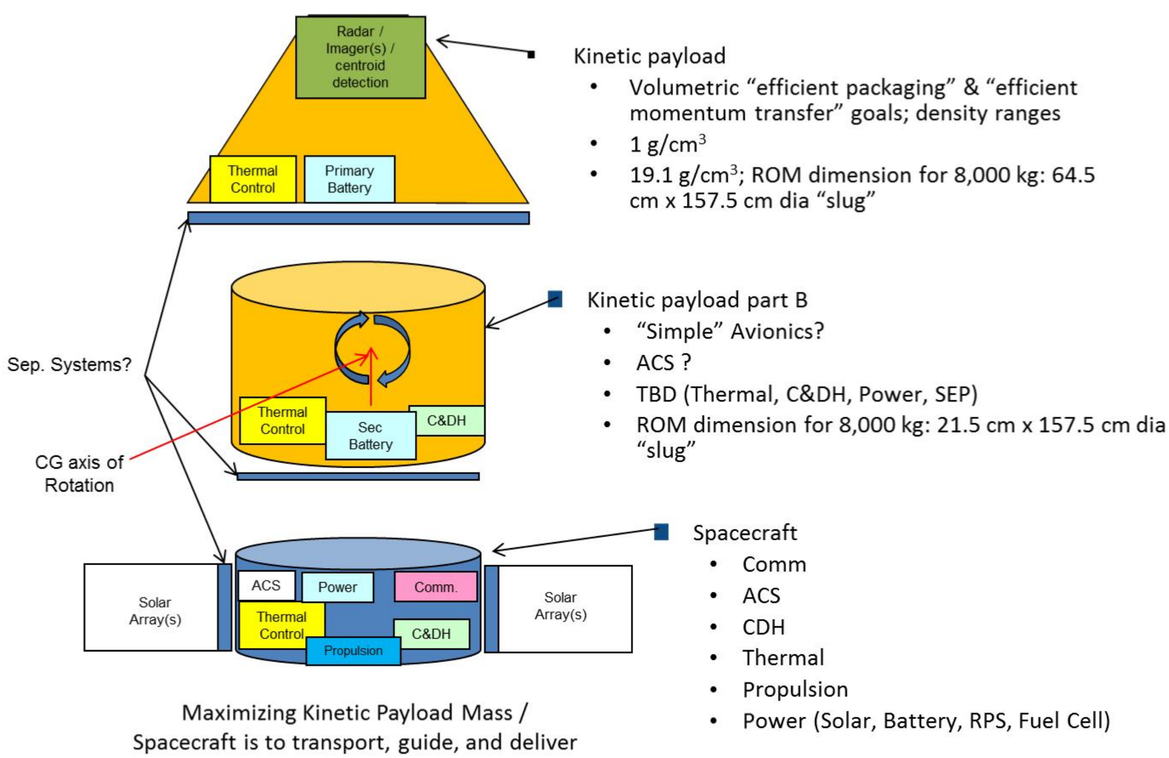


Figure 8. Modular Decomposition of HAMMER functions

8. FUNCTIONAL ALLOCATION FOR CASE 1

Figure 9 depicts the current representation for the Case 1 allocation of spacecraft functions and payload functions. Within the spacecraft there is a telecommunications subsystem, an avionics subsystem, propulsion subsystem, power subsystem, thermal subsystem, attitude control system and a control and data handling subsystem. On the lower left is the interface to the NED or kinetic energy device. It's a simpler interface to this modular exchangeable payload. It is a concept goal that one-way data and power are the only necessary interfaces for this modular payload. This concept should allow for modular exchange and very late integration of this payload with the spacecraft. Internal to this payload, it is anticipated that the payload would have its own batteries, control electronics, detection (camera-like) function and target acquisition system, as well as some internal navigation detection systems.

Figure 10 depicts the notional spacecraft concept. It is a rectangular structure where the mechanical/structural loads are carried along its length and through the center of the spacecraft with a thrust tube down the center. The potential NEDs are along the sides. This allows deployment of the NEDs, if necessary. The attitude control thrusters are in the corners of the spacecraft along with a propulsion system (+X axis) to allow release of the NEDs. The solar arrays and the single high-gain antenna are fully gimbaled. This is needed in order to maintain power from a distance of up to 1.4 AU from the Sun and communicate to Earth at a distance of up to 2.4 AU.

Figure 11 shows the spacecraft in the launch configuration within the Delta-IV Heavy. It occupies only about two-thirds of the volume since the spacecraft mass is concentrated. The overall fairing size of the Delta-IV Heavy is shown to be about 9.8 meters in length, about 4.6 meters in diameter within the dynamic fairing envelope. The spacecraft is attached to a 3-meter fairing adapter. The overall spacecraft length is just over 5 meters.

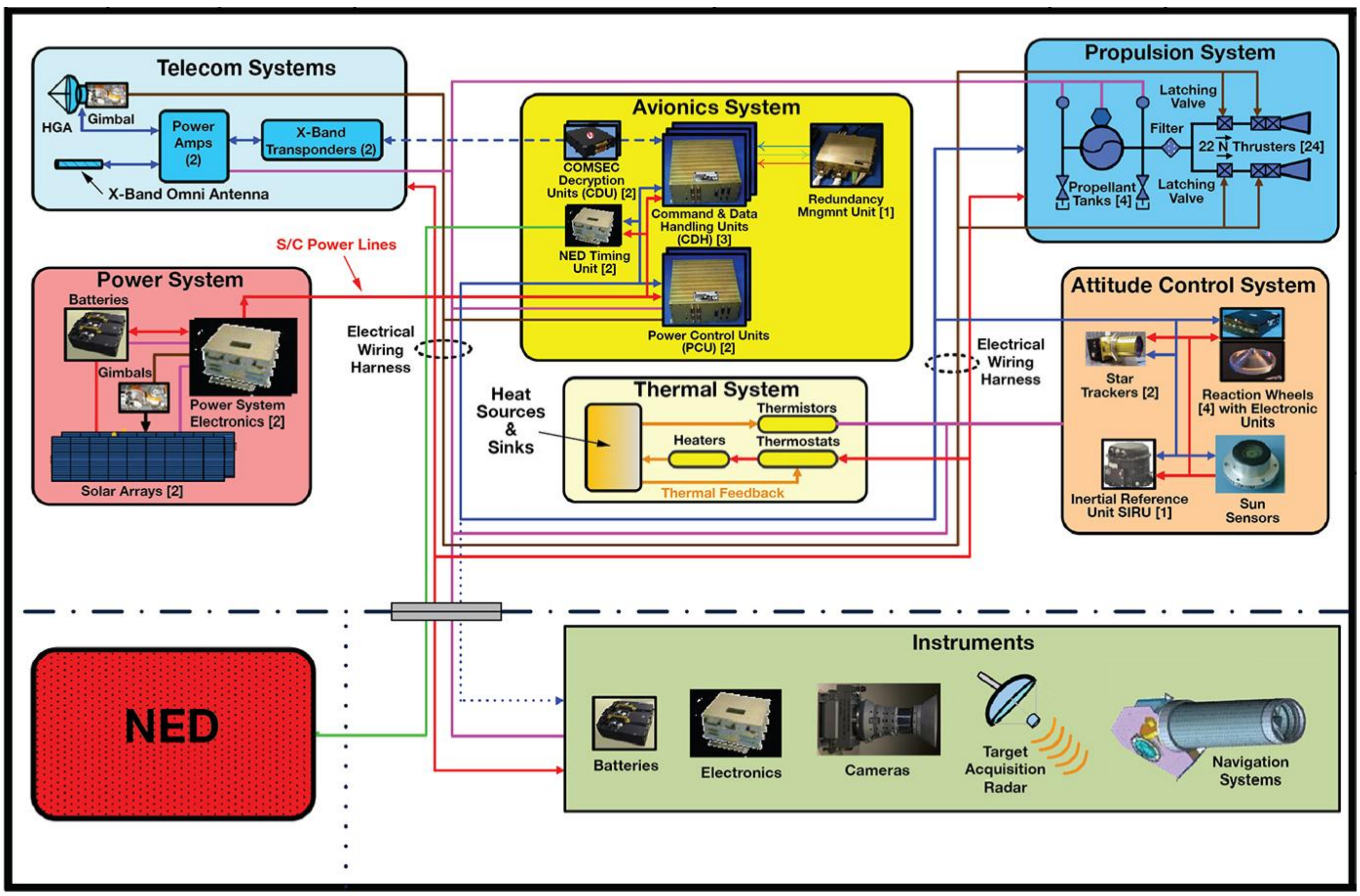


Figure 9. Spacecraft Subsystems Functional Block Diagram as Allocated for the Case 1 Study

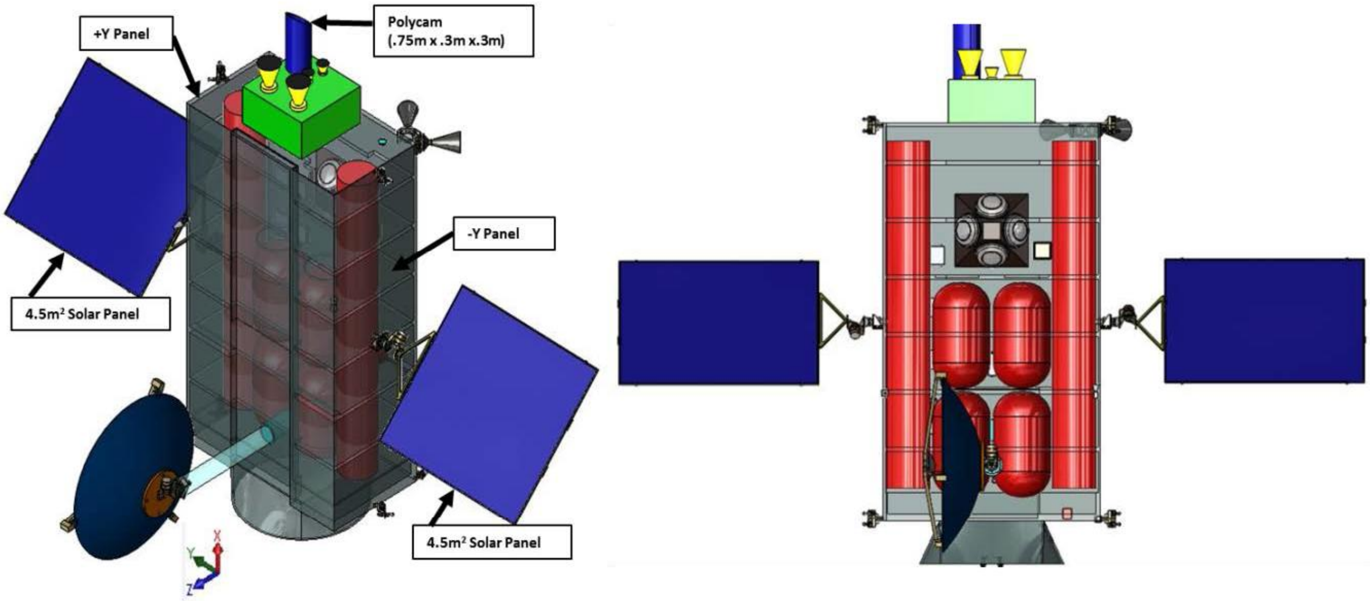


Figure 10. Case Study 1 Notional Point of Departure Spacecraft

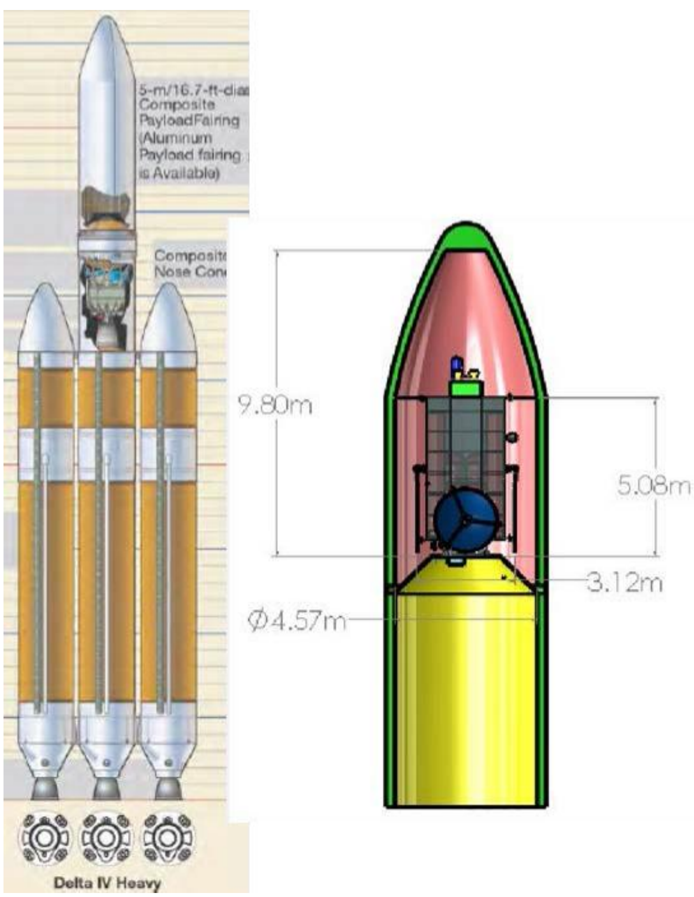


Figure 11. Case Study 1 HAMMER within the Delta-IV Heavy Fairing

Table 2 shows the mass rack up for the entire spacecraft + payload (observatory mass = total payload mass + total spacecraft wet mass). Please note that the payload carries a 0% contingency as it was used to completely maximize the mass of the launched payload to be delivered (Gross Liftoff Weight [GLOW]). Overall, at the concept Point-of-Departure level there remains just 7 kilograms of launch vehicle throw mass margin.

9. Alternative Launch Vehicles

Table 3 shows the options available for all other currently available launch vehicles and their launch capabilities for this point of departure mitigation target. Note that all other launch vehicles have significantly less capability of delivering a payload to the mitigation target.

Table 2. Case Study 1 Total Mass Summary (C3 of ~10 Km^2 / Sec^2) Unused LV Throw Mass – 7 kg

Hammer MASS Summary			
Payload Mass			
Hammer Payload Dry Mass	CBE	Cont.	MEV
Hammer Payload (Inert)	1000.0 kg	0%	1000.0 kg
Hammer Payload (Non-Inert)	1000.0 kg	0%	1000.0 kg
Hammer Payload (Balance)	600.0 kg	0%	600.0 kg
Polycam	9.0 kg	30%	11.7 kg
Radar	100.0 kg	30%	130.0 kg
Optical Camera	5.0 kg	30%	6.5 kg
Total Payload Mass	2714.0 kg	30%	2748.2 kg
Bus Dry Mass			
Hammer Spacecraft Bus Dry Mass	CBE	Cont.	MEV
Mechanical	3460.2 kg	30%	4498.3 kg
Thermal	36.1 kg	30%	46.9 kg
Attitude Control	57.5 kg	30%	74.8 kg
Propulsion	142.2 kg	30%	184.9 kg
Power	224.0 kg	30%	291.2 kg
Avionics (C&DH)	91.2 kg	30%	118.6 kg
Communications	97.1 kg	30%	126.2 kg
Spacecraft Bus Dry Mass Total	4108.3 kg	30%	5340.8 kg
Observatory Mass			
HammerObservatory Mass	CBE	Cont.	MEV
Payload Total	2714.0 kg	0%	2748.2 kg
Spacecraft Bus Dry Mass	4108.3 kg	30%	5340.8 kg
Observatory Dry Mass	6822.3 kg	19%	8089.0 kg
Propellant + Gas	774.0 kg	0%	774.0 kg
Observatory Launch Mass	7596.3 kg		8863.0 kg
Launch Vehicle Evaluation			
LV Throw Mass Margin* (Dry Mass) %			0%
Launch Vehicle Capability [Delta IV Heavy]			8870 kg
LV Throw Mass Margin*			7 kg

Key:	Notes:
CBE - Current Best Estimate Cont. - Contingency MEV - Maximum Expected Value	*MDL Margin Calculation, as weighted against LV capability versus launch mass.

Note: 30% contingency on S/C Bus; includes redundancy for Class A mission.

Table 3. Case Study 1, Launch Vehicle Options

Vehicle vs. C3 of 10 km2/sec2	Delta IV Heavy [2 launch pads, Space Launch Complex (SLC)-37 Eastern Test Range (ETR) and SLC-6 Western Test Range (WTR)]	Atlas V 551 (ETR and WTR (SLC-41, SLC-3E)	Falcon 9 v1.1 ETR and WTR (SLC-40, SLC-4E)	Ariane 5 (linear approx.) (1 Launch Pad, Kourou) Your Mileage May Vary
	8870 kg	5060 kg	2625 kg	6700 kg

10. NOTIONAL SPACE SEGMENT DEVELOPMENT AND DEPLOYMENT TIMELINE (1ST ARTICLE ONLY)

Figure 12 shows a notional mitigation mission development and deployment timeline. Note that it takes over 64 months in order to complete the development, design, and ATLO (Assembly, Test, and Launch Operations) of the mitigation vehicle and an additional 2 months for near earth on-orbit verification prior to the departure injection burn. Additionally, 25 months of transit time is needed in order to reach the target. In total, this accounts for over 7 years lead time (89 months) for an identified PHA target.

11. MISSION EFFECTIVENESS / MISSION SUCCESS

Preliminary mission mitigation effectiveness assessment was made for each of the specific subsystems used for the Case 1 study. As derived from the spacecraft system functional block diagram for this case study, each of the subsystems was modeled. See Figure 13 for the system spacecraft functional block diagram along with the root-sum-square (RSS) functional string assessment methodology. Table 4 provides the subsystem breakdown by mission mode / phase, and the reliability assessment for each of the subsystems.

Table 4. Case 1 Study Spacecraft Subsystem Reliability Including Duration and Modes

	Mode	Reliability
ACS	Full Mission	0.9989
Avionics	Cruise (710 Days)	0.9994
Avionics	Final 30 Days	1.0000
Instruments	Cruise (Standby)	0.9919
Instruments	Final 30 Days	0.9966
Power	Full Mission	0.9994
Propulsion	Full Mission	1.0000
Thermal	Full Mission	0.9987
Design Reliability	Full Mission	0.9849
Launch		0.9800
Mission Reliability		0.9652

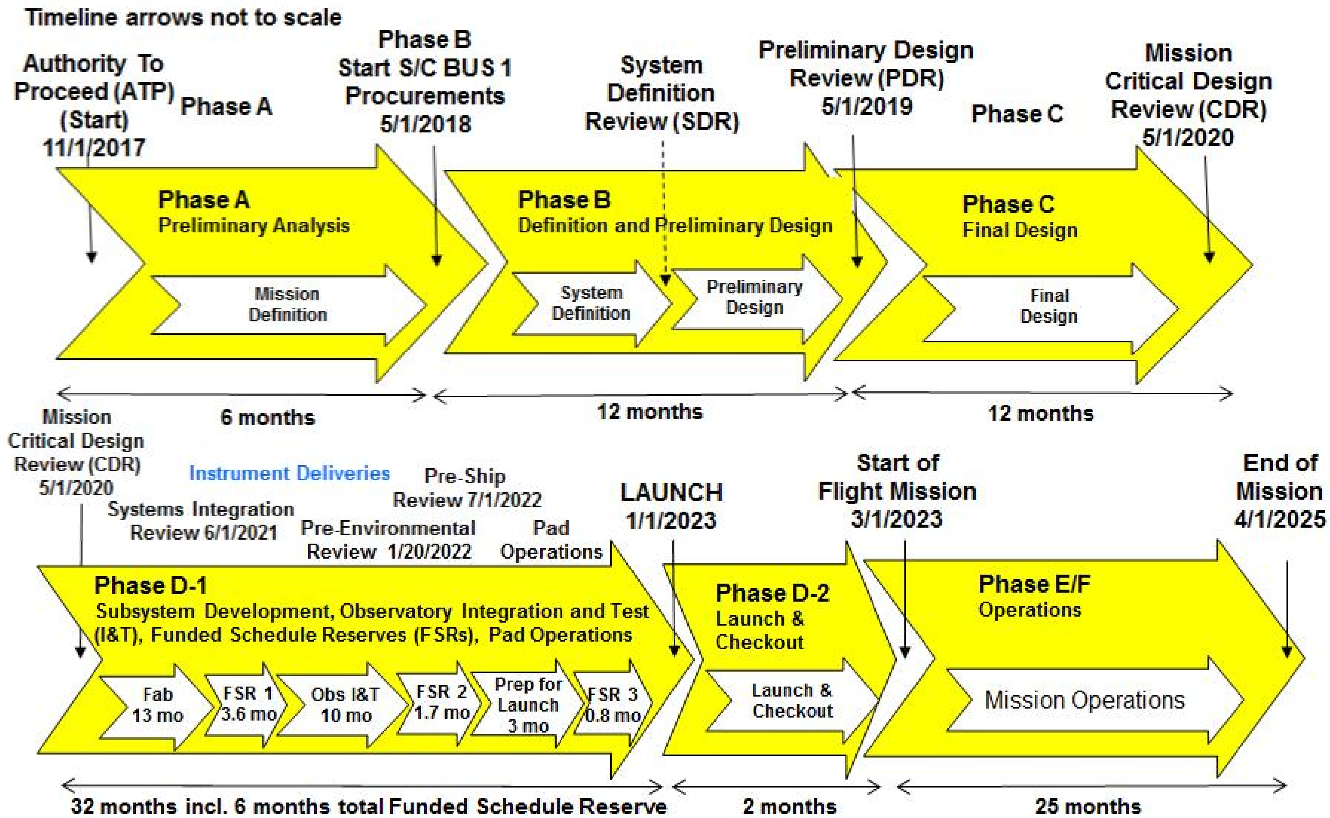


Figure 12. Case Study 1, Mitigation Mission Development and Deployment Timeline

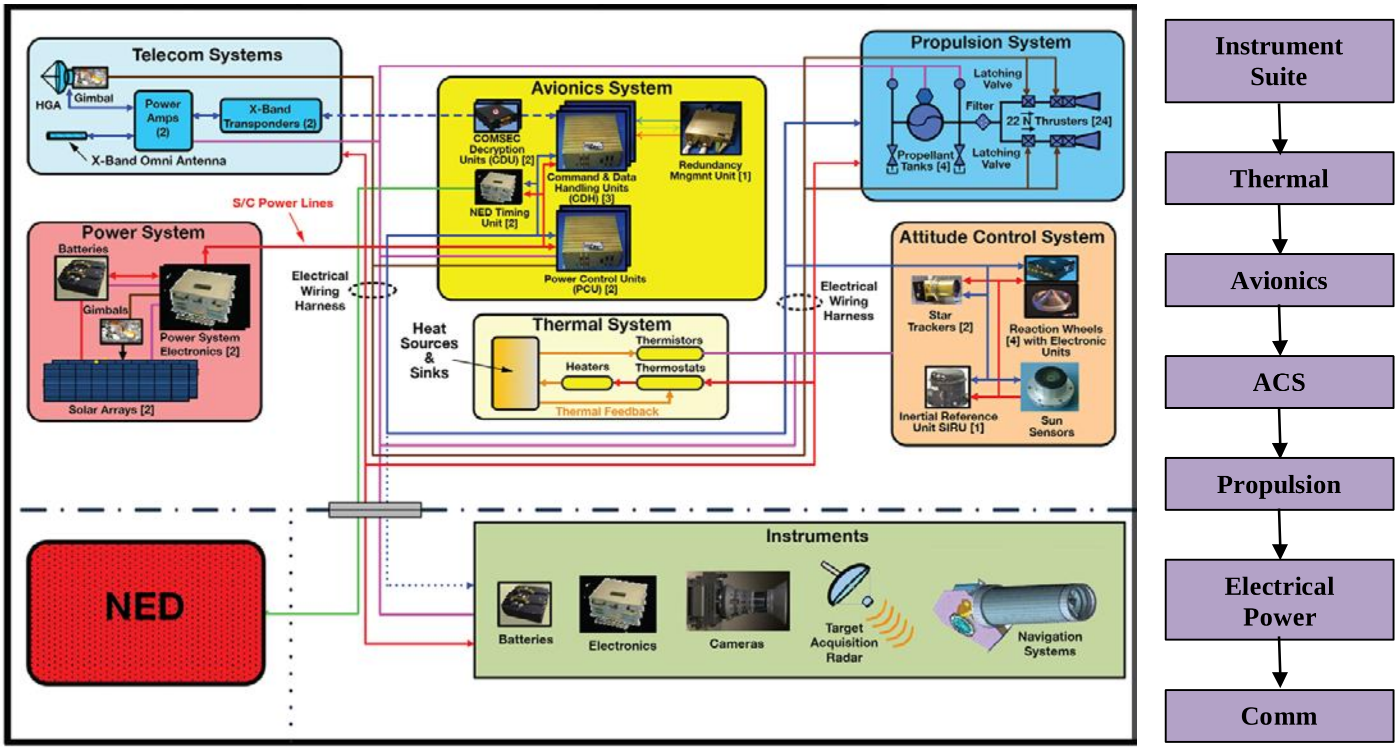


Figure 13. Preliminary Assessment of Case 1 Study Mission Reliability

Table 5 provides a multiple launch mission reliability trending assessment. K of N Mission reliability and confidence factors are depicted in the table. Key assumptions were made for both the kinetic energy/NED payload device(s) as well as software reliability. Both of these items were assumed to have a reliability factor of 1 for this initial analysis. From this table, one can see that multiple launches can achieve a high factor of confidence in delivering the payload to the PHA target. This table does not address the devaluation factors needed to account for both the software as well as the kinetic energy/NED devices. Significantly more detailed work will be needed to account for these additional subsystems as well as any future changes resulting from concept refinements and further developments.

What is noted here however, is that multiple launched missions have a significant effect in improving the overall mission confidence as denoted by each of the highlighting arrows. By employing multiple payload deliveries, one can achieve architectures or methods significantly mitigating the payload delivery system (spacecraft / launch vehicle aka transportation space segment) as an impacting factor to the overall mission reliability. It is therefore important to consider multiple launch delivery concepts in future case studies or mission level architecture or segment level concept trades.

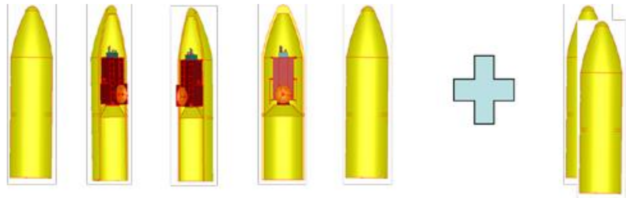
12. CONCEPT OF CAMPAIGN MODE

As an extension to the MDL study Case 1 activities, joint interagency team brainstorm activities included a reference to Table 5 under the 95% column for the 1 of 5, k of n. Note that a confidence factor of 1 was nominally / notionally achieved, however, this involved the launch or delivery of five payloads.

Table 5. Case 1 Study Delivery of Multiple of Payloads Improving Campaign Reliability

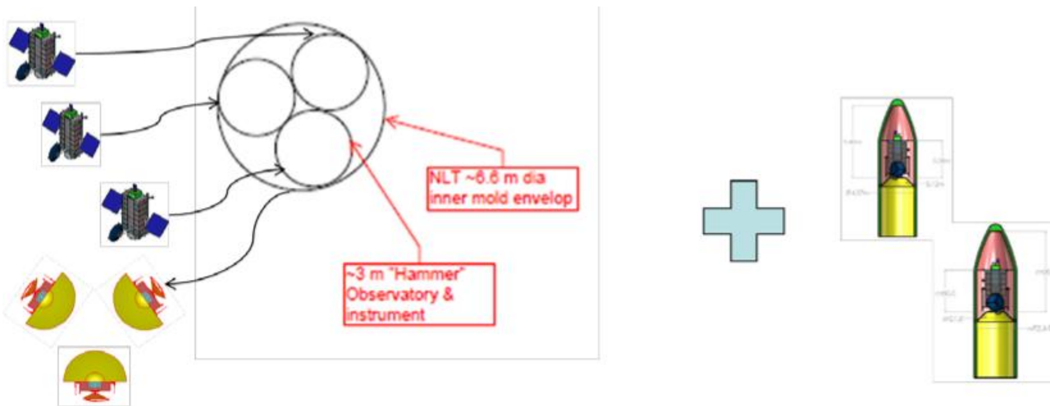
k of n	Mission Reliability	95% Confidence
1 of 1	0.9652	0.9362
1 of 2	0.9988	0.9959
2 of 2	0.9316	0.8764
1 of 3	1.0000	0.9997
2 of 3	0.9964	0.9883
3 of 3	0.8991	0.8205
1 of 4	1.0000	1.0000
2 of 4	0.9998	0.9990
3 of 4	0.9931	0.9776
4 of 4	0.8678	0.7681
1 of 5	1.0000	1.0000
2 of 5	1.0000	0.9999
3 of 5	0.9996	0.9976
4 of 5	0.9887	0.9642
5 of 5	0.8376	0.7191
1 of 6	1.0000	1.0000
2 of 6	1.0000	1.0000
3 of 6	1.0000	0.9998
4 of 6	0.9992	0.9955
5 of 6	0.9834	0.9486
6 of 6	0.8084	0.6732

This scenario includes a multiple launch delivery analysis suggesting five payloads plus two spares (see Figure 14). It is suggested that this concept may need further investigation or study along with the associated infrastructure assessments and analysis (launch facilities, launch processing operations, launch ops and support, etc.) For this concept, one may include the packaging of multiple payloads into a single launch vehicle thus reducing launch facility processing burden.



- What if from the previous multiple launcher analysis of adopting the suggestion a quantity of 5 mitigators plus 2 spares?
- This suggests the potential of ~5 launches, 5 launch vehicles, 5 launch processing facilities, and 5 launch processing teams?
  - This concept and the two additions will need further exploration
  - Other related / associated infrastructure questions were raised

Figure 14. Multiple Launchers



- What if the "transportation segment" could effectively deliver a set of at least ~5 "observatories" to the PHA of interest and only 1 of the 5 is needed for "mission success?
- What are ways to accomplish this?
  - What if these 5 HAMMERs could be grouped into a 3 - 1 - 1 delivery system?
    - One 30 mT TBD launch vehicle plus to TBD ELVs?
    - This might be done with 3 launch site + facilities?
      - Candidate for further Brainstorming

Figure 15. Concept of Packaging 3 Hammers into a Single Larger Launch Vehicle Shroud

This might be accomplished, in concept, within the Space Launch System (SLS) (see Figure 15). The SLS would be augmented by two additional Delta-IV Heavy launch vehicles. A single SLS block 1A might be capable of delivering 30 metric tons to a PHA target for mitigation (about three times the capacity of the Delta-IV Heavy). The two Delta-IV Heavy delivery systems would follow shortly thereafter as a mission reliability improvement concept providing both launch vehicle diversity as well as an additional independent payload delivery system. The total of five payloads would be delivered to the intended PHA while the goal is that only one of these is needed to achieve mitigation success. It is suggested that further case study be considered for these kinds of options.

13. Team Findings

Preliminary study findings of the effectiveness of a KE (Kinetic Energy only) HAMMER are summarized in Table 6 through Table 10. The differences in the analysis results in these tables come from the assumed difference of one PHA characteristic, β, varied from an assumed value of 1 to an assumed value of 2.5. This β momentum enhancement factor is coupled with the physics and physical characteristics of the PHA being mitigated. This factor is depicted in Figure 16.

The overall set of results for the various case studies is summarized in Table 6. These results include both Delta IV Heavy and SLS Block 1 launch vehicle options, either 10-year or 25-year launch lead time, single or multiple launch options, and either quantization of available spacecraft launch mass into a number of discrete HAMMER spacecraft, or utilization of all available spacecraft launch mass capability without packaging that mass into individual HAMMER spacecraft.

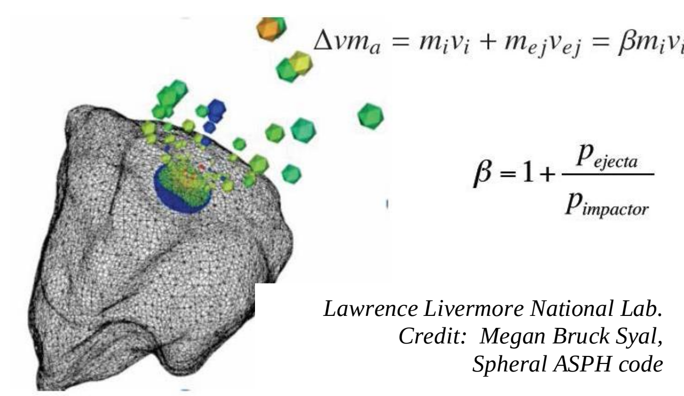


Figure 16. β, Momentum Enhancement for KI Mitigation Concepts[6]

Table 6. Deflection of at least ~1.4 Earth radii from Earth's surface and β=1

Available Launch Lead Time (yrs)	Launch Vehicle	Number of Launches	Return Impact within 50 Years	Used KI Mass (kg)	Total Specific Energy Imparted to NEO (J/kg)	Time from Defl. to Earth Encounter (yrs)	Δv Imparted to NEO (cm/s)	Defl. (R_E)	Defl. Perigee (R_E)	Defl. Bplane (R_E)
10	Delta IV Heavy	1	N/A	9072.64	1.051512	1.15	0.045632	-0.168	0.832	0.085
10	Delta IV Heavy (HAMMER)	1	N/A	7300	1.358724	1.07	0.046529	-0.169	0.831	0.065
10	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	1	N/A	7300	1.358724	1.07	0.046529	-0.169	0.831	0.065
10	SLS Block 1 w/iCPS	1	N/A	3085.54	0.750999	0.55	0.022490	-0.172	0.828	0.031
10	Delta IV Heavy	75	No	627090.90	258.23607	8.22	6.266800	**1.432**	**2.432**	**3.786**
10	Delta IV Heavy (HAMMER)	83	No	605900.00	307.56012	8.24	6.722607	**1.435**	**2.435**	**3.789**
10	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	83	No	605900.00	307.56012	8.24	6.722607	**1.435**	**2.435**	**3.789**
10	SLS Block 1 w/iCPS	29	No	628520.77	258.82489	8.22	6.281089	**1.440**	**2.440**	**3.794**
25	Delta IV Heavy	1	N/A	8405.80	2.750032	22.37	0.074874	-0.202	0.798	0.234
25	Delta IV Heavy (HAMMER)	1	N/A	7300	3.882061	22.34	0.082903	-0.213	0.787	0.244
25	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	1	N/A	7300	3.882061	22.34	0.082903	-0.213	0.787	0.244
25	SLS Block 1 w/iCPS	1	N/A	21811.95	7.135973	22.37	0.194289	-0.213	-0.787	0.607
25	Delta IV Heavy	16	No	134491.38	43.997952	22.37	1.197949	**1.384**	**2.384**	**3.733**
25	Delta IV Heavy (HAMMER)	17	No	124100.00	65.995730	22.34	1.409349	**1.444**	**2.444**	**3.799**
25	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	17	No	124100.00	65.995730	22.34	1.409349	**1.444**	**2.444**	**3.799**
25	SLS Block 1 w/iCPS	6	No	130870.07	42.813265	22.37	1.165693	**1.294**	**2.294**	**3.632**

The results in Table 6 show that, with an assumed β of 1, a maximum launch lead time of 25 years, and a requirement to achieve a deflection of at least ~1.4 Earth radii, neither the Delta IV Heavy nor the SLS Block 1b is able to deflect Bennu with a single launch. Furthermore, the number of launches required is extremely large, ranging from 29 to 83 for the 10-year launch lead time cases. Somewhat fewer launches are required for the 25-year launch lead time cases, in which the required number of launches is approximately 17 for the Delta IV Heavy or 6 for the SLS Block 1b. An additional concern is that the specific energy imparted to the asteroid is high (~250—300 J/kg) for the many-launch 10-year launch lead time cases that achieve ~1.4 Earth radii deflection. For reference, it is possible that the asteroid could be undesirably disrupted at deflection specific energy levels of ~100 J/kg, although this is currently an area that requires further research.

Note that the deflection perigee values achieved for the successful cases in Table 6 are on the order of ~2.3—2.5 Earth radii. It may be that a minimally successful asteroid deflection only requires a deflection perigee *altitude* on the order of ~0.25 Earth radii (to essentially just barely miss the Earth), corresponding to a deflection perigee *radius* of ~1.25 Earth radii. Thus, the results in Table 6 may be considered to be minimally robust, while a ~0.25 Earth radii deflection perigee results set could be considered minimally functional. To understand the reduction in launch costs associated with relaxing the deflection requirement to a deflection perigee of ~0.25 Earth radii, the study was repeated with that setting, and the results are summarized in Table 7.

Note that only requiring a ~1.25 Earth radii deflection perigee radius results in only requiring deflection values on the order of ~0.3 Earth radii, much smaller than the ~1.4 Earth radii deflection required in the first results set. The number of launches required for the 10-year launch lead time cases is still extremely high, ranging from 19 to 53 launches. The number of launches required in the 25-year cases are also proportionately reduced, but still formidable at 11 Delta IV Heavy launches or 4 SLS Block 1b launches.

We next consider the effects of β on the results. As noted previously, β will linearly scale the magnitude of the Δv vector imparted to the asteroid by the kinetic impactor(s). We return to the case of requiring a deflection of ~1.4 Earth radii (which we may, perhaps, consider minimally robust) and reassess performance with a β of 2.5 (rather than the original value of β=1). The results are presented in Table 8.

Note that changing from β=1 to β=2.5 has a nearly linear (inverse) effect on the required number of launch vehicles, modulated by the fact that we model an integer number of launches (i.e., we cannot, of course, have fractional launches). This is a significant effect, and so we are motivated to seek an improved understanding of β. That said, even with β>1 we would need to deploy tens of metric tons worth of payload in order to just barely deflect Bennu (or an asteroid of similar size/mass on a similar orbit to Bennu's).

From the foregoing results, we find a HAMMER spacecraft in kinetic impactor mode is not an adequate solution for deflecting Bennu (or similar / more challenging near-Earth objects [NEOs]). This raises the question of: for what size NEO can a single HAMMER in kinetic impactor mode produce an adequate deflection? Understanding the capability of a single kinetic impactor HAMMER is important, because we want a system that is fully capable of robustly achieving the threshold deflection mission with a single spacecraft. That allows us to then deploy a campaign of several such spacecraft for mission robustness through redundancy. By contrast, a deflection mission that depends on the success of several spacecraft is much less reliable.

Table 7. Deflection of at least ~0.25 Earth radii from Earth's surface and β=1

Available Launch Lead Time (yrs)	Launch Vehicle	Number of Launches	Return Impact within 50 Years	Used KI Mass (kg)	Total Specific Energy Imparted to NEO (J/kg)	Time from Defl. to Earth Encounter (yrs)	Δv Imparted to NEO (cm/s)	Defl. (R_E)	Defl. Perigee (R_E)	Defl. Bplane (R_E)
10	Delta IV Heavy	1	N/A	9072.64	1.051512	1.15	0.045632	-0.168	0.832	0.085
10	Delta IV Heavy (HAMMER)	1	N/A	7300	1.358724	1.07	0.046529	-0.169	0.831	0.065
10	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	1	N/A	7300	1.358724	1.07	0.046529	-0.169	0.831	0.065
10	SLS Block 1 w/iCPS	1	N/A	3085.54	0.750999	0.55	0.022490	-0.172	0.828	0.031
10	Delta IV Heavy	48	No	401338.2	165.271083	8.22	4.010763	**0.248**	**1.248**	**2.422**
10	Delta IV Heavy (HAMMER)	53	No	386900.0	196.392874	8.24	4.292750	**0.246**	**1.246**	**2.418**
10	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	53	No	386900.0	196.392874	8.24	4.292750	**0.246**	**1.246**	**2.418**
10	SLS Block 1 w/iCPS	19	No	411789.5	169.574927	8.22	4.115207	**0.300**	**1.300**	**2.485**
25	Delta IV Heavy	1	N/A	8405.80	2.750032	22.37	0.074874	-0.202	0.798	0.234
25	Delta IV Heavy (HAMMER)	1	N/A	7300	3.882061	22.34	0.082903	-0.213	0.787	0.244
25	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	1	N/A	7300	3.882061	22.34	0.082903	-0.213	0.787	0.244
25	SLS Block 1 w/iCPS	1	N/A	21811.95	7.135973	22.37	0.194289	-0.213	-0.787	0.607
25	Delta IV Heavy	10	No	92463.79	30.250348	22.37	0.823618	**0.381**	**1.381**	**2.575**
25	Delta IV Heavy (HAMMER)	11	No	80300.00	42.702672	22.34	0.911927	**0.292**	**1.292**	**2.466**
25	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	11	No	80300.00	42.702672	22.34	0.911927	**0.292**	**1.292**	**2.466**
25	SLS Block 1 w/iCPS	4	No	87247.81	28.543894	22.37	0.777157	**0.264**	**1.264**	**2.430**

Table 8. Deflection of at least ~1.4 Earth radii from Earth's surface and β=2.5

Available Launch Lead Time (yrs)	Launch Vehicle	Number of Launches	Return Impact within 50 Years	Used KI Mass (kg)	Total Specific Energy Imparted to NEO (J/kg)	Time from Deflection to Earth Encounter (yrs)	Δv Imparted to NEO (cm/s)	Defl. (R_E)	Defl. Perigee (R_E)	Defl. Bplane (R_E)
10	Delta IV Heavy	1	N/A	9045.11	2.203519	4.71	0.16489	-0.165	0.835	0.139
10	Delta IV Heavy (HAMMER)	1	N/A	7300	1.223178	0.06	0.11037	-0.172	0.828	0.017
10	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	1	N/A	7300	1.223178	0.06	0.11037	-0.172	0.828	0.017
10	SLS Block 1 w/iCPS	1	N/A	13507.25	9.752915	2.44	0.42392	-0.164	0.836	0.153
10	Delta IV Heavy	30	No	250836.36	103.2944	8.22	6.26683	**1.432**	**2.432**	**3.786**
10	Delta IV Heavy (HAMMER)	34	No	248200	125.9879	8.24	6.88461	**1.519**	**2.519**	**3.880**
10	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	34	No	248200	125.9879	8.24	6.88461	**1.519**	**2.519**	**3.880**
10	SLS Block 1 w/iCPS	12	No	260077.56	107.1	8.22	6.49771	**1.560**	**2.560**	**3.925**
25	Delta IV Heavy	1	N/A	8405.8	2.750032	22.37	0.18719	-0.224	0.776	0.584
25	Delta IV Heavy (HAMMER)	1	N/A	7300	3.882061	22.34	0.20726	-0.221	0.779	0.560
25	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	1	N/A	7300	3.882061	22.34	0.20726	-0.221	0.779	0.560
25	SLS Block 1 w/iCPS	1	N/A	21811.95	7.135973	22.37	0.48572	-0.191	0.809	1.517
25	Delta IV Heavy	7	No	58839.98	19.24910	22.37	1.31026	**1.703**	**2.703**	**4.083**
25	Delta IV Heavy (HAMMER)	7	No	51100.00	27.17471	22.34	1.45080	**1.546**	**2.546**	**3.911**
25	Delta IV Heavy (HAMMER) Adjusted Mass (max mass 7300)	7	No	51100.00	27.17471	22.34	1.45080	**1.546**	**2.546**	**3.911**
25	SLS Block 1 w/iCPS	3	No	65435.04	21.40663	22.37	1.45712	**2.126**	**3.216**	**4.540**

For the case of Bennu, we find that, with a 10-year launch lead time and β=1, a single HAMMER in kinetic impactor mode can adequately deflect an NEO up to 123.8 m in diameter by ~1.4 Earth radii, or deflect an NEO up to 143.62 m in diameter by ~0.25 Earth radii. Note that both cases assume an asteroid bulk density of 1 g/cm^3 and Bennu's orbit. This is important to keep in mind, because the answer will vary depending on NEO orbit, bulk density, launch lead time, warning time (which, as noted previously, is not the same as launch lead time), and other factors.

To quantify some of the variability in the size of NEO that can be dealt with via a single HAMMER, we hold the orbit constant at Bennu's orbit, and then vary the asteroid bulk density, β, and launch lead time. For each combination of those three parameters, we use our algorithms to compute the largest size asteroid that a single HAMMER spacecraft could deflect 1.4 Earth radii from Earth's surface. These results are presented in Table 9. Similar results are presented in Table 10 for deflection of at least 0.25 Earth radii.

Table 9. Effects of Density and β on HAMMER performance, when deflection of at least 1.4 Earth radii from Earth's surface is required

Available Launch Lead Time (yrs)	β	Asteroid Density (g/cm^3)	Diameter (m)	Deflection (R_E)	Specific Energy (J/kg)	Escape velocity fraction $\left(\frac{\Delta v}{v_{esp}}\right)$
10	1	1	123.8	1.436	307.587357	1.452552
10	1	2.6	90.03	1.436	307.606364	1.238811
10	2.5	1	168.02	1.436	123.040326	1.070316
10	2.5	2.6	122.19	1.436	123.041062	0.912753
25	1	1	210.028	1.444	65.995738	0.179480
25	1	2.6	152.74	1.456	65.995028	0.153056
25	2.5	1	285.052	1.456	26.397977	0.132241
25	2.5	2.6	207.300	1.456	26.398039	0.112773

Table 10. Effects of Density and β on HAMMER performance, when deflection of at least 0.25 Earth radii from Earth's surface is required

Available Launch Lead Time (yrs)	β	Asteroid Density (g/cm^3)	Diameter (m)	Deflection (R_E)	Specific Energy (J/kg)	Escape velocity fraction $\left(\frac{\Delta v}{v_{esp}}\right)$
10	1	1	143.62	0.252	197.008810	0.801965
10	1	2.6	104.4456	0.252	197.009251	0.683903
10	2.5	1	194.922	0.252	78.803923	0.590899
10	2.5	2.6	141.75	0.252	78.811186	0.503969
25	1	1	244.6	0.251	41.780494	0.097566
25	1	2.6	177.881	0.251	41.781198	0.083204
25	2.5	1	331.9733	0.249	16.712194	0.071887
25	2.5	2.6	241.123	0.256	16.774658	0.061610

We observe in the foregoing results that there is an apparent scaling relationship that may be exploited to predict asteroid deflection performance for a particular mission scenario without the need to execute the trajectory grid calculations. This allows us to predict the total spacecraft mass required to deflect a given asteroid mass by a certain amount, provided that we already know how much spacecraft mass is required to impart that amount of deflection to an asteroid of some other mass. We begin with the equivalency, following from linear momentum conservation, given by

$$\beta_1\left(\frac{\bar{m}_1}{M_1+\bar{m}_1}\right)=\beta_2\left(\frac{\bar{m}_2}{M_2+\bar{m}_2}\right) \tag{1}$$

where M_1 is asteroid mass, β_1 is the momentum enhancement parameter used in the calculation of the asteroid's deflection, and $\bar{m}_1$ is the total spacecraft mass used to deflect the asteroid. Those same parameters subscripted "2" correspond to a different case of interest for which we seek to solve for one of the three parameters given the other two. Note that, for a multiple launch scenario, the total spacecraft mass is simply the sum of the masses of the individual spacecraft used to impact the asteroid, given by

$$\bar{m}_1=\sum_{i=1}^{N_1} m_{1_i} \tag{2}$$

where N_1 is the number of launches and m_{1_i} is the mass of the i^{th} spacecraft. If the mass of each of the N spacecraft is the same, then Eq. (2) reduces to

$$\bar{m}_1=N_1\,m_1 \tag{3}$$

We apply these relationships by manipulating Eq. (1) to yield a scale factor, S_2, given by

$$S_2=\left(\left(\frac{\bar{m}_2\beta_2}{\bar{m}_1\beta_1\rho_2V_{unit_2}}\right)\left(\rho_1S_1^3V_{unit_1}+\bar{m}_1\right)-\frac{\bar{m}_2}{\rho_2V_{unit_2}}\right)^{\frac{1}{3}} \tag{4}$$

where $M_1=\rho_1V_1=\rho_1S_1^3V_{unit}$, ρ is the asteroid's density, S is the scale factor corresponding to a particular asteroid radius/diameter, and V_{unit} is the volume of the object when the radius is normalized to a maximum value of of 1 (unit radius volume). If each of the two asteroids being considered has the same shape, then the unit radius volume will be equal for both bodies.

When the mass of each asteroid is much greater than the total spacecraft mass impacting the asteroid ($\bar{m}_1$ and $\bar{m}_2$), and the spacecraft mass per launch is constant, then the scale factor reduces to

$$S_2\approx\left(\frac{\rho_1\beta_2N_2}{\beta_1\rho_2N_1}\right)^{\frac{1}{3}}S_1 \tag{5}$$

The scale factors used for the previous tables (9 and 10) are for the diameters of each object. In addition, the known parameters are variables with the subscript "1," and the input variables are β_2, N_2, and ρ_2.

A similar expression, manipulating equation (1) and assuming that each system launch vehicle has the same mass (i.e. $\bar{m}=Nm$), can be found for the number of launch vehicles required

$$N_2=\frac{M_2}{\frac{m_2\beta_2}{N_1m_1\beta_1}M_1+\left(\frac{\beta_2}{\beta_1}m_1-m_2\right)} \tag{6}$$

This equation can be further reduced when the following assumptions are made: $m_1=m_2=m$, asteroid is same shape, $M_1\gg N_1m$, and $M_2\gg N_1m$.

$$N_2\approx\frac{\rho_2\beta_1S_2^3}{\rho_1\beta_2S_1^3}N_1 \tag{7}$$

When investigating estimation of launch vehicles across launch vehicle types and deflections, it has been found that the number of launch vehicles can be closely predicted. This is done by taking ratios from other deflections and intercept dates. Note, the deflections in each lead time must be the same, but each lead time group can have a different deflection. See Tables 6, 7, and 8. The relation is as follows:

$$N_2(LT_2,LV_1)\approx\frac{N_2(LT_2,LV_2)}{N_1(LT_1,LV_2)}N_1(LT_1,LV_1) \tag{8}$$

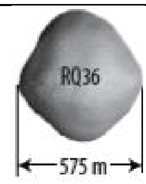
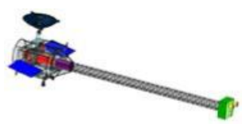
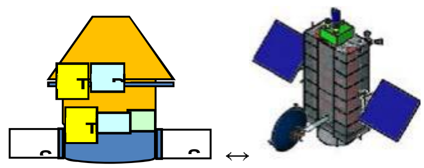
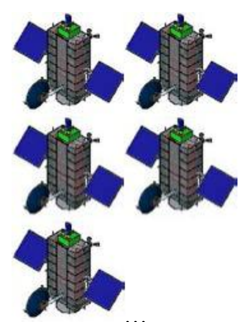
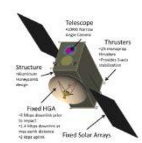
Where the LT is the lead time, LV is launch vehicle type, and N is the number of launch vehicles required as a function of LT and LV. An example to find the number of LVs needed to deflect the asteroid 0.25 RE with a 25 year LT using the data in Tables 7 and 8 is as follows: $N_1(LT_1,LV_1)=12$ (SLS), $N_1(LT_1,LV_2)=30$ (Delta IV), and $N_2(LT_2,LV_2)=10$ (Delta

IV). Inputting these values into equation 8 yields 4 (SLS), which is the number of Launch vehicles required for deflecting the asteroid about 0.25 Earth radii with a LT of 25 years. However, further investigations must be conducted to explore the reliability of launch vehicle estimation when intercept dates greatly vary in the same lead time.

14. SUMMARY

- The HAMMER in kinetic impactor (KI) mode is clearly not an adequate solution for deflecting Bennu (or similar / more challenging NEOs).
- Improved understanding of β is needed, as it can linearly decrease the required number of launches for kinetic impact deflection.
- Based on the above, even with $\beta>1$ the physics will likely dictate deploying tens of metric tons worth of spacecraft in order to just barely alter the Bennu trajectory. This study utilized both the largest vendor available launch vehicle, the Delta IV Heavy, as well as the projected capabilities of the future NASA SLS version 1b. The ability to use other less capable launch vehicles is highly unlikely.
- Additionally, the ability to deliver the coordinated quantity and coordinated simultaneity of these HAMMERs would be unprecedented within the currently existing national launch system infrastructure.
- Use of international infrastructure of this magnitude was beyond this current case study.
- Removing the constraints on minimum/maximum distance to the Sun resulted in marginal improvements to deflection performance with a closest approach to the Sun of 0.4 – 0.6 AU.
- Removing/loosening the other constraints (Declination of the Launch Asymptote [DLA], phase angle, Sun-Spacecraft-Earth [SSE] angle, maximum flight time, etc.) did not lead to notable deflection performance improvements (some did lead to an increase in the number of launch opportunities).
- However, the above outcomes are particular to Bennu's orbit; the situation will vary depending on the particular NEO orbit.
- A single HAMMER in kinetic impactor mode is probably adequate to deflect a NEO ≤180 m in diameter (with bulk density of 1g/cm^3 and Bennu's orbit) with a 10-year launch lead time. This mitigation approach will vary depending on NEO orbit, bulk density, and launch lead time, and warning time (which is different than launch lead time).
- It became clear from this case study that the use of multiple HAMMERs would need to be part of the top level concept formulation and trade space along with future work and analysis into the PHA physical characteristics, payload complement within HAMMER (mass centroid detectors, terminal guidance systems, longer range detection and guidance, telemetry and communications, etc.).
- The study confirms previous reports from both the National Research Council (NRC)[7] and NASA[8] where the NED option is needed. Table 11 shows the concepts captured thus far beginning in 2012 with Hypervelocity Asteroid Intercept Vehicle (HAIV) as a single spacecraft fight system, the Case 1 work completed in 2015, and potential options for 2016 and beyond with modular spacecraft delivery systems using multiple launch vehicles approached in campaign concept.

Table 11. PHA Mitigation Cases and Options being Considered

Option	A Hypervelocity Asteroid Intercept Vehicle (HAIV) Impactor leads NED, Class B+ - 2012	B Hypervelocity Asteroid Mitigation Mission of Emergency Response (HAMMER) Kinetic Impactor & NED combination, Class A+ - 2016 (NEO Mitigation – Bennu)	C ~QTY 5 to 7 "Modular Observatories" HAMMER – 20xx?	D AIM / DART (2 part mission); Didymos encounter; kinetic	FOM / Rating +3, +1, 0, -1, -3 VG, G, Nutr, B, VB
Asteroid Size			To be studied		TBD
Potential NEA size / mass effective-ness range	NEA 50m radius; 62000 metric tons	NEA 500 m radius; 6.2e7 metric tons	To be studied	800m with moon 150m; ~1.7 gm / cm3	TBD
					TBD
Mission TOF / duration	122 days	NLT 900 days + TBD days / years storage; development ~ 5 yrs; total ~5.6 to 8 yrs	TOF range: 1.6 - ~3 years; devel: ~ 5 years; total ~5.6 to 8 years (cascade / multi-thread w/ potential refresh)	~ 1 AU solar distance; 15 months for DART (~425 days)	TBD
Encounter Method-ology	NED	Kinetic Impactor or NED	Campaign mode: 3 - 1 - 1	AIM + DART	TBD

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