Multiple Mass Drivers as an Option for Asteroid Deflection Missions

John R. Olds[*], A.C. Charania[†], and Mark G. Schaffer[‡]
SpaceWorks Engineering, Inc. (SEI), Atlanta, Georgia, 30338

Several competing techniques have been proposed for deflecting asteroids on potential collision courses with Earth. This paper summarizes recent efforts to define, at a conceptual level, a proposed deflection technique using multiple mass drivers. The MADMEN concept (Modular Asteroid Deflection Mission Ejector Node) examined in this paper would use multiple mass driver-equipped landers to rendezvous and attach to a threatening asteroid, drill into its surface, and eject small amounts of the asteroid material away at high velocity using a mass driver (rail gun or electromagnetic launcher). The effect, when applied over a period of weeks or months, would eventually change the heliocentric velocity of the target asteroid and thereby alter its closest approach to Earth. The use of multiple, small landers provides a system-level redundancy against failure of individual landers and thereby improves the overall probability of successful deflection. This study extends and further develops previous conceptual design work on this topic. The mass driver concept is shown to result in a deflection concept that compares favorably to other non-nuclear detonation approaches based on launch mass, while offering a number of qualitative advantages. Key challenges and technical hurdles that pertain to the concept are also identified and discussed. Two deflection case studies are introduced to provide specific sizing and costing examples for the proposed approach.

Nomenclature

AOA	=	Analysis of Alternatives
Beta	=	Impactor Momentum Amplification Factor (resultant momentum/incoming momentum)
C_3	=	Square of the residual velocity at the sphere of influence (km^2/s^2)
ΔV	=	Delta-Velocity (m/s)
DDT&E	=	Design, Development, Test and Evaluation
Isp	=	Rocket Engine Specific Impulse (s)
JAT	=	JAVA Astrodynamics Toolkit
LCC	=	Life Cycle Cost
LEO	=	Low Earth Orbit
MADMEN	=	Modular Asteroid Deflection Mission Ejector Node
NASA	=	National Aeronautics and Space Administration
NEO	=	Near Earth Object
NIAC	=	NASA Institute for Advanced Concepts
PHA	=	Potentially Hazardous Asteroid
R_e	=	Equatorial Radius of the Earth (6378 km)
ROSETTA	=	Reduced Order Simulation for Evaluation Technologies and Transportation Architectures
SEI	=	SpaceWorks Engineering, Inc.
SSI	=	Space Studies Institute
TFU	=	Theoretical First Unit
TRL	=	Technology Readiness Level

---

[*] Principal Engineer and CEO, 1200 Ashwood Parkway, Suite 506, Associate Fellow AIAA.
[†] Senior Futurist, 1200 Ashwood Parkway, Suite 506, Member AIAA.
[‡] Project Engineer, 1200 Ashwood Parkway, Suite 506, Member AIAA.

I. Introduction

Given the knowledge that asteroids and comets have the potential to impact the Earth and cause widespread damage, researchers are proposing various mitigation techniques that might be used to alter the course of such objects and thus prevent impact. Several deflection options have been proposed[1,2]. Some of these options include the following:

- Nuclear Detonation (standoff)
- Nuclear Detonation (surface or subsurface)
- Kinetic Impactor
- Gravity Tractor
- Mass Driver
- Propulsive Tug
- Focused or Pulsed Laser Ablation of Surface Material
- Enhanced Yarkovsky Effect (albedo changing approaches)

Each proposed technique may be analytically demonstrated to offer advantages for a particular deflection scenario, but engineering work to refine and mature any of these concepts remains in its infancy. While no actual asteroid/comet deflection mission is currently necessary, researchers are using this lead time to explore the broad range of alternatives with the goal of producing a consensus for one or more of the options to receive more detailed study. In this early study phase, the field should also remain very open to the introduction of new ideas and innovative solutions. Further study of each concept is appropriate and warranted before one approach is designated as the preferred global option for any potential threat.

The authors here have chosen to concentrate initial investigations of mitigation techniques on the use of mass drivers as a low thrust, sustained technique to alter the trajectory of a Potentially Hazardous Asteroid (PHA). The authors are aware of no other organization actively investigating this approach, despite some of the attractive advantages it potentially offers. The study activity reported in this paper was initiated in 2004 under Phase 1 study sponsorship from the NASA Institute for Advanced Concepts (NIAC) and continues at a low level of effort under SpaceWorks Engineering, Inc. (SEI) internal company resources. The goal is to produce a reference configuration that represents the mass driver approach complete with preliminary mass, performance, cost, and top-level reliability estimates.

A. Mass Drivers

The basic physics of using a mass driver (i.e. electromagnetic accelerator or rail gun) as a propulsive device is based in the fundamental concept described in Isaac Newton's Third Law of Motion: For every action, there is an equal and opposite reaction. Applying a force in one direction to accelerate a small bit of ejecta material will have the effect of applying an equal reactive force in the opposite direction. Momentum is conserved in this exchange. Mass drivers then, may be thought of as "propellant-less" rocket engines (albeit with a smaller exit velocity), using only inert in-situ material as the ejecta. If the power and infrastructure of the ejection systems are sufficiently small, then this approach could offer advantages for sustained application of thrust versus a standard rocket that is naturally limited by its onboard propellant mass.

Mass drivers use rings of electromagnets, sequentially energized, to accelerate materials to high velocity along a track or tube. Ferrous materials can be accelerated directly. Non-ferrous materials require a shuttle bucket to interact with the magnetic fields produced by the magnets. This latter approach also requires a deceleration segment of track if the bucket is to be recovered and reused. Mass driver technology has been developed for terrestrial applications and is frequently suggested or used as the basis for ballistic accelerators, long range artillery, space launch, and super colliders[3, 4, 5, 6, 7].

As with many space innovations, mass drivers were conceived of in science fiction, later being envisioned to solve real future space problems. Mass drivers in space appear in early science fiction stories in the early twentieth century and continue to pervade contemporary science fiction television shows and movies[8]. In the 1970's, American scientist Dr. Gerard O'Neill and his colleagues at the Space Studies Institute (SSI) proposed several uses for large mass drivers in space including launching raw materials from the lunar surface and using the aforementioned momentum exchange effect to slowly alter the course of an asteroid (see Figure 1). In one case, O'Neil proposed capturing a small asteroid into Earth orbit to be used as a source of precious metals and other materials[9, 10].

In O'Neill's vision, the mass driver is usually depicted as a large, high power device requiring significant infrastructure emplacement. In the opinion of the authors, the use of smaller, modular mass drivers, deployed in an intelligent swarm, offers advantages in terms of increased mission flexibility, lower manufacturing costs, reduced individual launch mass, improved overall mission reliability, and higher duty-cycle surface operations.

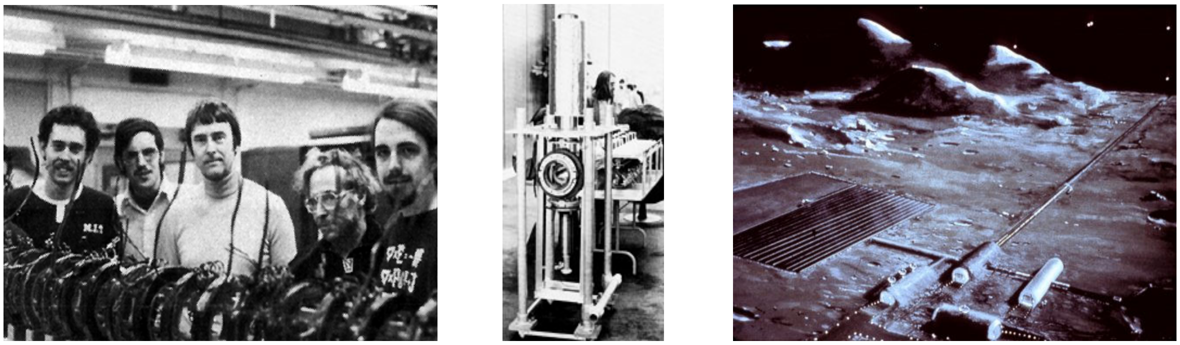

Figure 1. Dr. Gerard O'Neill (center) with Space Studies Institute (SSI) Colleagues, Demonstration Mass Driver at 1979 Princeton Conference, and Lunar Mass Driver Illustration[9, 10].

B. Comparison to Competing Deflection Concepts

Relative to competing deflection concepts, the use of multiple small mass driver-equipped landers may offer practical advantages. At a top-level, the relative advantages of this approach can be introduced and discussed qualitatively. Quantitative performance data is introduced in subsequent sections of the paper.

First, the technique does not require the proliferation of nuclear weapons into space. While nuclear warheads carry an extremely high energy to mass ratio, issues related to international treaties, public acceptance, and stockpile safety will continue to cast doubt over options depending on nuclear warheads.

Compared to kinetic impactors, mass driver solutions offer a controlled, metered application of perturbing force. Therefore there is an expectation that the approach will be more precise and regulated than the near instantaneous impulse applied when a high-speed impactor is slammed into a target asteroid. Uncertainties in impact location and timing, asteroid internal structure and mass distribution, and cratering physics are a concern for any kinetic impactor solution. Over the period of about one year, the lower thrust mass driver solution is able to achieve the same ΔV as a kinetic impactor of equivalent launch mass, but with much more precision and potential controllability.

Compared to other low thrust techniques such as the gravity tractor and the laser ablator, the mass driver solution provides significantly more impulse for a given action time and launch mass. Direct attachment obviates the need for station keeping and proximity control above a gravitational complex and physically irregular rotating body. Compared to other methods that depend on attachment to the target asteroid like a propulsive tug, the propellant-less approach of a mass driver can yield significantly longer action times for a given launch mass.

II. MADMEN Concept Summary

A. Overview

The concept of using small, modular mass driver landers to deflect an incoming asteroid was originally conceived of and defined in late 2003 and early 2004 by a team at SpaceWorks Engineering, Inc. (SEI). The original concept development effort was funded under a Phase 1 study award from the NASA Institute of Advanced Concepts (NIAC) concluding with a final report that summarizes that study[11, 12]. The concept of small, modular mass drivers for asteroid mitigation is referenced by the acronym MADMEN (Modular Asteroid Deflection Mission Ejector Node). Artist's conceptions of the multiple mass driver-equipped lander approach are given in Figure 2. CAD layouts are shown in Figures 3.

The original MADMEN concept consisted of a transfer stage and a mass driver-equipped lander. Multiple units are deployed to deflect each target asteroid and operate as an intelligent swarm that works collectively to accomplish the deflection mission while being able to tolerate and compensate for individual spacecraft failures. Once put into a rendezvous position by the transfer stage (matching the heliocentric velocity of the target), the lander has a small amount of onboard propulsion to aid its descent and landing on the asteroid surface. Mechanical anchor barbs at the base of each landing pad are fired into the asteroid and used to secure the lander to the surface. Note that this approach assumes that the asteroid can support mechanical shear forces and that the anchor is effective in securing the machine to the asteroid's surface. Once attached firmly to the surface, the lander is not mobile. Given the rotating nature of an asteroid and the dusty surface environment, the lander is to be powered by a small nuclear space reactor (< 50kW$_e$). Large radiators (relative to the spacecraft) are employed to reject waste heat originating from on-board subsystems including the power reactor.

The primary components of the lander spacecraft include the drilling/mining mechanism and the mass driver mechanism. The drill is envisioned to be a multi-segment coring drill with a rotary drill bit head. The specific dimensions of the drill vary with the particular mitigation scenario, but the core hole diameter typically ranges from 1 - 8 cm. Drill depth also varies with application, but is typically 3 - 6 m. The drill is mounted on a rotary base allowing it to rotate and radially translate to evacuate multiple holes beneath the base of the lander. The specific number of holes depends on the diameter of the lander base, the diameter of the holes, and an approximate solution to the hole-packing algorithm. For a large base diameter and small core hole, the total area of the smaller holes is roughly 86% of the surface area covered by the lander base. The drill is assumed to advance at a rate of roughly 3 cm/minute. Assuming that drilling power is roughly 230 W for each cm/minute of drilling rate, 690 W are required to sustain this rate[13].

Drilling is considered a near-constant operation while the MADMEN lander is on the surface. Asteroid material removed from the core tubes is gathered into uniform mass units to be conveyed to and then ejected by the mass driver. The size and mass of the ejected shots varies depending on the specific mitigation scenario. In exploration of preferred configurations throughout the design space, shot masses below 1 kg and velocities below 750 m/s are typical.

The mass driver is a deployable tower consisting of a series of electromagnet rings and a supporting structural strongback. The mass driver is automatically erected to its full height once the lander is on the surface. The strongback allows the mass driver to be pointed in an appropriate direction within roughly a 30° half-angle cone of operation. The size (length and diameter) of the mass driver depends on the specific sizing case. The mass driver consists of an acceleration segment for the shuttle bucket and the ejecta material and a deceleration segment to recover the bucket. The acceleration segment is assumed to occupy roughly 65% of the mass driver's length. This analysis typically constrained the mass driver length to be no more than 15 m for practical deployment reasons.

The mass driver operates according to a schedule or "duty cycle" that is determined by the frequency that the mass driver viewing angle can be brought into alignment with the firing vector - generally opposite the asteroid's heliocentric velocity. For an irregular and rotating asteroid, a duty cycle of only 15% is assumed for any individual lander. When operating, the mass driver is powered by a bank of capacitors that are recharged between shots by the nuclear power source. For a typical configuration, the firing time is extremely short (fraction of a second) allowing a significant step-down in the power demand on the reactor versus the much higher capacitor power required to shoot material (an ejecta shot). Power conversion losses of 15% are reflected in the model for this recharging process. For much higher shot frequencies, the reactor power requirement approaches the mass driver power, but shot frequencies of only 1 to 4 shots per minute (when active) were found to be typical.

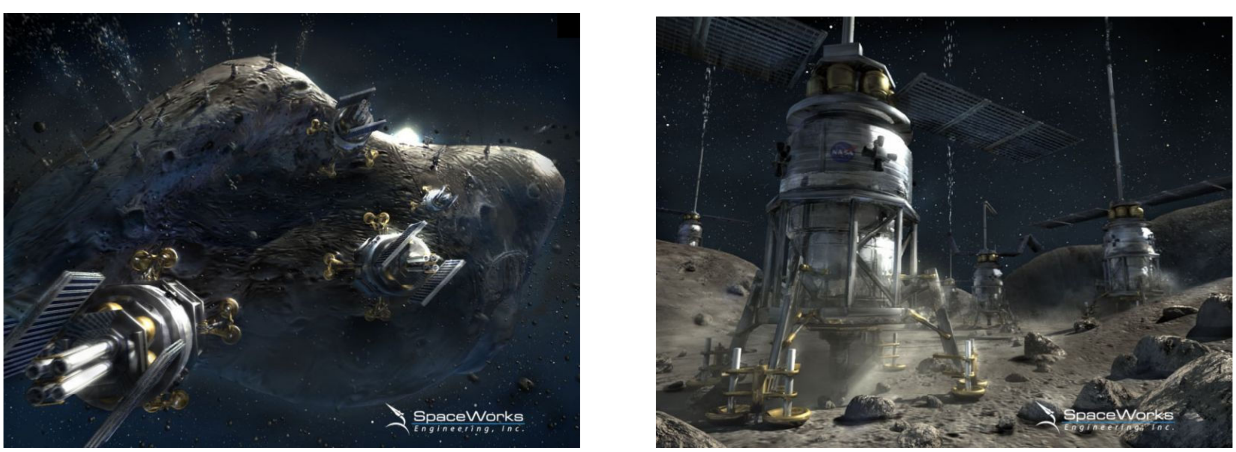

Figure 2. Artist's Conception of the MADMAN Proximity and Surface Operations [11].

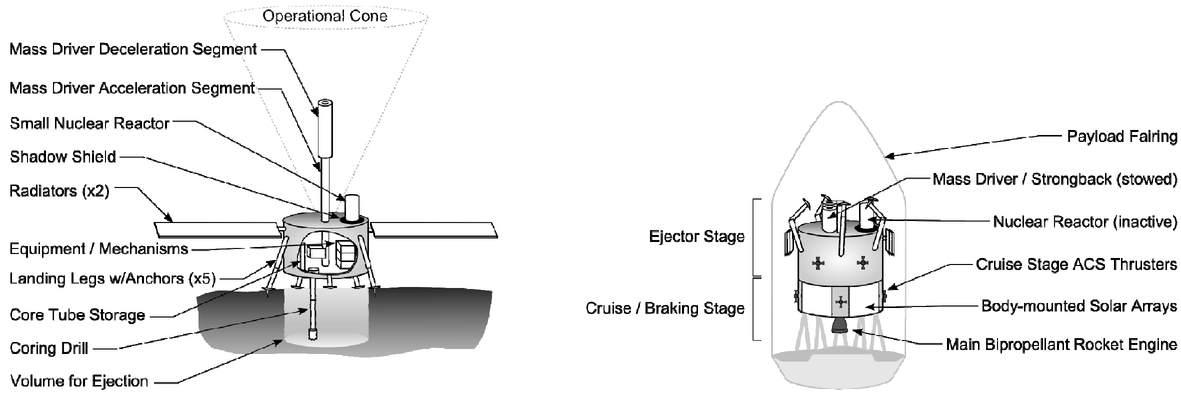

Figure 3. Layout of the MADMEN Mass Driver Lander on the Surface and Stowed in the Launch Vehicle Fairing (Shown for a Single Launch Solution)

B. Changes/Updates to Baseline Concept

Since the original 2004 NIAC-sponsored work on this concept, several revisions and improvements have been made to the initial assumptions and design decisions regarding the MADMEN concept. Some of the improvements include the following:

1) Less Generous Deflection Distances: In the original concept development work, it was assumed that a deflection mission would be required to change the miss distance between Earth and a threatening asteroid to one-half of the Earth-Moon distance (about 192,500 km). Upon further consideration, five Earth radii (5 R_e) is used as a reasonable miss distance (about 32,000 km measured from the center of the Earth). In the future, this perturbation distance should more properly be the result of a probabilistic calculation depending upon initial errors in the asteroid's position and uncertainties in the deflection process.
2) Longer Warning Times Allow Direct Launch: The original concept assumed that a very fast response would be needed to deflect an asteroid threat discovered with a very short lead times (a year or less) and therefore pre-staging the swarm of mass drivers in space would be required. As NASA completes more of the Spaceguard survey and more of the NEO population is documented, it is becoming much more likely that a large asteroid (140 m or greater) would in fact be discovered soon enough to allow many years of lead-time (advanced warning time) within which to mount a deflection mission directly launched from the surface of the Earth.
3) Simpler Cruise Stage: By launching directly from Earth using an appropriate launch vehicle to place the MADMEN lander on a transfer orbit, the in-space stage velocity change requirement can be reduced to a single arrival ΔV and some small mid-course correction maneuvers. The previous analysis was pre-deployed into Earth-Moon L4 or L5, and therefore required a larger total velocity change to reach a target asteroid. Using a direct launch to provide the initial Earth escape maneuver allows the replacement of the cryogenic transfer stage in the original concept with a simpler cruise stage in the current concept. The current cruise stage is fueled by hypergolic propellants and powered by body-mounted solar arrays. It is discarded after the asteroid rendezvous ΔV is applied.
4) Longer Surface Action Times: The original 2004 study suggested that mass driver operations at the asteroid should be limited to 60 days or less in order to satisfy public anxiety and yield a short action time after which the outcome of the deflection mission would be known. It is recognized that a significant advantage of this concept is that it can provide a much larger impulse (for a given launch mass) if allowed to operate for a long time. Given new assumption of longer warning times, surface operations are now allowed for periods up to one year or greater. Of course, long surface operations times add additional demand on the lander's hardware components given the relatively hostile operating environment.

C. MADMEN Parametric Sizing Model

A ROSETTA (Reduced-Order Spreadsheet for Estimating Technologies and Transportation Architectures) model, a type of parametric spacecraft sizing model[14], of the MADMEN concept has been created in Microsoft Excel© and has been used to conduct a series of studies to define key variables. The ROSETTA model consists of several interconnected worksheets within an overall workbook. Sizing sheets exist for the mass driver, the power system, the lander spacecraft, the cruise stage, and the summary vehicle. Key input variables include the target asteroid's mass, the required deflection ΔV, the lander size (base diameter), the mass driver length, the shot frequency, the shot mass (per individual shot), the shot exit velocity, the drill/coring rate, the duty cycle on the surface, the ΔV for rendezvous, the launch vehicle capacity to the required C_3 for Earth departure, and the number of landers employed on the mission.

Key outputs of the model include the required action time on the surface, the lander gross mass, the lander dry mass, the cruise stage mass, the total launch mass, the reactor power required, the coring hole depth (per hole), the coring hole diameter, and various subsystem masses and dimensions. These data are subsequently used to estimate the design development, test & evaluation (DDT&E) cost, Theoretical First Unit (TFU) cost, production costs, launch costs, mission operations costs, and overall mission reliability.

Life cycle costs are estimated using historically-based equations and analogous costs for similar space systems. These specific equations for DDT&E and TFU cost are based upon parametric historical cost estimating relationships (CERs) that scale for both mass and complexity. The top-down CERs used for this study have been correlated to historical spacecraft cost data and have been proven useful to estimate the costs of future space systems. The Life Cycle Cost (LCC) consists of overall technology development, DDT&E and acquisition costs for both the in-space cruise stage and each MADMEN lander, in-space operations costs, facilities development costs, and launch costs. Overall technology development cost, the cost to bring required technologies to a Technology Readiness Level[15] (TRL) of 6, is estimated to be \$100 M (FY2007). This is a general estimate for required concept technologies such as the mass driver, small nuclear power source, etc. In addition to hardware development and acquisition costs, additional "wraps" are also included in the calculation of DDT&E and TFU costs. These costs are normally a substantial part of overall costs and should not be excluded in any LCC analysis. For this analysis, these costs included System Test Hardware (STH) costs (nominally 35% of TFU cost) and overall systems integration costs that consist of the following (30% of hardware + STH costs): Integration, Assembly, & Checkout (IACO); System Test Operations (STO); Ground Support Equipment (GSE); System Engineering & Integration (SE&I); and Program Management (PM). On top of these costs there are additional percentages to account for contingency (20%), contractor fees (5%), program support (5%) and stage integration (2%). Contingency is added here since the base hardware costs calculations are based upon dry mass without margin. In-space operations costs are broken out into cruise phase and target operations costs per month. These costs are estimated based upon NASA operations costs as listed in both previous and projected NASA fiscal year budget documents for the Dawn, Deep Impact, Phoenix, and Mars Science Lab (MSL) missions[16]. Cruise phase operations costs (approximately \$0.79M / cruise month, FY2007) are based upon the first three missions and surface operations costs (assumed to be more complex) are based only upon MSL operations estimates (approximately \$4.78M/ mission month, FY2007), MSL being representative of a mission with multiple mission phases and science instruments. These estimates are taken to be rough but relevant approximations of such costs. Launch costs for the standard Falcon 9 launch vehicle is assumed to be \$30M while launch costs for NASA's Ares V is assumed to be approximately \$500M. Ancillary facilities development costs are estimated to be \$10M. Given that the cruise stage is based upon known design and manufacturing processes, a learning curve effect percentage of 95% is assumed for the cruise stage and 90% for the mitigation (MADMEN) stage.

Mission reliability ranges are estimated for the in-space segments only. Based on previous work by the authors, individual MADMEN spacecraft are estimated to have mission completion reliabilities in the 80% - 90% range, once they are successfully launched[11]. Overall mission reliability is simply estimated based on the number of spacecraft in the swarm and the number of failures that can be tolerated while still ensuring that the deflection is successful. This varies with the specific mitigation strategy and the "nominal" surface action time. For most of these preliminary analyses, it is assumed that up to one-half of the original landers can fail and the mission could still be accomplished simply by extending the surface action time of the remaining landers.

For a given set of user inputs, the ROSETTA model iterates among over 200 equations that estimate masses and power requirements for individual subsystems. The ROSETTA model's behavior is reasonably complex. For example, changing the shot velocity will directly increase the momentum imparted with each shot and decrease the required action time on the surface. However, the power requirement increases and thus the reactor and the radiators increase. In addition, the downward load on the mass driver strongback increases due to the increased acceleration required to shoot the material. This increases the mass of the strongback. Alternately, a decrease in the length of the strongback will reduce its size and buckling length, and therefore reduce the mass of this component. However, a shorter mass driver requires a larger acceleration to reach a desired velocity and therefore increases the load on the main spacecraft structure and the landing legs. In addition, more power is required. Conversely, increasing the shot frequency will decrease the surface action time proportionally, but has a negative effect on the mining subsystem. For a given shot mass and frequency, the drill/core unit must be designed to keep up with the required mass flow (while making proper allowances for the duty cycle effect - the mass driver is only operating for a fraction of the mission time, while the drill is operating nearly continuously). As a rule, as the shot frequency increases, the drill diameter increases and therefore the mass of the drill and coring tubes increases. Operating on the surface longer will increase the core hole depth and thus the mass of the drill tubing.

For a sizing problem with constraints on the launch mass, spacecraft diameter, and the maximum allowable hole depth, the complex interactions among the independent variables and the subsystems create a difficult non-linear and non-smooth optimization problem. Pi Blue Software's OptWorks: Genetic Algorithm has been coupled to the MADMEN ROSETTA model to aid in finding near-optimal solutions to various hypothetical mission scenarios. For a given scenario (asteroid mass, required ΔV, rendezvous date, mission constraints), optimization usually takes a few minutes on a standard PC or Macintosh computer. Using this approach, this analysis has explored hundreds of sizing cases for the MADMEN concept across a range of various sized asteroids with different mission warning times, deflection ΔV's, and launch vehicle constraints. The findings indicate that, with properly selected independent sizing variables, a mass driver solution delivers as much or more change in momentum to a target asteroid than competing non-nuclear explosion methods.

Figure 4 shows how a generic mass driver solution would scale to meet various deflection scenarios. Effective Momentum change is defined as the mass of the target asteroid times the required change in asteroid velocity. For comparison, data from other concepts investigated in NASA's Near Earth Object (NEO) Analysis of Alternatives (AoA) study is also shown[1]. Our analysis did not conduct an independent assessment of the alternative deflection techniques show on this chart. The options other than the mass driver solution are simply taken from NASA's NEO AoA Study and plotted for comparison[1]. Only non-nuclear detonation options are shown. In the NASA NEO AoA Study, the effectiveness of a nuclear standoff option is shown to be superior to all other choices, but does have political concerns. In Figure 4, it is considered "better" to yield a higher effective momentum change for a given launch mass (higher lines are better).

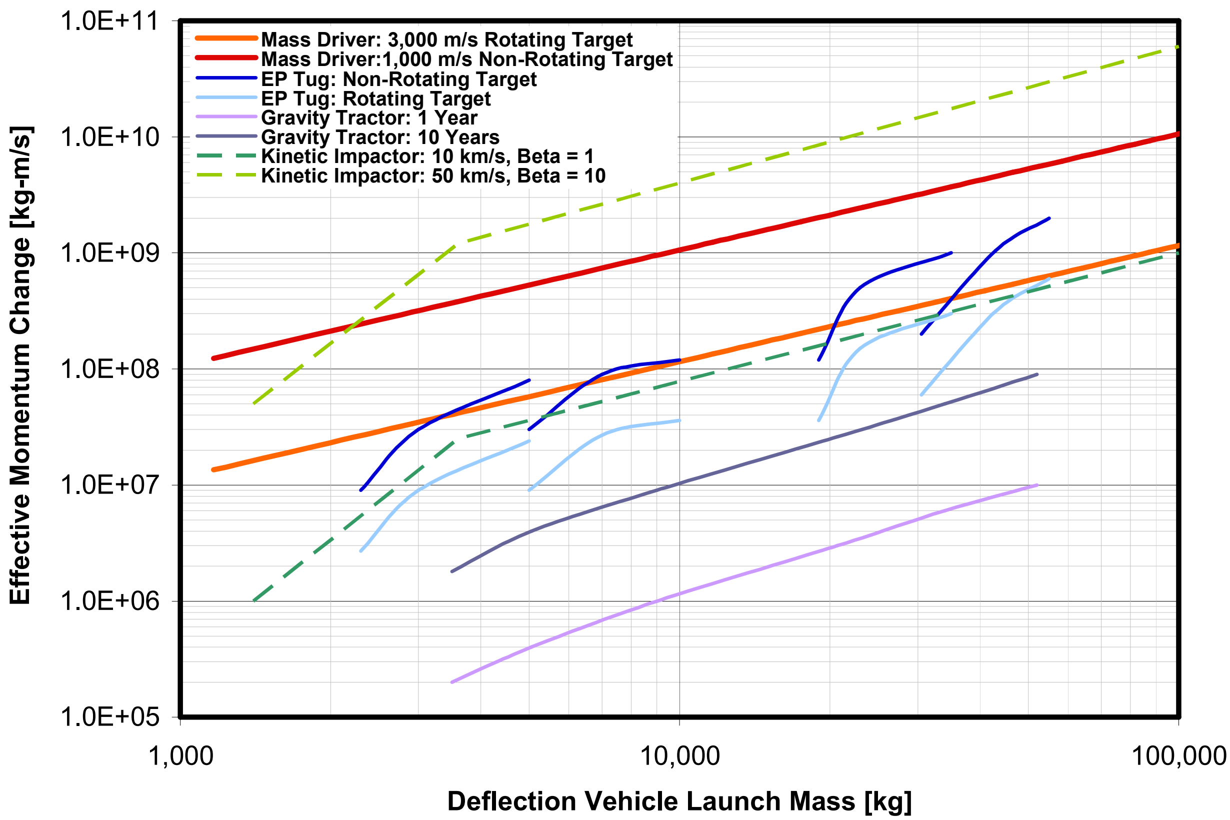

Figure 4. Top-Level Performance Comparison with Alternate Deflection Techniques (non-nuclear).

The two mass driver trend lines shown on this figure reasonably bound the resultant data that we produced in our exploration of the design space. The lower line represents a cruise stage sized for 3 km/s of arrival ΔV at the target asteroid and a duty cycle of 15% (our standard assumption). An arrival ΔV of 3 km/s is typical for a near-Earth object, but this number is ultimately dependent on the launch date and transfer orbit geometry. The upper line represents a more optimistic scenario that requires only 1 km/s of arrival ΔV from the cruise stage and has a duty cycle of 100% (an approximation of a non-rotating asteroid). Action times are one year on both cases. Note that even the "conservative" (lower line) MADMEN mass driver solution is clearly superior to the gravity tractor and the propulsive tug (the latter applied to a more realistic rotating target). Based on this analysis, its performance is also comparable to the asteroid tug applied to a non-rotating target (perhaps an unrealistic scenario) and the kinetic impactor with a 10 km/s impact velocity and a Beta (momentum amplification factor) of 1.0. The highest performing concept on this scale is the kinetic impactor with a strike velocity of 50 km/s and a Beta of 10. One could easily challenge the realism and the practicality of such an approach. In the end, it is concluded that the mass driver approach has quantitative performance advantages when compared to competing alternatives in a general application.

III. Deflection Case Studies

Two relevant case studies are presented in detail here. These cases are chosen to represent typical responses that may be mounted to deflect a "small" incoming asteroid (100 m - 300 m in diameter). Statistically, there are orders of magnitude more Earth-crossing asteroids in this size range than those have diameters of 1 km or more. Most of the undiscovered asteroids lie in the smaller size ranges as well. Each case study includes a selection of launch vehicle, launch and arrival conditions, and an assessment of required deflection ΔV.

A. Case Study 1: D'Artagnon

D'Artagnon is a fictional asteroid originally created in support of the 2004 Planetary Defense Conference[17]. It is a 130 m diameter type S stony asteroid with a high rotational period that was created to be an example of a low warning time impactor (about 5 years from discovery to impact). In order to make the case relevant for advanced concept development, some of the parameters of the orbital elements have been adjusted to reflect a discovery date of January 1, 2007 and an impact date of April 1, 2022. Semi-major axis, eccentricity, and inclination were preserved from the original dataset, but position angles were adjusted to reflect the new dates of discovery and impact. The "modified" D'Artagnon orbital characteristics are given in Table 1. All data is shown in the J2000 heliocentric-ecliptic reference frame.

Table 1. Target Asteroids for Selected Case Studies.

Parameter	Values for Case Study	Values for Case Study
	D'Artagnon (modified)	Apophis
Minor Planet Center (MPC) Designation	fictitious	99942 (2004 MN4)
Semi-major axis (a)	0.90220435 AU	0.92226142 AU
Eccentricity (e)	0.30245951	0.19105942
Inclination (i)	4.78620700°	3.33131464°
Longitude of Ascending Node	191.15627122°	204.45915230°
Argument of Periapsis	227.57988257°	126.38557131°
Mean Anomaly Angle	27.28864277°	307.36307853°
Epoch	January 1, 2017 0.0 UT	April 10, 2007 0.0 UT
Mass	2.7×10^9 kg	2.1×10^{10} kg
Diameter	130 m (irregular)	250 m (estimated from magnitude)
Asteroid Type	S-type (stony)	S-type
Density	3.0 g/cm^3	2.6 g/cm^3 (assumed)
Asteroid Class	Aten	Aten
Spin Period	19 minutes	30.54 hours
Impact Date	April 1, 2022	unknown

The orbit of D'Artagnon is normally propagated forward in time using an 8th/9th order n-body numerical propagator with a variable step size. The sun, all of the planets and the Earth's moon are considered in the gravitational model. The perturbing effects of the large asteroid-belt asteroids Ceres and Vesta are being added, but were not present in the current analysis. Solar pressure and the Yarkovsky effect are not modeled. The standard propagator is based on the Java Astrodynamics Toolkit (JAT) originally developed at the University of Texas.

The original JAT propagator was modified by SpaceWorks Engineering, Inc. (SEI) and our partner researchers at the Georgia Institute of Technology to accommodate a low thrust perturbation model. For a given mass driver ejection rate, shot velocity, and shot mass, a perturbing force is approximated as low, constant force applied in the in-track velocity direction. For a given start date, a search algorithm is used to determine the minimum number of days over which the force needs to be applied in order to change the minimum Earth miss distance by the desired amount. For D'Artagnon, it is assumed that the miss distance in 2022 would have to be at least five Earth radii. Once the minimum surface action time is computed for a given start date, the start date is incremented by 30 days and the process is repeated to create a family of solutions. For a range of solutions over a candidate period of 5 – 10 years, the algorithm takes several hours to execute on a standard PC.

The "integrated" ΔV at each point is calculated by summing the ΔV created from each mass driver shot across the total number of shots required to achieve the new miss distance. It has been found that the "integrated" ΔV is a reasonably constant parameter even when the shot velocity, shot frequency, and shot mass are varied by +-25% during the optimization process. The ΔV requirements as a function of the date of initial surface operations are shown in Figure 5. The impulse solution (the "instantaneous" ΔV) is shown on the same figure for comparison. For reference, five landers, a shot frequency of three shots per minute per lander, a duty cycle of 15% per lander, a shot mass of 0.75 kg, and a shot velocity of 750 m/s was used to produce the integrated ΔV chart. Using these parameters, the underlying action times resulting from this analysis ranged from 118 days at the low end to 525 days at the high end of the date range.

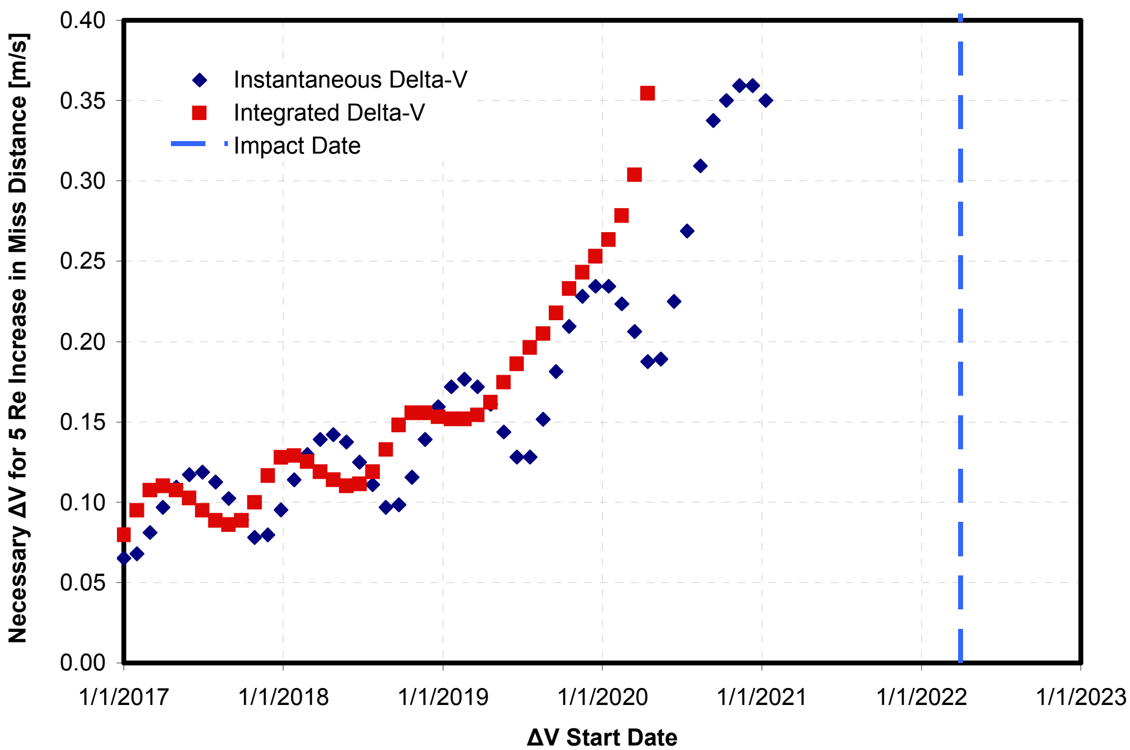

Figure 5. Integrated ΔV vs. Start Date of Surface Operations for D'Artagnon

Note the periodicity in both the integrated ΔV and instantaneous ΔV solutions. This is the physical bias toward applying the ΔV near perihelion of D'Artagnon's orbit. Also worth noting is the relative advantage, in terms of reduced ΔV, of applying the perturbation well before the impact date. Early application of the perturbation allows changes in the asteroid's period to compound over several heliocentric revolutions and therefore increase the ultimate miss distance with the Earth in 2022.

The Ares V launch vehicle, NASA's proposed heavy lift vehicle currently under development in support of the U.S. human lunar exploration mission, was chosen as the launch vehicle for this case study. The Ares V is large enough to allow the deflection mission to be undertaken with a single launch, and the launch vehicle is expected to be very reliable. The maximum payload performance to Earth escape orbits was estimated using publicly available stage weights and propulsion performance data. Launch trajectories were simulated from Kennedy Space Center, Florida USA for due east launches. The vehicle's estimated performance is shown in Figure 6. C_3 is the square of the maximum residual velocity at escape (twice the specific energy of the escape orbit).

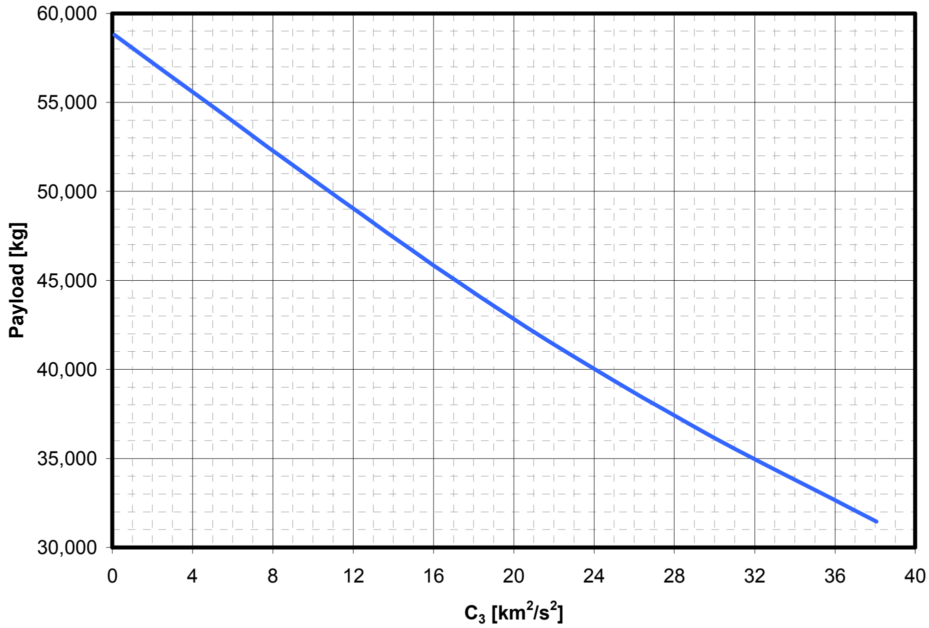

Figure 6. Predicted Escape Trajectory Performance of the Ares V (C3 vs. Payload).

For this case study, the launch C_3 was limited to 20 km^2/s^2 or less. This corresponds to a maximum launch capacity of approximately 43,000 kg. After a search of potential launch dates between 2017 and 2019, a transfer trajectory was identified that minimizes the arrival C_3 while limiting the departure C_3 to 20 km^2/s^2. The selected orbit is shown in Figure 7. Launch takes place on April 20, 2017. Arrival at D'Artagnon occurs on December 1, 2017. The transfer time is 225 days. The arrival C_3 is 8.737 km^2/s^2, requiring an arrival ΔV from each cruise stage of approximately 3,024 m/s (2,960 m/s plus a 2.5% margin). Gravity swingby trajectories were not considered in this solution. Note that within the time range considered for a launch response to this threat, a more optimal transfer alignment was not possible. With a longer warning time, the transfer to the asteroid might be achieved with a lower arrival ΔV.

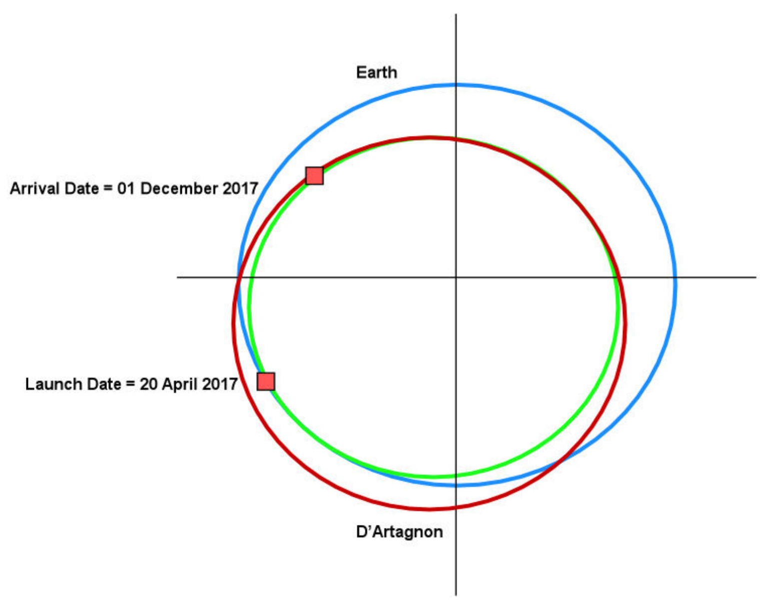

Figure 7. Selected Transfer Orbit to D'Artgnon.

Based on the arrival date and assuming a short rendezvous and landing phase at the asteroid, the required integrated ΔV to be applied to D'Artagnon in order to provide a miss distance of five Earth radii (32,000 km from the center of the Earth) is conservatively taken to be 12.5 cm/s from Figure 6.

This data was entered into the MADMEN ROSETTA sizing model and the parameters of the spacecraft and mass driver were optimized in order to minimize launch mass (increase payload mass margins) for a five-lander solution. Surface action time was limited to 365 days. The resultant solution for this test case is given in Table 2. For comparison, a four-lander solution for this test case results in a slightly larger and more expensive lander with a faster ejection velocity and higher shot frequency. The five-lander solution was preferred based on primarily on its higher level of redundancy. Figure 8 illustrates the breakout of estimated Life Cycle Cost (LCC) for this concept. The overall LCC is over $2.3B (FY2007), with a large percentage of this cost consisting of both DDT&E and acquisition costs for the mitigation stage. Another significant cost driver is launch cost (Ares V).

Table 2. Solution for D'Artagnon Case Study.

Parameter	Values for Case Study
Number of MADMEN Landers	5
Lander Dry Mass	1,470 kg
Lander Wet Mass	1,650 kg
Cruise Stage Dry Mass (without lander)	470 kg
Cruise Stage Wet Mass (without lander)	4,180 kg
Total Launch Mass (each, with payload adapter)	5,830 kg
Total Launch Mass (all, with payload adapters)	30,600 kg
Ares V Launch Vehicle Capacity (to C_3 = 20 km^2/s^2)	43,000 kg
Launch Margin (allowable payload growth)	40%
Mass Driver Shot Frequency	3 shots per minute
Mass Driver Ejecta Shot Mass (per shot)	0.50 kg
Mass Driver Shot Velocity	570 m/s
ΔV per shot	1.056 E-07 m/s
Spacecraft Base Diameter	3.5 m
Mass Driver Rail Length (Deployed)	15 m
Acceleration Segment Length	65% of total length
Drill/Core Hole Diameter	5.64 cm per coring hole
Drill/Core Hole Depth	4.70 m per coring hole
Drill/Core Rate	3 cm per minute
Reactor Power (kW_e/KW thermal)	16.5 kW_e/ 92 kW_{th}
Duty Cycle (% of time in firing alignment)	15%
Nominal Surface Action Time	365 days
Individual Lander Reliability	80% - 90%
Post-Launch Mission Reliability (assume 3 of 5)	99.38% - 99.95%
MADMEN Lander Spacecraft DDT&E Cost (mitigation stage only)	$723.2M (FY2007)
MADMEN Lander Spacecraft TFU Cost (mitigation stage only)	$118.5M (FY2007)
Mission Life Cycle Cost (including launch)	$2,255.5M (FY2007)

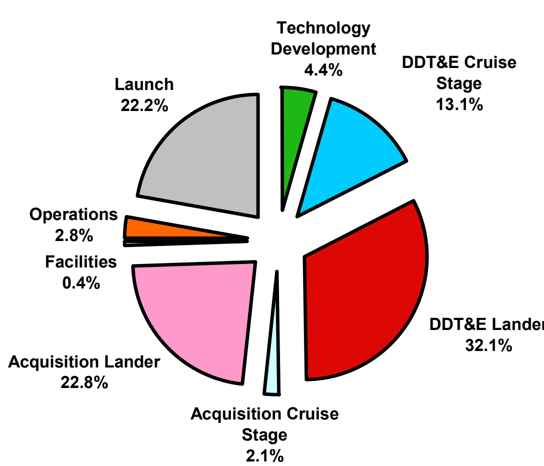

Figure 8. Life Cycle Cost (LCC) Breakout for D'Artagnon Case Study.

B. Case Study 2: Apophis

Asteroid 99942 Apophis is a unique PHA having a close approach to Earth in 2029. As a result of that approach, there is a small chance that Apophis will have a resonant return to Earth in 2036 that will result in an impact[18]. A small keyhole, roughly 600 m long in the major axis, exists in the b-plane (Earth-centered plane normal to the incoming earth-relative velocity at the edge of the gravitational sphere of influence) of the April 13, 2029 close approach pass. If Apophis passes through that region of near-Earth space, then there is nearly a 100% chance that it will hit the Earth in 2036[19]. Conversely, if Apophis' trajectory can be altered to miss the keyhole, then the chance of hitting Earth in 2036 will be zero. Astronomers currently do not know definitively whether Apophis will or will not pass through the 2029 keyhole. There is only a small probability that it will (currently 1 in 45,450), based on trajectories propagated from current observations of its position at various times and the associated uncertainties in those observations. The probability of passing through the keyhole will be updated as new observations are made. Technical specifications of the orbit of Apophis as well as its physical characteristics were given in Table 1.

This case study was created under the assumption that Apophis is indeed predicted to pass through the 2029 keyhole. This mission requires a much smaller and less massive mass driver solution than that proposed for Case Study 1. Here, a very small deflection of the asteroid's 2029 flyby is sufficient to avoid future impact. For conservatism, it is assumed that an increase of 60 km in the closest approach to Earth during this flyby is sufficient to completely avoid the 600 m keyhole in the b-plane.

Figure 9 shows the results of a sweep of the date of initial surface operations vs. the integrated ΔV required to deflect the asteroid from the keyhole. The orbital propagation tool was again used at each time step to find the number of days that a surface mass driver would need to operate to achieve the required change in Earth miss distance. As in Case Study 1, this data was converted to integrated ΔV for plotting (sum of all of the individual shot ΔVs). For reference, a two lander mission solution was assumed for this trajectory analysis with a shot mass of 0.15 kg, a shot frequency of 2 shots per minute per lander, an ejection velocity of 150 m/s, and a duty cycle of 15%. Before converting to integrated ΔV, the number of days required to deflect Apophis ranged from 75 days at the low end to 420 days at the high end of the date range.

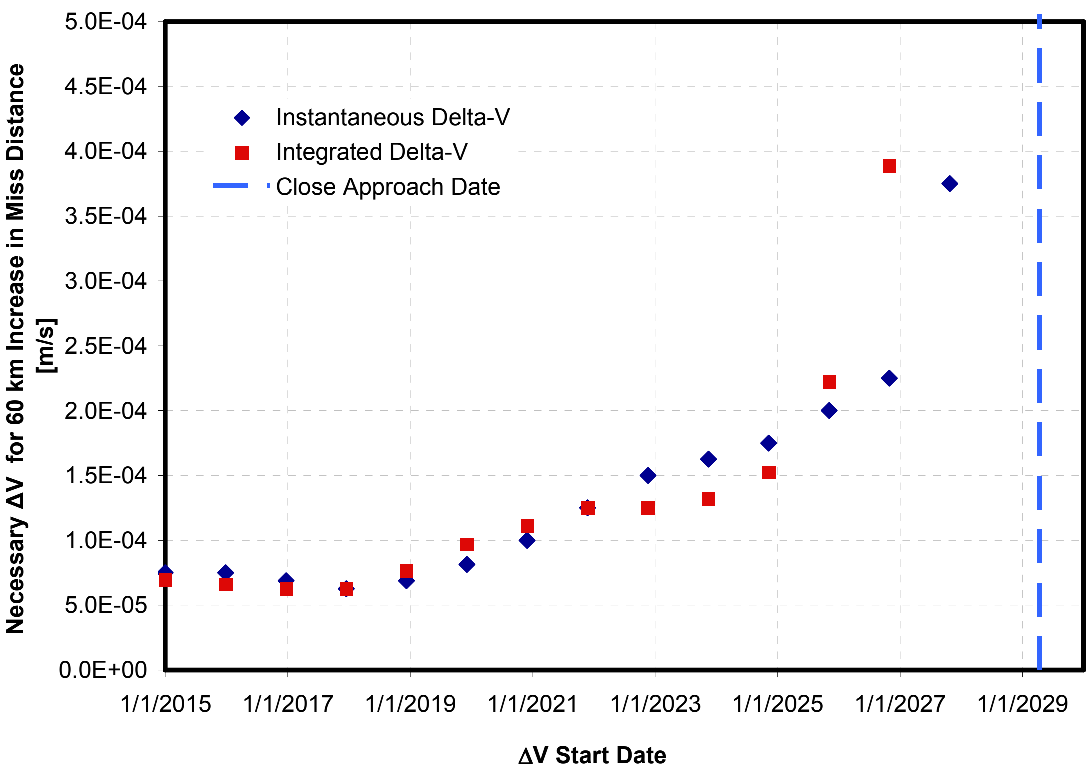

Figure 9. Integrated ΔV vs. Start Date of Surface Operations for Apophis (Keyhole Deflection).

The notional MADMEN mass driver deflection mission proposed requires only two smaller landers. Each lander is launched separately on a Space Exploration Technologies (SpaceX) Falcon 9 rocket (standard Falcon 9, non-heavy variant)[20]. Since the Falcon 9 is currently still under development, a launch trajectory simulation for that rocket was developed based on public information on its size, mass, and propulsion performance. The resultant payload capability to Earth departure orbits is shown in Figure 10. Launch is assumed to be from Cape Canaveral Air Force Station, Florida, USA to a direct due east departure orbit. For this case study, departure C_3 was limited to no more than 15 km^2/s^2. This corresponds to a launch capacity of approximately 1,350 kg based on this analysis.

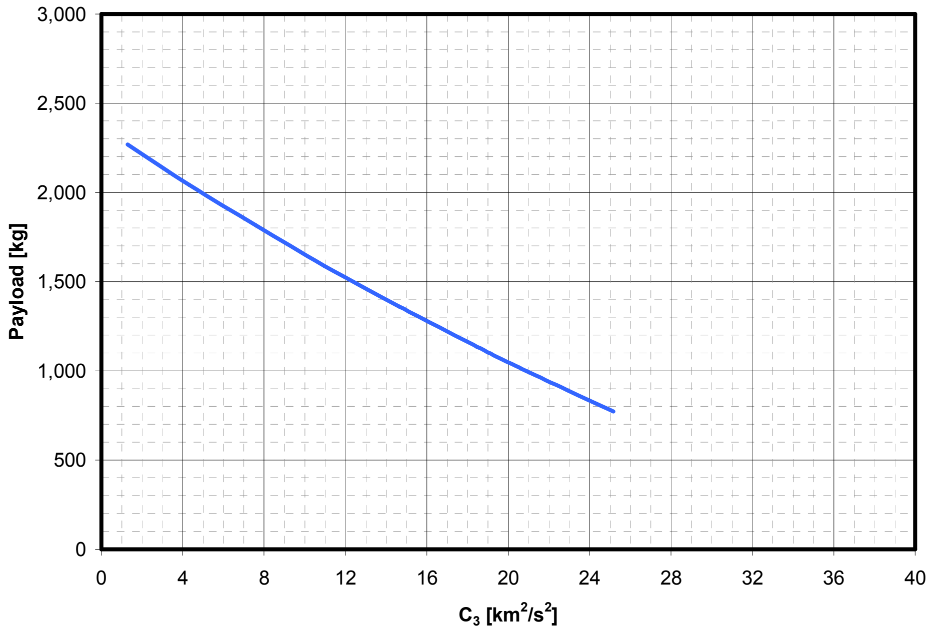

Figure 10. Predicted Escape Trajectory Performance of the SpaceX Falcon 9 (C_3 vs. Payload).

For Apophis, there are two good opportunities for efficient transfer from Earth to the target asteroid in the next decade or so. One opportunity exists in the 2012/2013 timeframe. The other exists in the 2022/2023 timeframe. Despite the relatively lower deflection ΔV requirement available from an earlier rendezvous, the later window was selected in order to allow a longer period over which to mature the technologies and techniques required for this concept. A transfer trajectory was chosen that minimizes arrival ΔV (or C_3) subject to a 15 km^2/s^2 limitation on departure C_3. Only direct missions without gravity flybys were considered. Transfer times were limited to 450 days or less.

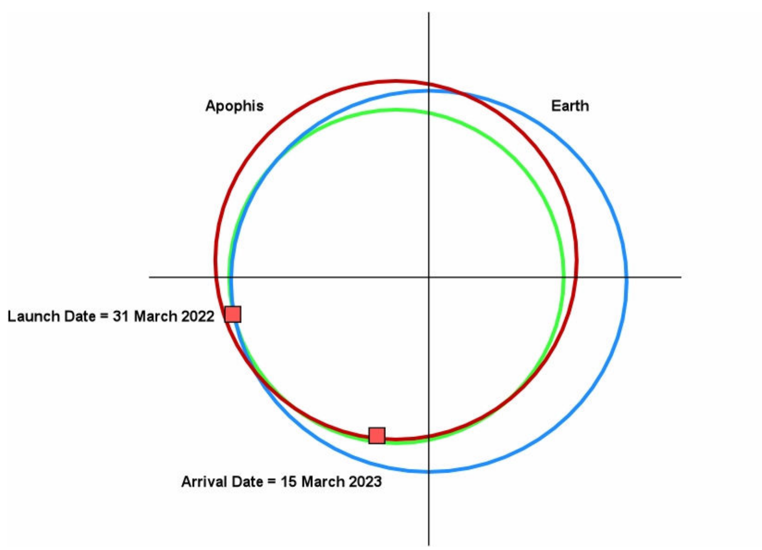

Figure 11. Selected Transfer Orbit to Apophis.

Launch of both missions is assumed to be approximately March 31, 2022. Rendezvous with Apophis takes place on March 15, 2023 after one full revolution about the sun (see Figure 11). Transfer time is 349 days. Arrival C_3 is only 1.611 km^2/s^2 requiring a rendezvous ΔV from each cruise stage of approximately 1,265 m/s. Assuming a short phase for rendezvous, landing and setup of surface operations, the required ΔV to be applied to Apophis is taken to be 1.3E-04 m/s (only 0.13 mm/s) from Figure 9.

For this case study, the required deflection ΔV is very small and easily within the capability of a single MADMEN lander operating over a period of one year. However, the MADMEN mission philosophy favors solutions with natural redundancy. Therefore, a two-lander solution is proposed as a better alternative. For this case, a Genetic Algorithm coupled with the MADMEN ROSETTA model was used to optimize the configuration so that there is a reasonably short surface action time requirement, while maintaining a generous payload weight growth margin for the selected Falcon 9 launch vehicle. The resultant solution is given in Table 3. For the Apophis case study, the low cost Falcon launchers provide significant savings to the overall mission. Figure 12 illustrates the breakout of estimated Life Cycle Cost (LCC) for this concept. The overall LCC is approximately \$815M (FY2007), with a large percentage of this cost consisting of DDT&E costs for the mitigation and transfer/rendezvous stage. These costs are substantially less than the first case study given the lower number of required MADMEN lander spacecraft (two versus five), the lower dry mass of each lander spacecraft (405 kg versus 1,470 kg), and lower launch costs (\$60M for two Falcon 9 launches versus \$500 M for one Ares V launch). The total LCC for the MADMEN spacecraft for an Apophis case study is approximately equivalent to the cost of the two Mars Exploration Rovers (MERs).

Table 3. Solution for Apophis Case Study.

Parameter	Values for Case Study
Number of MADMEN Landers	2
Lander Dry Mass	405 kg
Lander Wet Mass	455 kg
Cruise Stage Dry Mass (without lander)	155 kg
Cruise Stage Wet Mass (without lander)	485 kg
Total Launch Mass (each, with payload adapter)	940 kg
Total Launch Mass (all, with payload adapters)	N/A (single lander launch)
Falcon 9 Launch Vehicle Capacity (to C_3 = 15 km^2/s^2)	1,350 kg
Launch Margin (allowable payload growth)	37%
Mass Driver Shot Frequency	2 shots per minute
Mass Driver Ejecta Shot Mass (per shot)	0.15 kg
Mass Driver Shot Velocity	150 m/s
ΔV per shot	1.0714 E-09 m/s
Spacecraft Base Diameter	1.25 m
Mass Driver Rail Length (Deployed)	3 m
Acceleration Segment Length	65% of total length
Drill/Core Hole Diameter	2.52 cm per coring hole
Drill/Core Hole Depth	2.85 m per coring hole
Drill/Core Rate	3 cm per minute
Reactor Power (kW_e/KW thermal)	1.5 kW_e/ 8.6 kW_{th}
Duty Cycle (% of time in firing alignment)	15%
Nominal Surface Action Time (both landers)	140 days
Individual Lander Reliability	80% - 90%
Post-Launch Mission Reliability (assume 1 of 2)	96% - 99%
MADMEN Lander Spacecraft DDT&E Cost (mitigation stage only)	\$317.7M (FY2007)
MADMEN Lander Spacecraft TFU Cost (mitigation stage only)	\$48.8M (FY2007)
Mission Life Cycle Cost (including launches)	\$815.2M (FY2007)

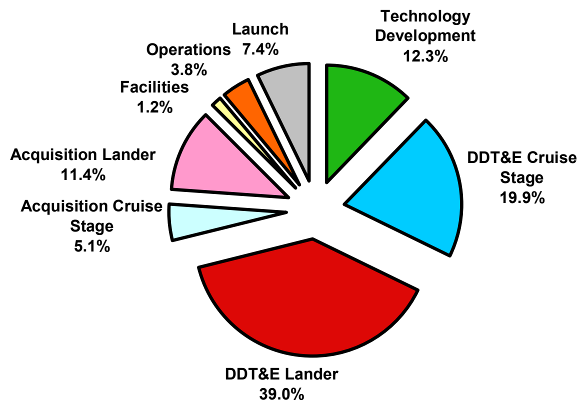

Figure 12. Life Cycle Cost (LCC) Breakout for Apophis Case Study.

C. Case Study Observations

Use of the ROSETTA model yields valuable insights into the nature of the baseline design solutions. Note that for both case studies, a relatively low duty cycle of 15% was assumed in order to account for the effects of a rotating asteroid. The duty cycle is defined as the percentage of time that the mass driver is in alignment to fire shots (recall that mining operations take place nearly continuously). Depending on the spin rate and spin axis, this may or may not be proven to be a good assumption. As noted, the steerable mass driver strongback moves to bring the firing direction into alignment with the negative velocity vector. Further geometric analysis is necessary to better define this variable. As a general rule, the use of more MADMEN landers on a particular target reduced their individual mass, but not necessarily the total launch mass. The use of more landers reduces the nominal work load on each lander, enabling it to use a lower shot velocity and/or a lower shot mass. Hole depths are shallower, thereby reducing drill/core requirements. Except as limited by packaging, longer mass driver lengths are generally preferred by the optimizer. This reduces power requirements and downward load effects on the lander structure, albeit at the expense of higher structural loads on the mass driver strongback due to increased characteristic buckling length. A larger base diameter on the spacecraft will increase the spacecraft bus mass, but will allow the drill to address more surface area of the asteroid. Recall that the drill is able to move radially and circumferentially on a rotating platform under the base of the mass driver lander. For a problem where the hole depth is constrained (as in these case studies), the ability to drill more holes will reduce the depth of each individual hole. However, this effect must obviously be limited by the launch vehicle fairing diameter and other practical limitations.

As might be expected, there is a tradeoff between shot velocity, shot mass, and shot frequency. A given increase in the change in asteroid momentum can be achieved by raising any one of the three. However, each has its penalties. Increased shot velocity impacts the mass of the mass driver. Increased shot mass and/or frequency increases demands on the mass driver, the power system, and the drilling system. Increasing shot mass and/or frequency also results in a deeper core hole (per hole), which may be undesirable. Our analyses have identified a broad and relatively flat design space with only a small sensitivity of lander wet mass with respect to these three interdependent variables, if they are selected judiciously. That is, as one was increased and another simultaneously decreased, the resultant effect on the lander wet mass could be minimized. Typical shot frequencies ranged from 1 to 4 shots per minute. Shot masses typically ranged from 0.15 kg to 1 kg. Preferred shot velocities were found to lie in the range of 150 m/s to 750 m/s. These solutions are obviously dependent on the settings of the other variables.

There is also a strong coupling between the launch vehicle payload capability, the transfer trajectory, and the launch mass of the deflection technique. In this research, the total launch mass was arbitrarily limited as a result of selecting a departure C_3 constraint. The search for transfer trajectories was limited to those direct transfers that meet the departure C_3 constraints and minimize arrival C_3. This approach minimizes ΔV demand on the cruise stage, but may not be optimal in the multidisciplinary sense. A lower C_3 at departure may in fact allow a larger escape mass and subsequently more than make up for the additional in-space ΔV required. This effect is expected to be dependent on the upper stage Isp of the launch vehicle. A higher Isp on the launch vehicle upper stage (as in the case of the Ares V), may favor a higher departure C_3. Given more time and resources, it may be useful to explore this complex interaction more. As an approximation, a larger allowable departure C_3 for the Ares V case was assumed versus the Falcon 9 case.

Similarly, there is an interdisciplinary coupling that exists between the heliocentric transfer solution and the amount of ΔV required to deflect the target asteroid. In general, an earlier arrival at the target asteroid will reduce the required ΔV needed to impart a given deflection. It is conceivable then that a faster transfer trajectory, one that does not minimize the arrival C_3 but arrives at the asteroid earlier, may be a more globally optimum solution than the approach taken here. This remains a topic for further investigation.

In terms of overall Life Cycle Cost (LCC), the nominal assumptions taken for the assessment indicate that each case study will cost several hundred million to several billion dollars (US, FY2007). The mass and number of each MADMEN lander spacecraft as well as the launch cost are major drivers for the LCC. These choices are coupled with the particulars of any mission design obviously. For the Apophis mission case study in particular, even using slightly more conservative assumptions than those used for this analysis, a practical NEO mitigation mission using multiple mass drivers can be accomplished for under $1B (FY2007).

IV. Practical Issues Associated with Fielding a Mass Driver in the Next Decade

The technical and political advantages associated with a mass driver solution were discussed earlier in this paper. However, the concept is not without challenges and other disadvantages. In the interest of completeness, some of the major challenges associated with the MADMEN multiple mass driver concept are discussed.

A. Attachment

For the mass driver concept to be feasible, it must be capable of firmly attaching itself to the asteroid's surface. A piton-like device with a cable is currently envisioned - explosively buried several meters into the asteroid's surface. Alternate mechanical attachment mechanisms might include anchor-like spikes or penetrating tubes with expanding subterranean sections. Each of these techniques would depend on shear forces to maintain the integrity of the anchor. If the asteroid is a loosely-attracted rubble pile, as some have speculated to be the case for smaller asteroids, then it may not be able to support any structural shear loads on its surface. Proper characterization of a target asteroid's structural properties will be required to determine the feasibility of mechanical anchoring. Loose rubble piles may not at all be an appropriate target for this type of deflection technique. As an alternative, new and novel anchoring techniques should be investigated.

B. Dust Environment

The drilling, coring, conveying of ejecta and mass driver surface operations will create a dusty operational environment for the MADMEN lander. An asteroid's gravity field is very low, preventing the dust from settling rapidly. In addition, the nature of the dust particles may be more abrasive than dust on the Earth given the absence of eroding effects. The specific impact of this dust on spacecraft mechanisms is currently unknown, but will surely present a challenge to operating a surface spacecraft for long periods of time.

C. Ejecta Dispersion

One obvious question with this concept is, where does the ejecta go? Over a typical mission, there are hundreds of thousands of shots of small bits of the asteroid shot from the mass driver at high speed. Each bit of ejecta will easily escape the asteroid's gravity well and enter its own, slower heliocentric orbit. This debris field could conceivably create a hazard for future space exploration or operating space assets. In current simulations for this concept, the individual shot masses are very small (less than 1 kg). Naturally occurring pieces of debris of this size may already number in the trillions, so adding a million more pieces of man-made ejecta would not statistically affect the nano-asteroid population. These pieces would not survive Earth entry in any case. Still, a study that seeks to understand their orbits and whether they are indeed Earth crossing would be welcomed. One approach might be to limit mass driver operations to only those segments of the asteroid's trajectory where the resultant ejecta would enter an orbit lying entirely within that of the Earth (aphelion less than 1 AU).

V. Technology Maturation Challenges

Critics of the multiple mass driver concept often suggest that it is too technologically challenging relative to other mitigation techniques such as nuclear detonation, kinetic impactor, and the gravity tractor. That may indeed currently be the case, but given the pace of technology development in space exploration and in other industries (robotics, mining, etc.) in general, it is not inconceivable that the key technologies required for this concept could be matured to flight readiness by 2015 with proper funding and urgency. Progress has already been made in key areas in spite of difficulties in obtaining monies for new technologies within the limits of the NASA and overall U.S. federal budgets.

A. Drill

Deep drilling and coring technology is mature for terrestrial applications, but unique challenges remain for application in a space environment. Cooling the drill, removing shavings, applying pressure to the drill bit, and automatically adding core tubes are difficult in a low gravity, air-less environment that may have 15 minutes of round trip communications lag time between Earth and the site of operations. Nonetheless, progress on the development of a deep coring drill for robotic Mars exploration is being made by NASA and commercial companies such as Honeybee Robotics[21, 22]. These developments are a step in the right direction toward an autonomous asteroid drill.

B. Small Nuclear Power Source

The key to the mass driver deflection concept is that the lander brings the power source with it, not the propellants. A small, lightweight, and reliable power source is needed to make the concept viable. Depending on the size of the nuclear reactor, the ROSETTA model's nuclear reactor sizing equations predict a specific mass (including shadow shielding and radiator) of between 105 kg/kW$_e$ (for small power needs of approximately 1.6 kW$_e$) to as low as 15 kg/kW$_e$ (for higher power demands of 16.5 kW$_e$). These ranges are in-line with current predictions for nuclear reactor specific mass.

The U.S. DoE/Lawrence Livermore National Labs has ongoing research programs investigating the feasibility of space-based nuclear reactors. This work has significant historical precedence in the SNAP reactors programs of the 1960's and the more recent SP-100 of the 1980's and 1990's. Two modern concepts, the HOMER-15 (Heatpipe-Operated Mars Exploration Reactor) heat pipe transfer reactor and the SAFE-400 nuclear reactor (Safe Affordable Fission Engine-400), if fully developed, would offer specific masses well within the range needed for both of the case studies offered for this analysis[23, 24, 25]. In addition, these two preliminary development efforts are targeting absolute electrical power ranges of 15 kW$_e$ and 100 kW$_e$, respectively. These design power levels could presumably be scaled down to meet the smaller requirements of the MADMEN lander.

C. Artificial Intelligence for Robotic for Swarm

As envisioned, the MADMEN concept would deploy multiple landers for more reliable and robust operations. These landers would each be autonomous, but would need to work in coordination with the other landers in the swarm to ensure that the mission is completed. They would have to be cognizant of failures within members of the group, off-nominal performance, lower than expected ΔV's, etc., and respond accordingly. Intelligence among members of a robotic group will offer significant flexibility and cost savings relative to ground operator-controlled and coordinated surface operations. Ongoing autonomous computing and swarm logic research is underway a variety of software development fields ranging from retailing to banking and gaming[26]. As these market-driven, non-aerospace applications mature, the results will be ready to be integrated into a future aerospace system such as the one envisioned here.

D. Mass Driver/Shuttle Bucket

Electromagnetic launchers, rail guns, and mass drivers continue to receive research attention for terrestrial applications, notably for long-range artillery and supercolliders[27]. Application of this technology to space launch is also under investigation. For in-space applications, researchers at the Space Studies Institute (SSI) at Princeton and MIT created a prototype mass driver for ejection of lunar material[9,10]. This early work is directly applicable to a future asteroid mass driver.

VI. Candidate Precursor Mission

A. Overview

Given the uncertainty in the type of future PHA threat and technology challenges for the baseline MADMEN concept, a mitigation precursor mission is deemed useful to building overall mission confidence. The goal of any precursor mission would not be the return of scientific data per se, but the accumulation of data needed precisely for mitigation and testing of relevant mitigation technologies (science return would be substantial but not drive mission requirements). The objective of such a precursor mission would be to observe a change in the trajectory of an asteroid through the use of mass drivers. The technologies that would be involved in such a mission would be subset of those for the full MADMEN concept described earlier. Such technologies on a precursor mission include: attachment, drilling, mass driver ejection, and swarm multi-spacecraft intelligence. Some characteristics of the target asteroid include it being small, a non-binary, non-Earth crossing body, and available for rendezvous within a timeframe of 2011-2015. After a preliminary search amongst various candidates, asteroid 2002 XY38 (Aten, diameter = 70-160 m) is chosen as a potential target with a potentially favorable trajectory solution for this timeframe. Such a mission is envisioned to consist of two asteroid landing spacecraft. Each spacecraft would test different attachment mechanisms and drills. Communication could be provided by an orbiter. A small launch vehicle such as the SpaceX Falcon 1 or 9 could be utilized for such a mission. Overall mission life cycle cost (LCC) would be capped at between one to two times the cost of a nominal NASA Discovery class missions ($450M to $900M).

VII. Conclusions

This paper has elaborated on previous work by the authors on a concept of using multiple, smaller landers, each equipped with a mass driver and drill/coring system, to deflect an hazardous asteroid. By ejecting mass away from the asteroid at high speed, a small change is induced in the asteroid's velocity resulting in a change in its Earth-crossing trajectory. The use of mass drivers is one deflection option that is frequently mentioned as a possibility for future asteroid impact mitigation, but is not currently the subject of widespread conceptual investigation.

As envisioned, the MADMEN mass driver concept is found to have performance (deflection capability per unit of launch mass) comparable to competing non-nuclear options such as the space tug, the gravity tractor, and the kinetic impactor. It does not depend on proliferation of nuclear weapons into space, nor long-term storage and maintenance of nuclear stockpiles. In addition, the concept offers a number of advantages such as precision control of the applied ΔV to the target asteroid.

Two specific case studies were examined. For the notional asteroid D'Artagnon (roughly 130 m in diameter), a solution was proposed to use five MADMEN landers launched on a single Ares V launch vehicle in 2017. The preferred solution was found to have a lander + cruise stage mass of 5,830 kg per lander. The selected design variables were a shot velocity of 570 m/s, a shot mass of 0.50 kg, and a shot frequency of 3 shots per minute. The duty cycle of actual mass driver operations was assumed to by 15% of the surface time due to the rotation of the asteroid. The surface action time was constrained to be one year. This solution yielded 99.38%-99.95% post-launch mission reliability at an estimated LCC of $2.3B.

The second case study considered a small deflection applied to Apophis prior to its encounter with Earth in 2029. A small trajectory deflection of 60 km was assumed to be sufficient to miss the resonant-return keyhole and therefore avoid Earth impact in 2036. The proposed MADMEN solution to this problem would use two landers launched separately on Falcon 9 launchers in 2022. Each lander + cruise stage was estimated to have a mass of 940 kg. For this solution, the mass driver operated with a shot velocity of 150 m/s, a shot mass of 0.15 kg, a shot frequency of 2 shots per minute, and a duty cycle of 15%. The surface action time for this solution was only 140 days with both landers operating nominally. This solution resulted in 96%-99% post-launch mission reliability at an estimated LCC of $815M.

In addition to the strengths of the concept, weaknesses and technical challenges were noted. The need for a firm surface attachment and mechanical operation in a dusty environment were discussed as concerns. In addition, the need to understand the resultant orbit of the ejecta masses was highlighted. The major technology maturation needs were discussed qualitatively, with needs identified in power, swarm intelligence, software, mass driver technology, and drilling/coring.

This work represents a complete mission analysis of a conceptual mass driver asteroid deflection solution, with preliminary consideration of launch vehicle, transfer trajectory, surface operations, spacecraft design, mission costs, and expected mission reliability. The authors firmly believe that it is premature to downselect to a preferred mitigation technique without proper consideration of the advantages and disadvantages of a wide range of options for various types of asteroid or comet planetary threats. It is hoped that this paper contributes to open and engaging discussion within the asteroid deflection community.

Acknowledgments

The authors gratefully acknowledge the contributions of colleagues at SpaceWorks Engineering, Inc. (SEI) who contributed to this paper. Mr. Doug Nelson developed the C_3 vs. payload curves for the Ares V and Falcon 9 launch vehicles using trajectory and mass properties estimations. Mr. Jon G. Wallace developed the CAD models used to help define the concept. Mr. Gary Miller helped perform several of the trade studies used to explore the design space of the key independent variables used to size the MADMEN lander.

Mr. Ben Stahl, an M.S. degree candidate in the School of Aerospace Engineering at the Georgia Institute of Technology (Atlanta, Georgia), conducted the analyses associated with the deflection ΔV requirements for both case studies. He is also responsible for modifying the original JAT tools to propagate a perturbed n-body trajectory so that the effects of various mass driver parameters on the resultant heliocentric orbit could be explored. Mr. Stahl's graduate research is sponsored by SpaceWorks Engineering, Inc (SEI).

References

[1]NASA, "Near-Earth Object Survey and Deflection Analysis of Alternatives: Report to Congress," National Aeronautics and Space Administration, March 2007.

[2]Gehrels, T., Hazards Due to Comets and Asteroids (T. Gehrels, ed.), University of Arizona Press, Tucson, Arizona, 1994.

[3]Modern Mechanics and Inventions, Fawcett Publications, Inc.; Chicago, IL.

[4]"The Electric Gun: German Experiment with Electrically Launched Projectiles," Intelligence Bulletin, May 1946. URL: http://www.lonesentry.com/articles/electricgun/index.html.

[5]Weise, T. H., "Overview on the German R&D Programs on ETC Gun Technologies for Main Battletank Weaponization," Accession Number: ADP012452. April 26, 2001

[6]ESA; "Electromagnetic railgun technology for the deployment of small sub-orbital payloads," ESA Contract No. 13420, September 2003. Accessed on March 27, 2007, URL:
http://esamultimedia.esa.int/docs/gsp/completed/comp_i_03_N02.pdf

[7]Salustri, F., "Design of a Mass Driver System Concept Using Open Source Principles," IAC-04-U.1.07, 55th International Astronautical Congress of the International Astronautical Federation, the International Academy of Astronautics, and the International Institute of Space Law, Vancouver, Canada, Oct. 4-8, 2004.

[8]"The Long, Twilight Struggle." Babylon 5. Director: John C. Flinn III, Writer: J. Michael Straczynski. Season 2, Episode 20. United States Syndication. August 1, 1995.

[9]O' Neill, G.K, "NASA 1977 Ames Summer Study on Space Settlements and Industrialization Using Nonterrestial Materials," Ames Research Center, Moffett Field, California. Accessed on March 27, 2007, URL: http://www.nas.nasa.gov/Services/Education/SpaceSettlement/spaceres/III-3.html

[10]Snively, Leslie O., O'Neill, Gerard K., "Mass Driver III: Construction, Testing and Comparison to Computer Simulation (AAS 83-240)" in Space Manufacturing 1983. Advances in the Astronautical Sciences, James D. Burke, James D., Whitt, April 2, ed., Vol. 53, Proceedings of a Conference held at Princeton University, New Jersey, Univelt, Incorporated, May 9-12, 1983, pp. 391

[11]Olds, J. R., Charania, A., Graham, M., Wallace, J. G., "The League of Extraordinary Machines: A Rapid and Scalable Approach to Planetary Defense Against Asteroid Impactors," Version 1.0, Phase I Final Report, Call for Proposals CP-02-02, NASA Institute for Advanced Concepts (NIAC), April 30, 2004.

[12]Charania, A., Graham, M., Olds, J. R., "Rapid and Scalable Architecture Design for Planetary Defense," AIAA-2004-1453, 1st Planetary Defense Conference: Protecting Earth from Asteroids, Orange County, California, February 24-27, 2004.

[13]Anttila, M. and Ylikorpi, T., "Defining the Technical Requirements for Subsurface Mars Driller," Paper 3020, Sixth International Conference on Mars, Pasadena, California, July 20–25, 2003.

[14]Crocker, A., Charania, A., Olds, J., "An Introduction to the ROSETTA Modeling Process for Advanced Space Transportation Technology Investment," AIAA-2001-4625, Space 2001 Conference and Exposition, Albuquerque, NM, August 28-30, 2001.

[15]Mankins, John C., "Technology Readiness Levels: A White Paper", April 6, 1995, URL: http://www.hq.nasa.gov/office/codeq/trl/trl.pdf, cited 06 March 2007.

[16]NASA, " FY 2008 Budget Request", URL: http://www.nasa.gov/pdf/168652main_NASA_FY08_Budget_Request.pdf, cited 06 March 2007.

[17]Lynch, D. and Peterson, G., "Athos, Porthos, Aramis & D'Artagnan—Four Planning Scenarios for Planetary Protection," 2004 Planetary Defense Conference: Protecting Earth from Asteroids (Orange County, CA, Feb. 23–26, 2004), AIAA Paper 2004-1417.

[18]Asteroid 99942 Apophis (2004 MN4), Last accessed March 19, 2007, URL = http://neo.jpl.nasa.gov/cgi-bin/db?name=99942, cited 06 March 2007.

[19]Chesley, Steven R., "Potential Impact Detection for Near-Earth Asteroids: The Case of 99942 Apophis (2004 MN)," from Asteroids, Comets, Meteors, S. Ferraz-Mello & D. Lazzaro, eds., Proceedings IAU Symposium No. 229, 2005.

[20]Space Exploration Technologies (SpaceX), Last accessed March 6, 2007, URL = http://www.spacex.com.

[21]Rafeek, S., el al, "Sample Acquisition Systems for Sampling the Surface Down To 10 Meters Below the Surface for Mars Exploration," Paper 6239, Concepts And Approaches For Mars Exploration, Lunar and Planetary Institute, Houston, Texas, July 18–20, 2000.

[22]Basso, B, et al, "Surface Anchoring and Deflection Concepts for Earth-crossing Comets and Asteroids," Paper 3032, Workshop on Spacecraft Reconnaissance of Asteroid and Comet Interiors, Santa Cruz, California,

[23]Poston, D.I., 2002, "Nuclear design of HOMER-15 Mars surface fission reactor", Nuclear News, Dec 2001.

[24]Carsey, F.D., el al, "Palmer Quest: A Feasible Nuclear Fission "Vision Mission" To The Mars Polar Caps," Paper 1844, 36th Lunar and Planetary Science Conference, Houston, Texas, March 14–18, 2005.

[25]Poston, D., "SAFE-400 Space Fission Reactor". Nuclear News, December 2002, p. 28.

[26]Curtis, S. A., et al , “Use of Swarm Intelligence in Spacecraft Constellations for the Resource Exploration of the Asteroid Belt,” Third International Workshop on Satellite Constellations and Formation Flying, Pisa, Italy, 24-26, 2003.
[27]Calame, J., et al, “The Design of a High Power Gyroklystron for Supercollider Applications” IEEE 1987 Particle Accelerator Conference, Paper 1904, March, 1987.
[28]NASA's Exploration Systems Architecture Study -- Final Report, URL = http://www.nasa.gov/mission_pages/exploration/news/ESAS_report.html, cited 06 March 2007.