THE LEAGUE OF EXTRAORDINARY MACHINES:
A RAPID AND SCALABLE APPROACH TO
PLANETARY DEFENSE
AGAINST ASTEROID IMPACTORS

Version 1.0

SpaceWorks Engineering, Inc.

NASA INSTITUTE FOR ADVANCED CONCEPTS (NIAC)
PHASE I
FINAL REPORT

THE LEAGUE OF EXTRAORDINARY MACHINES: A RAPID AND SCALABLE APPROACH TO PLANETARY DEFENSE AGAINST ASTEROID IMPACTORS

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Table of Contents

APPENDICES

List of Acronyms

ACS	Attitude Control System
ANTS	Autonomous Nano-Technology Swarm
AU	Astronomical Unit
CAPS	Comet Asteroid Protection System
ConOps	Concept of Operations
DDT&E	Design, Development, Testing, and Evaluation
DEFT	Defined Threat
DES	Discrete Event Simulation
ETO	Earth To Orbit
GEO	Geostationary Orbit
HEDS	Human Exploration and Development of Space
ISTS	In Space Transfer Stage
K-T	Cretaceous-Tertiary
L1, L2	Lagrange Points
LCC	Life Cycle Cost
LD	Lunar Distance
LEO	Low Earth Orbit
LM	Lunar Module
LOEM	League of Extraordinary Machines
LPC	Long Period Comet
MADMEN	Modular Asteroid Deflection Mission Ejector Node
MBS	Mass Breakdown Structure
NEO	Near-Earth Object
NIAC	NASA Institute for Advanced Concepts
PBR	Particle Bed Reactor
PHO	Potential Hazardous Object
RASC	Revolutionary Aerospace Systems Concepts
ROSETTA	Reduced Order Simulation for Evaluation of Technologies and Transportation Architectures
SEI	SpaceWorks Engineering, Inc.
TFU	Theoretical First Unit
USRA	Universities Space Research Association

Foreword and Acknowledgements

Acknowledgments are given to the NASA Institute for Advanced Concepts (NIAC), part of the Universities Space Research Association (USRA), for funding of the Phase I study that encompasses research addressed in this paper. Particular appreciation is extended to NIAC director Bob Casanova for providing material and collateral support on this project. Gratitude is also expressed to the late Gerard K. O'Neill for his discussion on the subject of mass drivers. Additional appreciation is also expressed to Dr. Steven A. Curtis of NASA's Goddard Space Flight Center and his associated NASA colleagues for their investigative work on concepts for space exploration using a swarm of spacecraft.

Executive Summary

A new approach to mitigate and protect against planetary impactor events is proposed. The primary objective of this systems concept is to apply small perturbations to Near-Earth Objects (NEOs) in an attempt to divert them from their path toward Earth impact. Unlike past proposals from other individuals or organizations, this project concept proposes a rapid and scalable solution consisting of tens, hundreds, or thousands of small, nearly identical spacecraft that will intercept the target body and conduct mass driver/ejector operations to perturb the target body's trajectory to the point where an impact with Earth can be avoided. There have been many ideas woven together to develop the asteroid concept presented herein. From mass driver technology to notions of swarm intelligence, various technologies and approaches have been combined to develop a potentially unique approach to planetary defense against asteroids and comets.

Such a spacecraft, referred to here as a Modular Asteroid Deflection Mission Ejector Node (MADMEN) spacecraft could conceivably be nuclear powered, pre-deployed outside of low earth orbit (likely at an Earth-Moon or Earth-Sun libration point), and be capable of using chemical propulsive boost to rapidly intercept an incoming target. Upon arrival at the target, each MADMEN spacecraft will begin to eject small amounts of mass from the asteroid that will, over time, have the effect of slightly changing the heliocentric orbit of the target so that Earth impact is avoided. Such a design philosophy focuses on developing rapid and scalable NEO mitigation plans incorporating the world's current launch vehicle/spacecraft bus manufacturing capability. Specific planetary threats are examined, each with different impact times and masses, and based upon predetermined fictitious Defined Threat (DEFT) scenarios. Potential advantages envisioned in such an architecture design include: integrating the analysis of spacecraft development/deployment/launch, ability to complete the mission given the loss of part of the swarm, scalability of response for different size threats, and flexibility to initiate an immediate response leaving the option to develop more advanced systems.

The Phase I portion of the project presented the MADMEN architecture as a novel and potentially valuable technique for NEO deflection. The assessment in Phase I has essentially indicated that a swarm of several, identical spacecraft, launched on existing Earth-To-Orbit (ETO) launch vehicles may be able to drill out enough material over the course of several months to change a locally/regionally-devastating Near Earth Object (NEO) trajectory by enough of a miss distance (several Earth diameters) using a nuclear powered mass driver rail gun, losing about half of the fleet in the process. It was shown that for a 100 m diameter object less than two hundred MADMEN lander spacecraft were required for mission success with a surface operations time of 60 days. This indicated that the concept could be practical given the very conservative, self-imposed surface operation limit of 60 days. The general feasibility of the MADMEN architectural concept has been established. Conceptual level modeling proved the basic tenants of the concept given assumptions about the on-board subsystems. A conceptual level model of a nuclear powered MADMEN swarm was developed that included trajectory analysis, lander mass sizing, lander power budget, impactor definition, in-space stage sizing, preliminary end-to-end system reliability, and life cycle cost analysis. Preliminary trajectory analysis indicated current launch vehicles can be utilized for this concept. A nominal cost per spacecraft similar to NASA Discovery-class missions has been demonstrated. Substantial reductions can be made in the total number of spacecraft and/or spacecraft mass if both surface operation time and deflection distance are traded-off in the analysis. Specific use was made of industry-defined fictional threat scenarios to present a case study of this planetary defense architecture.

1.0 Introduction

Both recent observations of planetary bodies and geological records confirm the ever present threats from asteroids and comets that could be large enough to cause the widespread destruction of modern society. For instance, a massive impact occurred in the Tunguska region of Siberia in 1908, likely from a 40-60 meter wide asteroid, that devastated 2,000 square kilometers with a destructive force equivalent to 10-50 megatons of TNT (several hundred to several thousand times the energy unleashed by an atomic bomb over Hiroshima in 1945). Additionally, the Earth's surface still shows scars of previous larger-scale impacts. It is believed that the Meteor Crater in Arizona was caused by the impact of a 100-meter diameter meteorite over 20,000 years ago. The more massive K-T (Cretaceous-Tertiary) impact (10 km diameter object), which took place approximately 65 million years ago, is believed to have led to the extinction of the dinosaurs. While K-T class impacts are very infrequent, objects with diameters of approximately 1 km can be expected to intercept the Earth every 100,000 years[1]. Figure 1.1 depicts the relationship between the destructive potential of an impactor and its probability of occurrence.

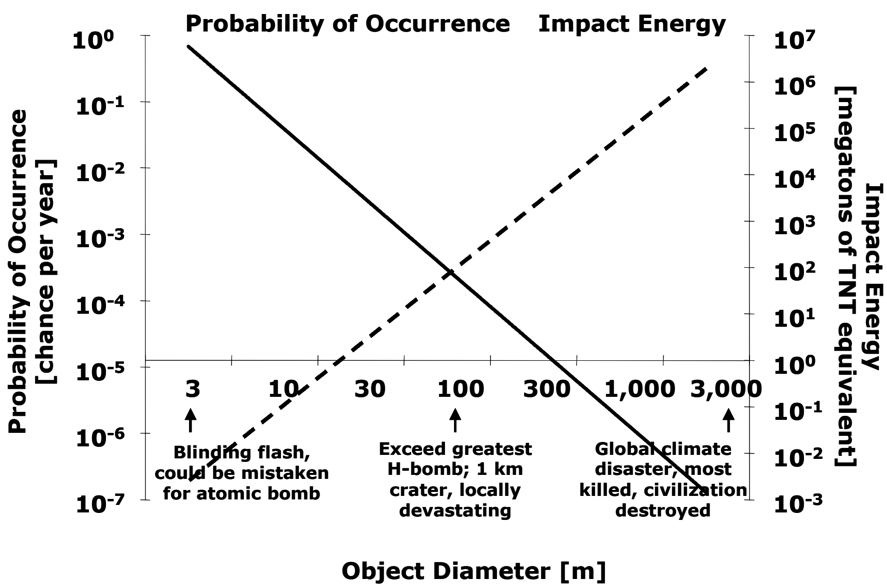


Figure 1.1. Impact Potentials[2]

Clearly, some thought and planning must take place in order to provide a reasonable level of protection against such disastrous events. Identification and cataloging of Near-Earth Objects (NEOs) and celestial bodies such as that of the Spaceguard survey is an important first step. Figure 1.2 contains imagery of several of these objects. As seen in Figure 1.3, cataloging efforts over the past several decades have gradually exposed the sheer number of NEOs. Observation and tracking of small (1 km or less) objects is a difficult task given the low albedos of the target bodies and their small size. However, accurate long-term orbital prediction models must be developed to allow for adequate response time. NASA and Cal Tech's Jet Propulsion Lab maintain an active list of NEO objects sorted by a weighted scale indicating their approach distance and destructive potential (the Palermo scale). In 2003, over 10 objects were identified that passed within 1.5 times the distance from the Earth to the Moon (1 lunar distance = 384,000 km). The smallest objects in the JPL database are 20 meters in diameter with typical NEOs in the 500 m - 1 km diameter range[2]. Currently, the highest active object on the Palermo scale is asteroid 1997 XR2, a 230 m diameter asteroid having an impact probability of 9.7x10-5 in the year 2101. NASA has proposed new observatories that will be able to detect even smaller objects. For example, NASA's Revolutionary Aerospace Systems Concepts (RASC) program conducted a study called the Comet Asteroid Protection System (CAPS) that promoted a lunar telescope installation for conducting NEO detection research[3,4]. The question remains: "What should be done if a planetary impactor on a collision course with Earth is actually confirmed?"

Effective planetary defense architectures must overcome a variety of challenges. The large variance in size, shape, composition, and detection time of NEO impactors requires a robust system. Economics and launch vehicle capacity limit the size and scope of the vehicles. Designs must be able to easily incorporate technological advances to maximize efficiency. A modular architecture of smaller devices can help overcome many of these challenges by providing the means to build up defensive capability immediately while allowing for system improvements and modifications over time.

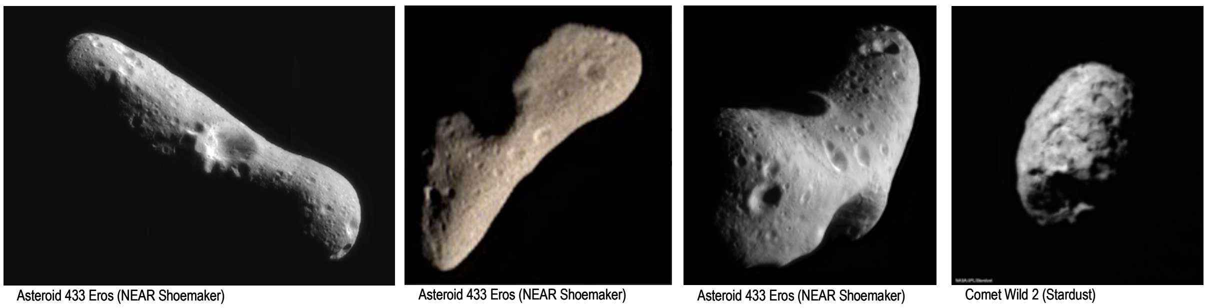


Figure 1.2. Sample Near Earth Objects (NEOs)[5,6]

The preliminary results described herein have been performed under a Phase I contract sponsored by the NASA Institute for Advanced Concepts (NIAC). The study is entitled "The League of Extraordinary Machines: A Rapid and Scalable Approach to Planetary Defense Against Asteroid Impactors" and was awarded in October of 2003 under NIAC Call for Proposals CP-02-02. This report summarizes the activities and accomplishments of the Phase I study. Presentations about this concept were made at the NIAC 5th annual meeting, November 5-6, 2003 (Atlanta, Georgia) and at the NIAC Fellows Meeting, March 23-24, 2004(Arlington, Virginia). The project poster displayed at the November 5-6 meeting is shown in Figure 1.4.

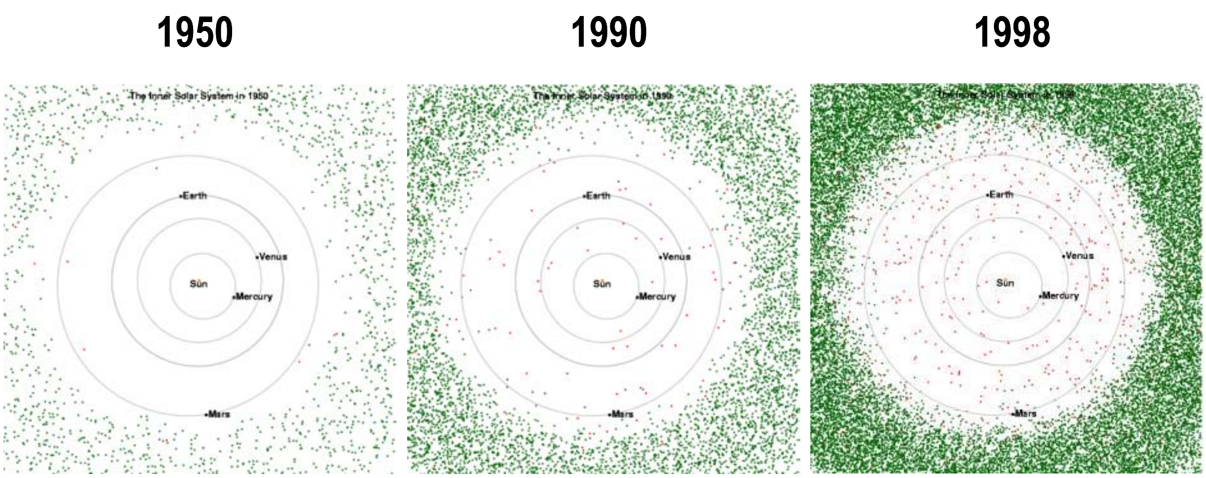


Figure 1.3. Chronological Near Earth Object Maps of the Inner Solar System[7]

NIAC's charter is focused on grand, revolutionary concepts for architectures and systems. As such the institute provides an independent, open forum for the external analysis and definition of space and aeronautics advanced concepts that complement the advanced concepts activities conducted within the NASA Enterprise. NIAC is a part of the Universities Space Research Association (USRA).

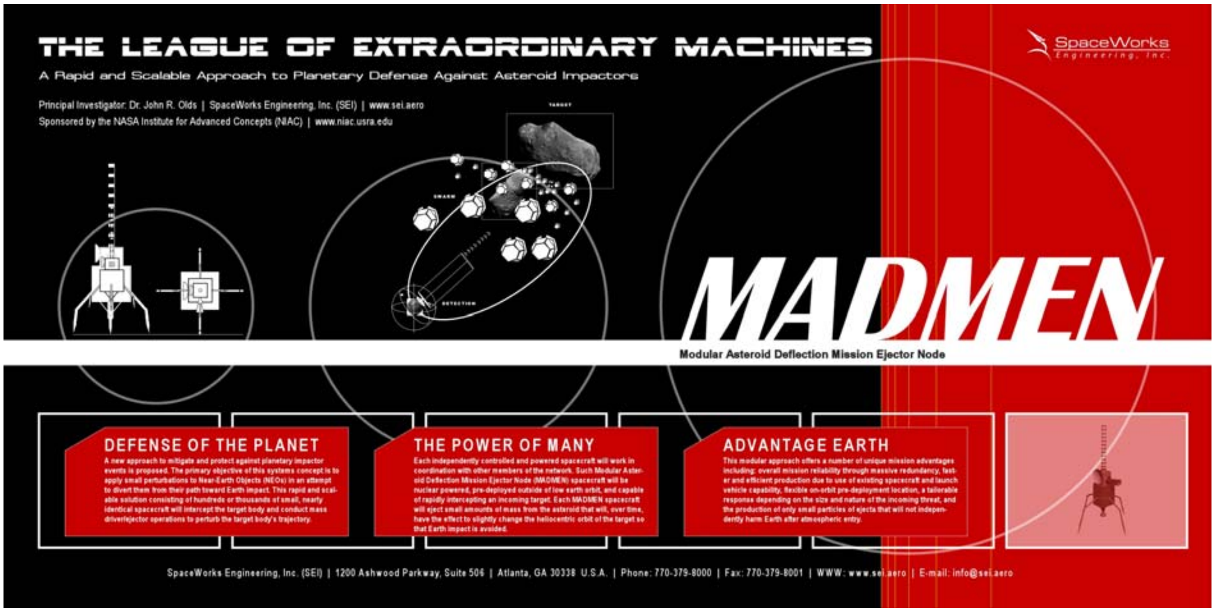


Figure 1.4. Project Poster Presented at NIAC Fifth Annual Meeting, Atlanta, GA, November 5-6, 2003

2.0 Background

In 1998, Hollywood released two movies on the subject of planetary impactors, Deep Impact (Paramount Pictures) and Armageddon (Buena Vista). In both of these films, writers created situations in which large impactors are discovered on collision trajectories with Earth with short warning times. In both cases, nuclear devices are used to try to fracture the incoming asteroid or comet. In one case, the nuclear device is launched on a military missile. In the other case (Armageddon), the rather implausible approach of sending a crewed shuttle out of Earth orbit to intercept the comet and manually implant a nuclear device is adopted. These fracture/fissure approaches are probably best left as Hollywood solutions. Given the uncertainties in the makeup and mass distribution of a NEO impactor, it is unlikely that a single impulsive blast from a nuclear device would deliver adequate ΔV in a desired direction to divert the asteroid. The nuclear blast would, however, have a large effect on the rotation, composition, and condition of the impactor and could potentially reduce the chance further mitigation attempts would be successful.

Given a reasonable lead-time, a sustained impulse or force applied to the impactor for an extended duration may be the best way to achieve the desired collision avoidance. Applied far from Earth, changes in the asteroid's velocity of less than 1 m/s are generally sufficient to change the minimum approach distance by several hundred thousand kilometers. Even with such a small ΔV requirement, propellant-oriented solutions are still impractical. To change the velocity of Asteroid 1997 XR2 by 1 m/s would, for example, require 175 MT of propellant even assuming a very high performance electric propulsion system. Either propellantless or in-situ propellant (i.e. reaction mass from the target body itself) solutions are more practical.

Some researchers have proposed solar or magnetic sails as mitigation concepts, but the resultant thrust available from these devices is too low to have a timely impact on the asteroid's course[3]. In the aforementioned CAPS study, the NASA RASC team briefly considered a GW-class laser-based approach that would ablate volatile surface material from the asteroid or comet[4]. The escaping gases would impart a small change in momentum to the target body. While this approach has the advantage of creating relatively benign ejecta from in-situ sources, the launch, deployment, and station-keeping requirements of such a large laser would seem impractical. The thrust of this system would also be small, requiring extended operations. In addition, limitations are placed on the type of impactor that can be affected in this way, given that the surface must be ablated. A summary of such mitigation techniques are given in Table 2.1.

Renowned physicist Gerard K. O'Neill, founder of the Space Studies Institute at Princeton University, was an advocate of mass-driver solutions for extraterrestrial purposes. These mass-driver devices use electromagnetic coils to accelerate resource material or in-situ debris to speeds of 100 m/s or more. A reactive force is applied to the target body in the process. For example, O'Neill advocated the use of solar-powered mass drivers on the moon to launch lunar resources into low lunar orbit. In addition, he advocated the use of mass-drivers to maneuver small asteroids into high Earth orbit-again for the purpose of mining their metallurgical resources.

Table 2.1. Alternate Mitigation Techniques

Item	Main Effects
Solar Sails	The orbit of a Near Earth Object (NEO) could be altered by attaching sails designed to catch the Solar Wind streaming from the Sun. For large asteroids, however, the size of sail required may be too large to be realistic.
Mass Driver	A device that ejects materials from the surface of an object that would slowly change its orbit.
Solar Mirrors	The orbit of a Near Earth Object (NEO) could be changed by focusing sunlight (or artificial laser light) onto the surface of the object. The jet of gas produced would change the path of the object particularly if it contains abundant water or carbon such as a C-type asteroid.
Engines	Engines, either attached to the NEO or on a spacecraft, could be used to move the object. On some NEOs water locked up in their minerals could be used as fuel.
Impactors/Explosives	These (chemical or nuclear explosives) could be used to generate a crater on an NEO. The ejection of materials from the asteroid will change its motion. For comets a crater could form a new active area producing a jet of gas which will change the orbit still further.

Before his death in 1992, Prof. O'Neill suggested that a mass-driver might also be used in planetary defense applications. Asteroid material would be used as ejecta ("propellant"). In a paper entitled "A Robust, Non-nuclear Defense Against Asteroids" presented at an early Asteroid Mitigation Workshop, O'Neill encouraged further work on the concept[8]:

> A concept that has received relatively little attention in the literature is the employment of a mass driver to apply a steady acceleration to the asteroid which, given sufficient time, will develop lateral movements that can convert a strike on Earth into a miss. The major advantage of this approach is that all the energy comes from the sun and all the reaction mass is obtained from the mass of the asteroid itself. The only mass that needs to be transported to the asteroid is that of the solar collectors, power supplies and acceleration coils which convert electrical energy to kinetic energy. It is recommended that this little studied approach receive research funding. One key issue that deserves examination is the design of mass drivers which are optimized towards low velocity ejecta since these designs convert the highest percentage of the collected solar energy to momentum transfer to the asteroid.

His calculations indicate that given an operating time of 10 years, a 140 kW mass driver could move the closest approach distance of a large asteroid (1×10^{12} kg) by a few Earth diameters[8]. O'Neill considered a single, monolithic mass driver solution that would nominally be solar-powered.

Advancement in robotic technology has progressed substantially over the decades. Recent research has focused on the harnessing the potential capabilities of biologically-inspired design as seen in nature[9,10,11]. This has also extended to the concept of using multiple robots in combination to perform tasks. Dr. Steven A. Curtis of NASA's Goddard Space Flight Center and his other NASA colleagues (including those at JPL) have proposed various exploration concepts that utilize swarm intelligence. They draw inspiration from biological analogues as stated[10]:

> ANTS [The Autonomous Nano-Technology Swarm] is a mission architecture drawn in analogy to biological social insects…In nature, biological ants are one of the most successful species to appear. Elements of their success lie in their division of labor, their collaboration, and their numbers. Their numbers provide reliability through redundancy for many functions of the swarm. However, individual ants are themselves highly autonomous and capable creatures, however they do depend on other members of their society to perform tasks necessary for their survival and procreation. Because individual ants are

specialized to a given task, they are much more likely to succeed in that task than a non-social generalist. The ants and their swarm benefit from these successes, and the colony continues on with its competitive advantage. In some sense, we aspire to produce systems with the robustness and adaptability of ants. Biological ants are capable of relatively sophisticated behaviors with relatively simple neural structures and communication mechanisms, we seek to emulate both the individual and swarm capabilities in the ANTS mission architecture.

Recent studies of asteroid missions from a wide variety of sources have included some aspect of the swarm philosophy. Specific examples from the 1st Planetary Defense Conference: Protecting Earth from Asteroids, (Orange County, California, February 24-27, 2004) include:

- The U.K. QinetiQ's Smallsat Intercept Missions to Objects Near Earth (SIMONE) mission utilizing a fleet of low-cost microsatellites that will individually rendezvous with a different Near Earth Object (NEO)[12].
- A recent Aerospace Corporation study which indicated a need to incorporate redundancy into mission design. This included both spacecraft and launch pads (launch failures taking out a pad). They included some preliminary estimates for multiple small spacecraft and launch vehicles[13].
- As presented by G. Somer from RAND, the Project CARDINAL reference design included a swarming approach to the mitigation architecture[14].

Many ideas were woven together to develop the planetary defense concept architecture presented herein. From mass driver technology to notions of swarm intelligence, various technologies and approaches have been combined to develop a potentially unique approach to confront the threat to Earth posed by asteroids and comets.

3.0 Objective

A new approach to mitigate and protect against planetary impactor events is proposed. The primary objective of this systems concept is to apply small perturbations to Near-Earth Objects (NEOs) in an attempt to divert them from their path toward Earth impact. Unlike past proposals from other individuals or organizations, this project concept proposes a rapid and scalable solution consisting of hundreds or thousands of small, nearly identical spacecraft that will intercept the target body and conduct mass driver/ejector operations to perturb the target body's trajectory to the point where an impact with Earth can be avoided.

In the nominal configuration, each spacecraft will be independently controlled and powered, but will work in loose coordination with other members of the network. Such a spacecraft, referred to here as a Modular Asteroid Deflection Mission Ejector Node (MADMEN) spacecraft could conceivably be nuclear powered, pre-deployed outside of low earth orbit (likely at an Earth-Moon or Earth-Sun libration point), and be capable of using chemical propulsive boost to rapidly intercept an incoming target. Upon arrival at the target, each MADMEN spacecraft will begin to eject small amounts of mass from the asteroid that will, over time, have the effect to slightly change the heliocentric orbit of the target so that Earth impact is avoided.

This modular approach can offer a number of unique mission advantages including: overall mission reliability through massive redundancy, faster production capability due to use of existing spacecraft bus production capability, efficiencies-of-scale of the MADMEN spacecraft during production, flexible and practical launch and transfer to an on-orbit pre-deployment location, a tailorable response depending on the size and nature of the incoming threat, and the production of only small particles of ejecta that will not independently survive Earth atmospheric entry.

This six month Phase I study effort established key quantitative data for the masses, dimensions, costs, schedules, and technology requirements to assess concept viability. Comparisons have been made with alternate approaches to verify the stated advantages. This Phase I investigation specifically examined the following:

- Developed trend line curves for power requirements vs. mass throughput and ejection velocity.
- Developed a preliminary parametric design for the mass-driver component of the MADMEN spacecraft and conceptual configuration based on these results.
- Developed a preliminary parametric design and configuration for the MADMEN lander spacecraft.
- Performed basic in-space trajectory simulations to rendezvous with representative NEOs to determine ΔV and mission time requirements.
- Architecture optimization to determine the launch dimensions and mass constraints appropriate in order to provide access to a large number of launch vehicles.
- Architecture cost analysis to achieve a preliminary estimate of the architecture cost (non-recurring: DDT&E and production, and deployment) as a function of technologies and scale of deployment (i.e. number of spacecraft sent per target).
- Used the cost and architecture design results to determine an appropriate size for the individual spacecraft (i.e. fewer and larger vs. more and smaller, sized by power as an independent variable).

The results of the work plan listed above include an improved understanding of the design space, a more detailed baseline or reference concept design, preliminary weight statements, cost estimates, and schedules. Concept illustrations and renderings of the proposed concept are also presented.

4.0 Analysis Overview

This report summarizes results from Phase I of the investigation. A general description of the concept is presented along with a subsequent list of preliminary parametric studies. These were performed to provide some guidance as to the development of the baseline system concept. Development of the in-space transportation portion to reach a sample NEO is included. Descriptions of proposed future work are also provided.

This project utilized a conceptual modeling process to examine both spacecraft development and launch vehicle manifesting for various NEO threats. This process generated candidate designs based upon current spacecraft buses (commercial, civil, military) incorporating mass driver technology used to eject material from the NEO target. The modeling determined the optimum use of existing and future launch vehicles given constraints on availability. The process also examined the range of capability of such architectures; for various out times (time before impact) and program start years, in order to maximize NEO size that can be affected by the design. Specific planetary threats (one comet, three asteroid-based) are examined each with different impact times and masses based upon predetermined fictitious Defined Threat (DEFT) scenarios. Limited trade studies conducted include alternate spacecraft propulsion (solar, nuclear), mass ejection, and drilling schemes. The architecture's mission effectiveness will be examined probabilistically to determine the number of spacecraft in the swarm needed for various certainty levels that the NEO's path will be deflected. This will help develop a realistic assessment of the state of the possible for NEO threat mitigation given technology and production limitations.

5.0 Concept Overview

A modular/swarm architecture, based upon existing spacecraft buses and launch vehicles, is proposed to mitigate near-Earth object (NEO) planetary threats. Each spacecraft is part of a large swarm or collection of similar spacecraft that would utilize mass driver technology to remove mass from the object to yield an Earth-avoiding trajectory. The notion of a swarm of spacecraft intercepting a detected threat is illustrated in Figure 5.1. Each MADMEN spacecraft would be a self-contained mass driver. The concept envisions a fleet of hundreds or thousands of small mass-driver landers that would loosely coordinate with each other to “attack” a target body much as a swarm of killer bees might attack a large target (see Figures 5.2 to 5.3). Each spacecraft would be separately powered, maneuvered and controlled.

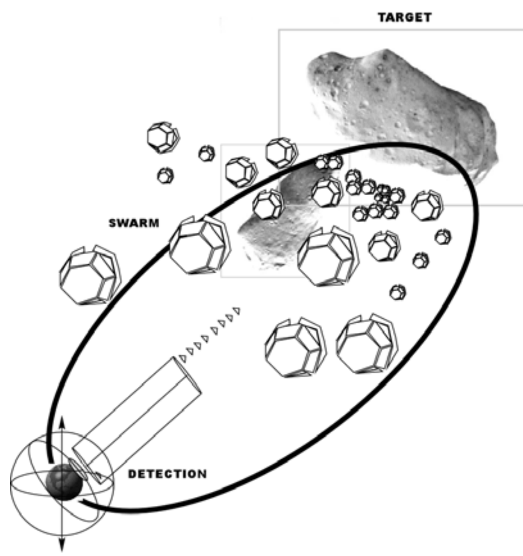


Figure 5.1. Illustration of Swarm Concept

Such a design philosophy focuses on developing rapid and scalable NEO mitigation plans incorporating the world’s current launch vehicle/spacecraft bus manufacturing capability. Potential advantages envisioned in such an architecture design include: integrating the analysis of spacecraft development/deployment/launch, ability to complete the mission given the loss of part of the swarm, scalability of response for different size threats, and flexibility to initiate an immediate response leaving the option to develop more advanced systems.

The revolutionary aspect of this particular architecture is the use of several types of existing technologies (in combination) to develop a robust and scalable solution to the planetary protection problem. Many previous attempts at these types of solutions have relied upon unproven technologies that can require long development times/testing, large deployments, and massive expenditures of resources. If a NEO threat is less than a few years away from impact, many of the previously discussed concepts will be worthless given the immaturity of their technical development. A more rational strategy would be to have a plan wherein the Earth can develop a more rapid, proven, easily implementable and scalable response to such a NEO threat. The existence of such a plan still leaves the option open of simultaneously pursuing more exotic mitigation strategies that will take longer to develop.

Figure 5.2. Notional Illustration of MADMEN Lander Spacecraft Approach and Operation

Figure 5.3. Notional Illustration of MADMEN Lander Spacecraft Operating on Surface

The approach relies on the use of mankind's current experience and infrastructure obtained over the many decades of space exploration. Thus mitigation options can be based upon the entire planet's combined space resources (specifically manufacturing and launch capability). This approach is coupled with one other capability achieved when examining the solution from such a global perspective. One of the key strengths of the global space community is not the sheer mass that can be constructed and launched at any one time, but the capacity to build and launch multiple spacecraft; the power of Earth's response to a NEO threat lies in the quantity of production possible. Such a capability would indicate an approach based upon quantity of spacecraft. This philosophy is one of the hallmarks of the concept described herein.

5.1 MADMEN SPACECRAFT

Each Modular Asteroid Deflection Mission Ejector Node (MADMEN) spacecraft will be equipped with a power source, a drilling/pulverizing mechanism, landing anchors, a mass driver accelerator, and associated subsystems. A notional representation of a MADMEN spacecraft can be seen in Figure 5.4 below.

Figure 5.4. Concept Illustration of Nominal MADMEN Lander Spacecraft

The Phase I study determined masses and dimensions for the final baseline spacecraft. Early estimates (based upon parametric studies described further herein) indicate that each MADMEN spacecraft would be approximately 500-1,000 kg in mass and approximately 5-20 meters in length once fully deployed on the target body. Such estimates also suggested that each MADMEN spacecraft will be able to grind and eject approximately 100 kg of asteroid material per hour and eject that debris at a velocity of 100 m/s. Early estimates assumed a power requirement of less than 300 W per spacecraft. Depending on warning time, mass of the target asteroid, its rotation rate/orientation, and the desired ΔV to be imparted, multiple MADMEN spacecraft will be dispatched to address the threat. Tens, hundreds or thousands of such spacecraft may be required for a given threat. The notional architecture also includes separate in-space stages to get the MADMEN spacecraft out to the impactor. The key elements of the MADMEN architecture are outlined in Table 5.1.

Table 5.1. Key MADMEN Lander Spacecraft Characteristics

Concept Characteristic	Benefit
Mass-Driver	Propellant-less operation uses asteroid's material as ejecta to deliver sustained impulse to the target without the requirement to provide and manufacture additional propellant.
On-board Power	Baseline power source is nuclear power for long life and deep space compatibility. Consider solar power as a trade study.
System Modularity	Allows massive system redundancy and increases overall mission reliability. Individual spacecraft can fail and still have mission success.
MADMEN Design Commonality	Ensures high production rates and economies of scale during production. Opens competition to a vast array of spacecraft bus manufactures.
Small Spacecraft Design	Allows launch and deployment on a variety of domestic and international launch vehicles. Launch of multiple MADMEN on small or large launchers can be accommodated. Lower launch costs and faster response time.
Small Ejecta Mass (per piece)	Creates smaller objects in ejecta debris field that are unlikely to survive entry into Earth's atmosphere

5.2 MADMEN PRODUCTION AND DEPLOYMENT

The proposed end-to-end solution used here examines the entire architecture associated with NEO mitigation, integrating spacecraft response, as well as launch. The relatively manageable size and simplicity (excluding the power source) of the MADMEN spacecraft will enable its developers to make use of worldwide spacecraft production capabilities that will both ensure competition and achieve a high rate of annual spacecraft manufacture. Since a large number of spacecraft will be required, each manufacturer will be able to reach a certain economy of scale during production through process automation, robotics, labor learning effects, commonality, and standardization of parts and supply lines, etc. An initial assumption is to manufacture the MADMEN spacecraft using relatively "low-tech" materials and components. This is not to suggest a dismissal of new technologies; certain spacecraft technologies may be unique for this architecture, namely the mass-driver components, zero gravity drill, and autonomous control systems. However, compared to other solutions, the inclusion of such technology is expressed in only the most critical areas where current capabilities are lacking. The result will be an extremely affordable spacecraft relative to the massive and expensive one-of-a-kind lasers or mass drivers previously discussed.

An additional benefit of the small mass and stowed size of the MADMEN spacecraft is the availability of the world's current fleet of existing launch vehicles to be used as launchers. No custom heavy-lift launcher is required. Multiple manifesting options exist with larger launchers (e.g. Proton, Atlas V, Delta IV, Ariane V), while smaller launchers could be used to deploy fewer MADMEN to the assembly point in a single launch. New reusable launchers could be employed as they are developed, but the ability to launch on current expendable launch vehicles is a key characteristic of our concept. Not only does this enable the concept to be responsive to near-term threats (next 10 or 15 years), it ensures that the launch will be as economical as possible due to competition.

5.3 SYSTEM RELIABILITY AND ROBUSTNESS

With the survival of thousands or millions of humans at stake, the reliability of a proposed asteroid deflection system cannot be compromised. The MADMEN architecture achieves reliability through massive redundancy of the lander spacecraft. Since each MADMEN is essentially independent, the loss of one or more of the spacecraft will not compromise the overall mission success. Mission planners will simply dispatch more spacecraft than is necessary to achieve the required orbital adjustment. The use of multiple, identical spacecraft in such a configuration against a target is notionally referred to here as a "swarm". These swarms are robust enough (through design and embedded intelligence) to complete the objective. Even excluding failures on the outbound journey, the harsh circumstances of the environment near potential NEO threats themselves dictate multiple backups. A swarm concept has the backup design feature inherent in its architecture.

5.4 SCALABILITY

A swarm approach using multiple MADMEN spacecraft allows scalability to any response. Scalability is achieved by tailoring the population level of the swarm to meet the demands of a particular threat. Given the varied nature of future threats (size, shape, composition, etc.), many possible mission scenarios exist. The similarity of each member of the swarm to other members means that scalability can be realized without the effort associated with designing entirely new systems for each threat.

5.5 CONCEPT OF OPERATIONS (CONOPS)

The MADMEN architecture can be viewed as an actionable plan of attack against an Earth impactor object. This plan, based upon the use of existing spacecraft resources, can be updated as time progresses. The landers can be produced even before a potential threat and pre-positioned. A sample concept of operations for the MADMEN architecture could include the following:

- Manufacture an adequate number of MADMEN spacecraft. Likely done before the identification of a specific threat.
- Deploy the MADMEN to an orbital assembly point. Tradable location but likely somewhere above LEO. Perhaps an Earth-Moon or an Earth-Sun libration point. Figure 5.5 below describes the deployment options for a planetary defense system.
- Identify a target planetary impactor on a collision course with Earth.
- Dispatch an adequate number of MADMEN toward the target (a responsive swarm with redundancy), chemical propulsion boost stages are used for in-space transfer. Figure 5.6 depicts a swarm of MADMEN approaching a target asteroid. Figure 5.7 illustrates what a swarm might look like as it closes in on a comet.
- Arrive at the target body and land MADMEN on surface.
- Each MADMEN spacecraft begins to pulverize and eject asteroid material at a given mass per hour (variable power vs. ejection velocity and mass). See Figure 5.8 to visualize several MADMEN at work on the surface of an asteroid.
- MADMEN work as a team to affect the orbit of the asteroid so that its new trajectory does not intercept Earth. The concept of a swarm of MADMEN acting together to deflect an asteroid is shown in Figure 5.9.

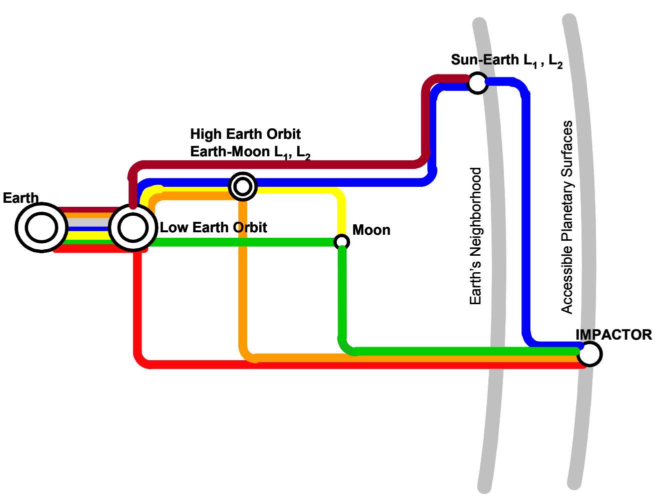


Figure 5.5. Deployment Options for Planetary Defense System[15]

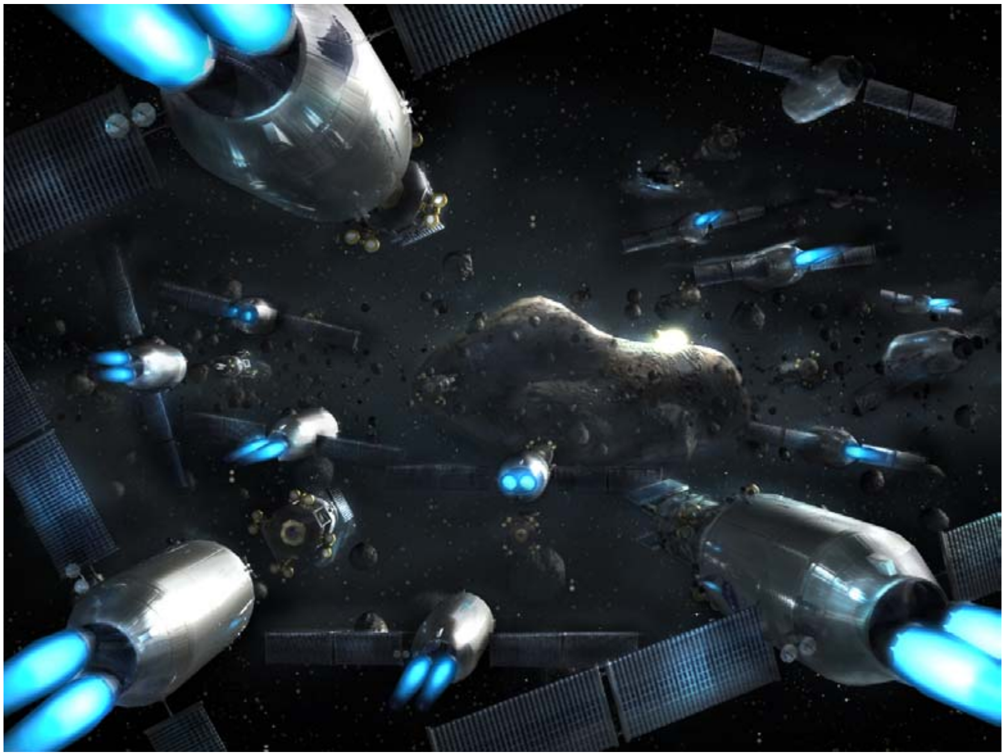


Figure 5.6. MADMEN Spacecraft Intercept an Asteroid

Figure 5.7. MADMEN Spacecraft Approach a Comet

Figure 5.8. Several MADMEN Commencing Operation on the Surface of an Asteroid

Figure 5.9. MADMEN Swarm Ejecting Mass from Asteroid

6.0 Initial Parametric Analyses

The opening stage of research involved several parametric analyses to investigate the potential design space. Specifically these included trades of ΔV requirements, ejection power/energy/lander force, source power and launch power, and the number of MADMEN-type spacecraft required for a particular object. The potential NEO threats examined for these trades were not based on any one specifically defined object. The outputs of these parametric analyses, the reference configuration specifications, were used as guidelines for the design of the baseline MADMEN spacecraft. Some initial specifications generated for the reference configuration were modified given more detailed information about specific NEO threats (such as those in the defined threat scenarios) that were subsequently examined.

6.1 ΔV REQUIREMENT

This concept entails the use of impulse provided by numerous mass driver/landers to adjust the heliocentric orbit of a potential impactor. An assumption is that the most likely NEO threats or impactors will come from the population of Apollos and Atens asteroids that orbit near the vicinity of Earth's orbit. In addition, it is assumed that the potential impactor is detected with enough lead-time to allow the swarm of spacecraft to reach it while it is near its apohelion.

Figure 6.1 highlights a trade study used to establish the baseline ΔV requirements for the mass drivers. For this case the impactor orbit has a perihelion distance of 1 AU and the potential impact therefore occurs at perihelion. For simplicity, it is assumed that the impulse applied by the swarm of landers is delivered instantaneously, rather than over a significant time. A parametric study was performed to determine the ΔV required to change the impactor's perihelion distance by a small amount, measured in units of Lunar Distance (LD) or the distance from the Earth to the moon (1 LD = 384,000 km). For this class of NEO, apohelion distances are likely to be near 1.0-1.5 Astronomical Unit (AU), but apohelion distances up to 3 AU were considered. For small changes in perihelion radius (Δr), and the most likely apohelion radii, the required ΔV is under 10 m/s. While this seems like a small ΔV requirement, recall that the enormous mass of the asteroid makes even a small ΔV difficult to accomplish.

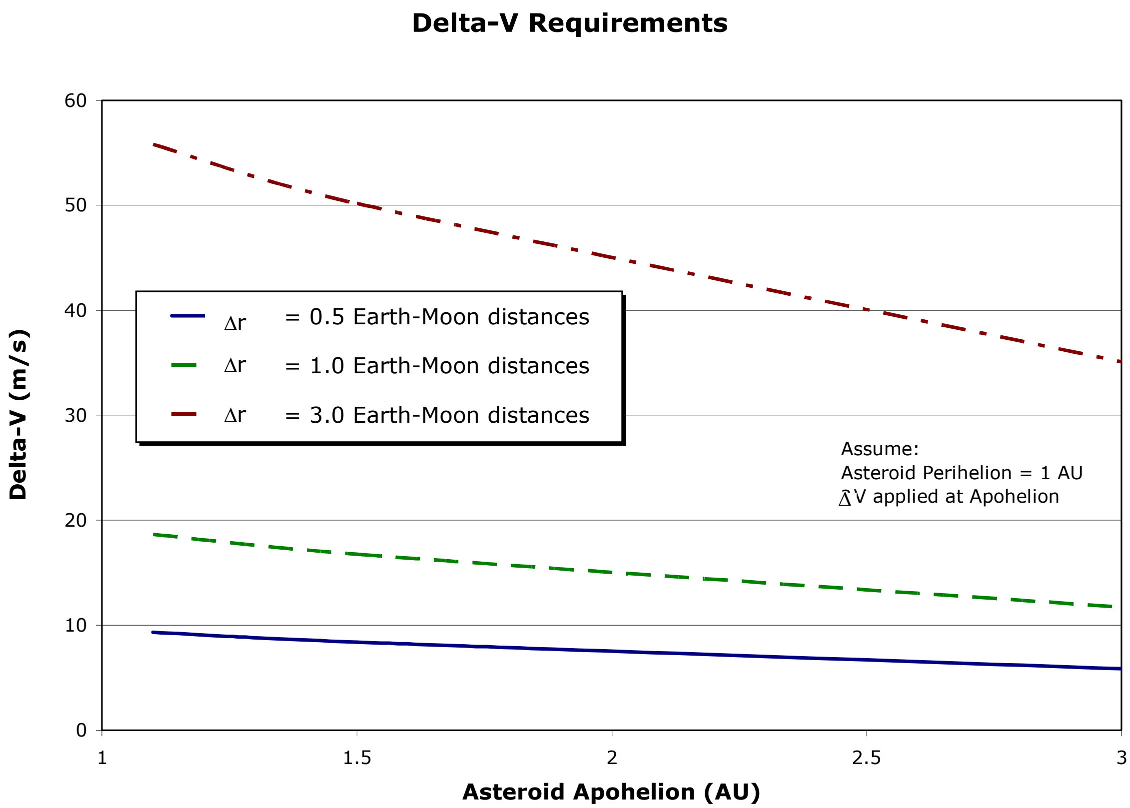


Figure 6.1. ΔV Requirements to Change Impactor Heliocentric Orbit

6.2 EJECTION POWER, ENERGY, AND LANDER FORCE

A series of parametric trades were performed to help determine the appropriate ejecta mass per shot, its launch velocity, and the mass ejected per shot. It is recognized that the momentum imparted to the asteroid in the opposite direction is proportional to the momentum of the departing ejecta (mass per shot times launch velocity). Mass per shot is likely limited by the size and efficiency of the mining/extractor to less than 1 kg per shot for the small spacecraft considered. The temptation therefore is to increase the launch velocity as much as possible. However, there are competing considerations. For the trades conducted, it is assumed that the launch rail should be as long as reasonable but launch packaging and stiffness considerations can limit the launch rail to no more than 10 m.

As the launch velocity increases for a given shot mass, the mass driver power increases in a cubic fashion, thus driving up the size and mass of the mass driver capacitor units, as seen in Figure 6.2. The energy (or work) used to accelerate the ejecta increases proportional to the square of the launch velocity thus requiring a larger spacecraft power supply (or alternately a longer cycle time between shots) in order to recharge the capacitor units.

As shown in Figure 6.3, the compressive force on the lander and rail increases relative to the square of launch velocity. The downward force will benefit the mining process to some degree, but an excessive compressive load would require a massive lander structure and thus exacerbate the launch and deployment problem for the spacecraft.

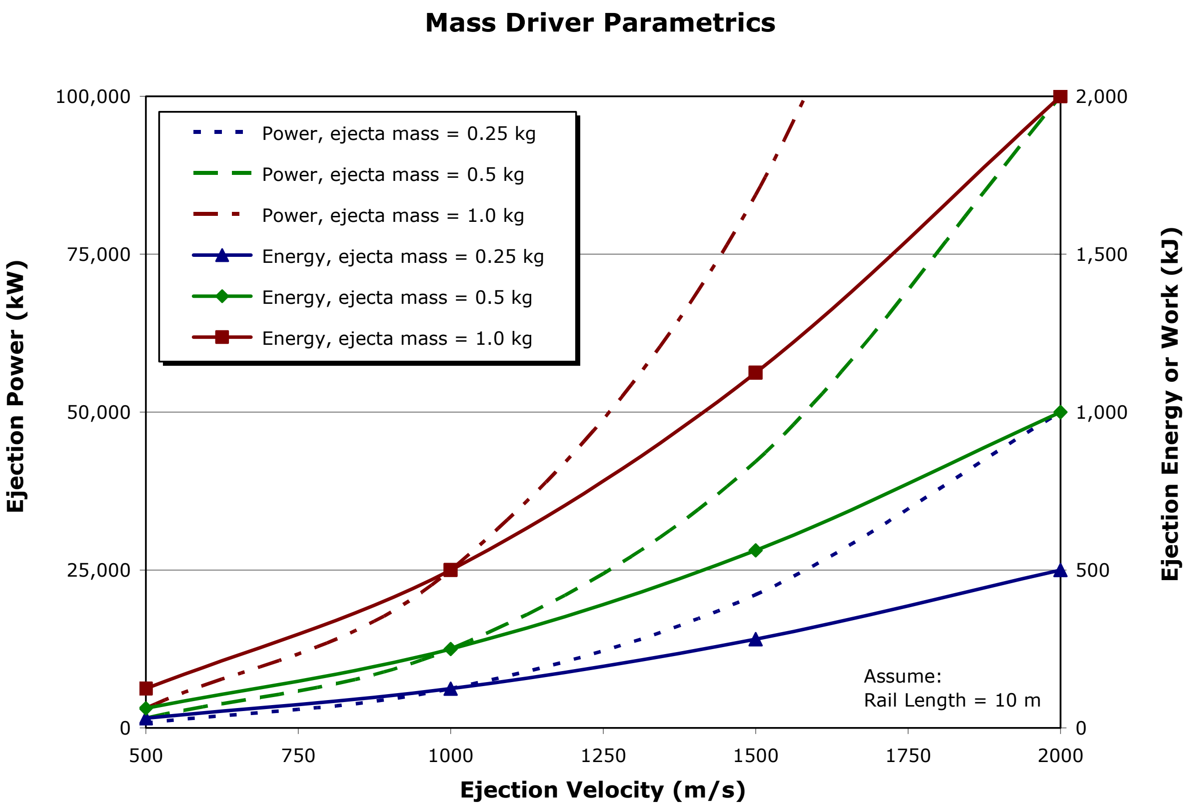


Figure 6.2. Ejecta Launch Power and Energy Trades

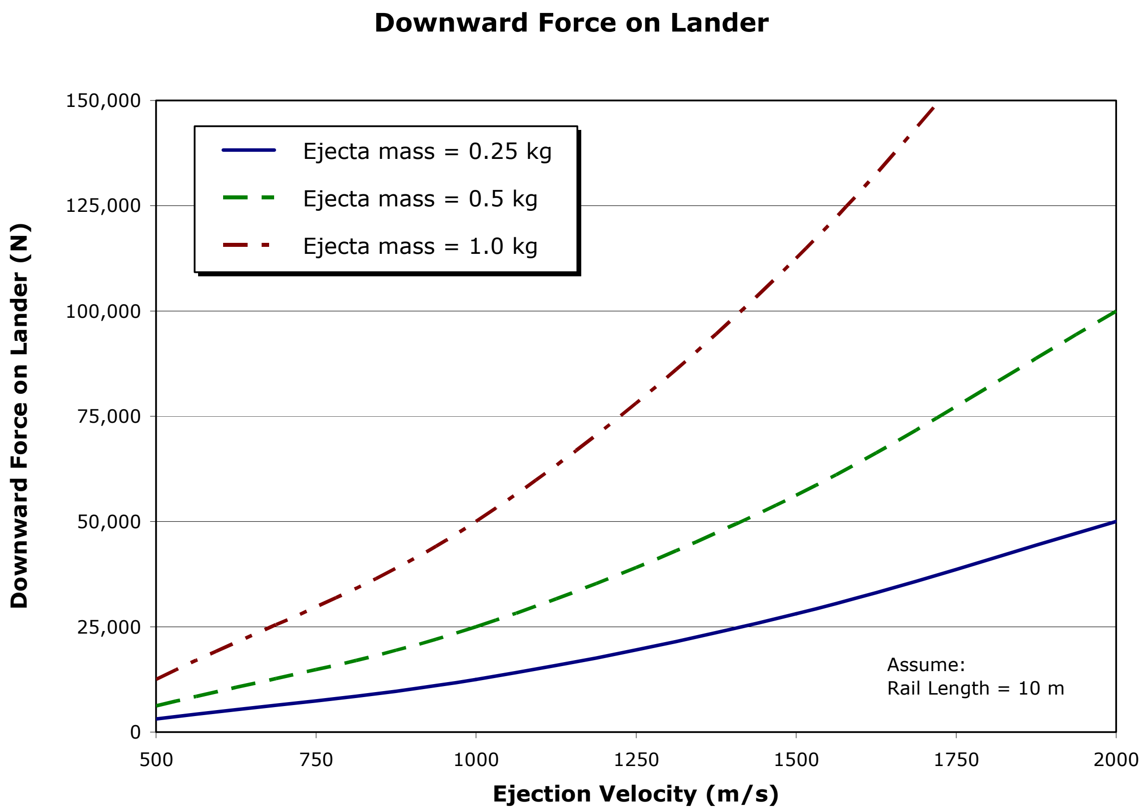


Figure 6.3. Effect of Launch Velocity on Lander Downward Force

6.3 SOURCE POWER TRADE STUDY

While both ejecta launch conditions and rail length determine the mass driver power required per shot, the source power required to recharge the mass driver capacitor units is largely a function of shot frequency, which is how much time is allowed to recharge the capacitors between charges. A fast cycle time will benefit the overall architecture by requiring fewer landers to accomplish the mission in a given time period. However, faster cycle times require larger power sources onboard the landers, thereby raising unit cost and weight.

The relationship between source power and launch power is shown on a log-log scale in Figure 6.4 for three candidate launch rates. A preliminary launch rate of 1.0 shot per minute was initially selected. For a sample configuration (launch power = 12.5 MW) and an assumed power conversion efficiency of 50%, the source power requirement for the mass driver can be determined to be 8.33 kW. Assuming a similar power requirement for mining operations and a 25% margin for spacecraft telemetry and housekeeping power, the lander power requirement is less than 25 kW per lander. Additional, if the spacecraft is to be powered by nuclear electric propulsion, excess power might be available in lander/mass driver mode.

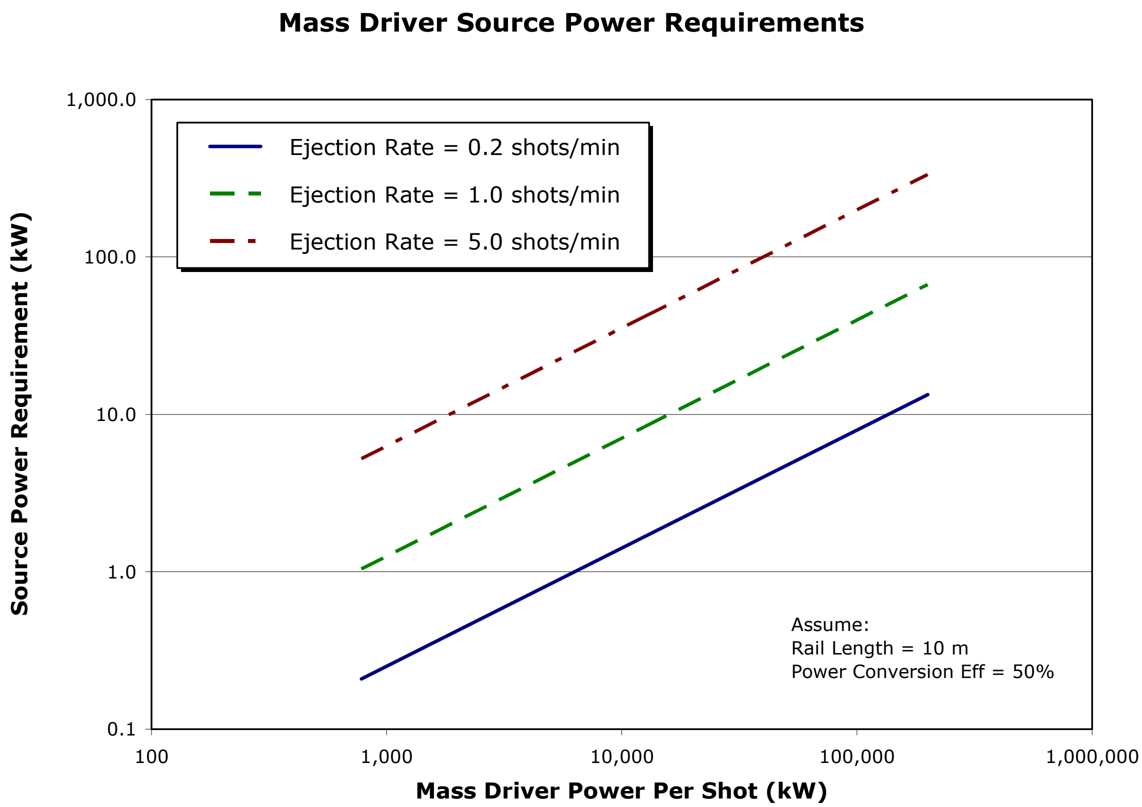


Figure 6.4. Mass Driver Source Power vs. Shot Power

6.4 NUMBER OF LANDER SPACECRAFT REQUIRED

It is assumed that there should be a reasonably short period of action between when the spacecraft swarm arrives at the target and when surface operations can be stated to be complete. While long periods of operation are more efficient, there will certainly be great anticipation by the public that the mission be completed and that there be adequate time to determine post-mission orbit of the asteroid to verify success. With this in mind, we have established a nominal period of action of 60 days to apply the required ΔV to the target impactor.

As shown in semi-log plot of Figure 6.5, the number of landers increases dramatically with the mass of the asteroid to be maneuvered. For an ejection rate of 1 shot per minute with velocity slightly under 200 m/s on a "small" object, approximately the mass of the Tunguska impactor, the number of required lander spacecraft (with specifications based upon the previous sample parametric analyses) is approximately 20-25 depending on the system-level redundancy desired (to impart a ΔV of 0.2 m/s). For asteroids in the 10^{10} kg class (roughly 300 m in diameter) approximately 2,200-2,300 lander spacecraft would be required. For even larger asteroids such as those in the 10^{12} kg class, using the current concept would require over 200,000 landers to accomplish the mission in 60 days.

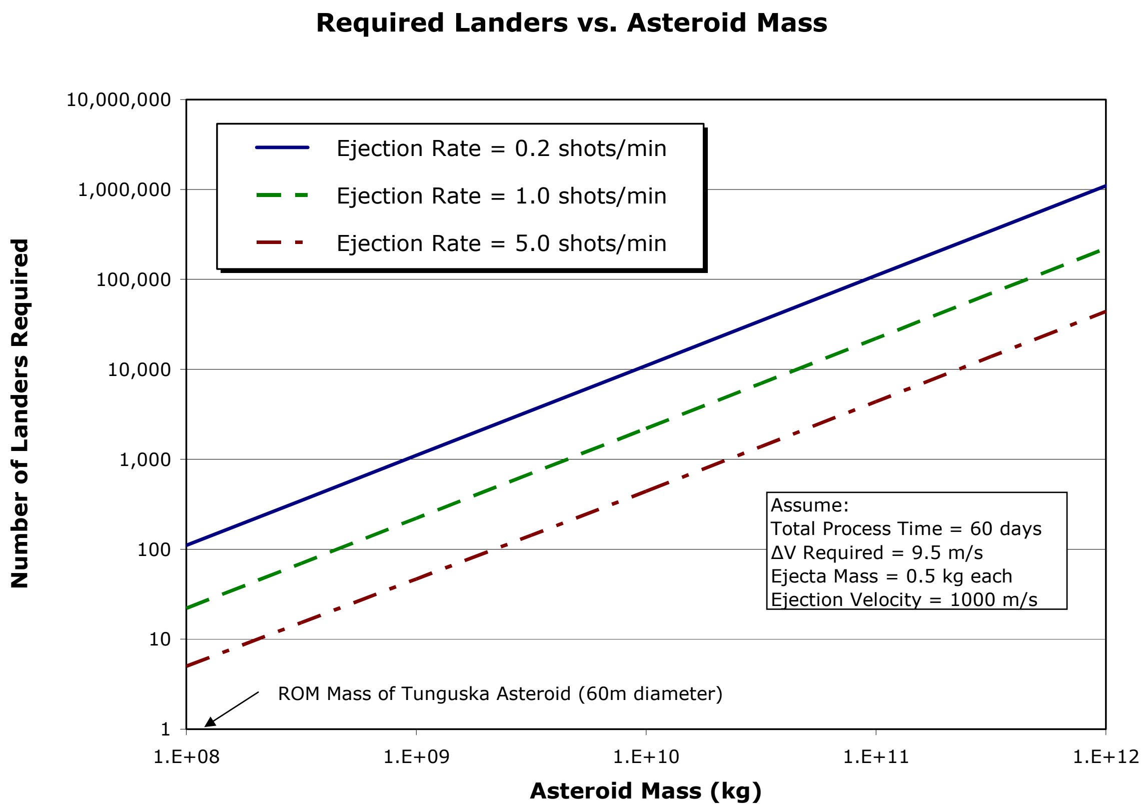


Figure 6.5. Number of Landers Required vs. Asteroid Mass

7.0 DEFT Scenarios

Based upon the most recent data, no current object is known to be in a threatening position with regards to the Earth. No object has been cataloged with a Torino Scale reading above a 2 (only one object has a reading of 1)[16]. Thus fictional threats were obtained from scenarios developed for the AIAA sponsored 2004 Planetary Defense Conference: Protecting Earth from Asteroids (Orange County, California, February 23-26, 2004). Specific guidelines for the threats are stated as follows[17]:

> For the 2004 Planetary Defense Conference, four fictitious Defined Threat (DEFT) scenarios are posed where, without mitigation, solar system bodies (one comet, three asteroids) will strike Earth. These scenarios will encourage detailed designs of rendezvous, intercept and deflection missions and focused discussion of how the world community might prepare for mitigation efforts or possible disaster from policy, public education, and other perspectives… Three of the four scenarios are completely defined (asteroids D'Artagnon and Athos, comet Porthos)…The four scenarios are named for the approaching objects, D'Artagnon, Porthos, Athos, and Aramis. These threat concepts do not and cannot span the range of orbital characteristics, warning times and object properties that could occur. They are, however, plausible possibilities and are intended to provide focus and direction in mission planning.

These scenarios call for smaller miss distances and provide increased detection times than the original design formulated by the just described parametric analyses. The baseline design presented here is rooted around one of these defined threats, D'Artagnon, an asteroid with a mass of 2.7×10^9 kg[17]. Table 7.1 lists the specific properties of the D'Artagnon asteroid used in this analysis. The DEFT scenario for this target states[17]:

> D'Artagnan is an Aten-type asteroid that is discovered by Spaceguard on February 22, 2004. This solid asteroid spends most of its time closer to the Sun than the Earth and is difficult to observe. However, at the time of detection it is close to the apogee of its orbit and partially illuminated by the Sun against the dark sky. Prediction of the nominal orbit indicates that the next time the Earth and D'Artagnan come close, in September of 2009…this object is specifically designed to force consideration of what can be done with today's technology in a "crash program" situation involving a modestly sized asteroid.

For scenarios involving smaller asteroids and long lead times, only a few landers may be required. However, larger threats such as Long Period Comets (LPCs) can require hundreds to thousands of landers, stretching the limits of manufacturing capability and launch capacity. Due to uncertainties in the size and orbit of each object, additional lander spacecraft can be sent to ensure adequate ΔV is applied.

Table 7.1. Defined Threat Specifications for D'Artagnon[17]

Item	Value
Time/Date of Detection	February 22, 2004 00:00:00: UT
Expected Date of Impact	September 14, 2009 11:04:26.117 UT
Approximate orbital elements at time of detection	q (perihelion distance) = 0.639030 AU e (eccentricity) = 0.288063 I (inclination) = 4.788754 degrees Ω (right ascension of ascending node) = 350.540144 degrees ω (argument of perihelion) = 230.750220 degrees M (mean anomaly at time of detection) = 254.275083 degrees Period = 0.849613 years
Type	Type S Asteroid
Size	130 m x 120 m x 110 m
Mass	2.7×10^{12} g ±40%
Density	3 ± 1 g / cm^3

For the analysis presented here, the D'Artagnan impactor is chosen as the test case for the MADMEN architecture. The assumed mass is 2.7×10^9 kg with a density of 3 g/cm^3. The assumed total ΔV to be imparted to the impactor over the course of surface operations is 1.0 m/s, equivalent to moving the impactor approximately 0.053 Lunar Distances (Earth-Moon distances) or 1.59 Earth diameters if this ΔV is applied simultaneously at the apohelion of the impactor's orbit (the best case scenario). The ΔV for the MADMEN architecture is not applied instantaneously but for this analysis it is assumed to be equivalent to an instantaneous change.

8.0 Baseline MADMEN Lander Design

8.1 OVERVIEW

A baseline configuration was established for the MADMEN spacecraft in which specific top-level assumptions were made about various subsystems in terms of enabling and enhancing technologies. For this preliminary baseline, architecture, it is assumed that a completely separate in-space transfer stage (ISTS) would be used to transport the MADMEN spacecraft to the target impactor.

The MADMEN lander spacecraft consists of several primary components (see Figure 8.1). A nuclear reactor provides moderate power levels over long periods of time for the entire system. High energy density capacitors are charged for each individual shot, reducing the total load on the reactor. A mining system, similar to terrestrial coring drills, is used to extract material from the impactor (see Figure 5.8). A swivel mounted drill bores into the NEO beneath the lander, extracting material from a moderate sized tube tens of centimeters in diameter and several meters long. The material within the tube is removed, processed, and launched from the ejector. The drill is then swiveled to a new position and the drilling process repeated, allowing each lander to access a reasonable volume of material while firmly anchored into place.

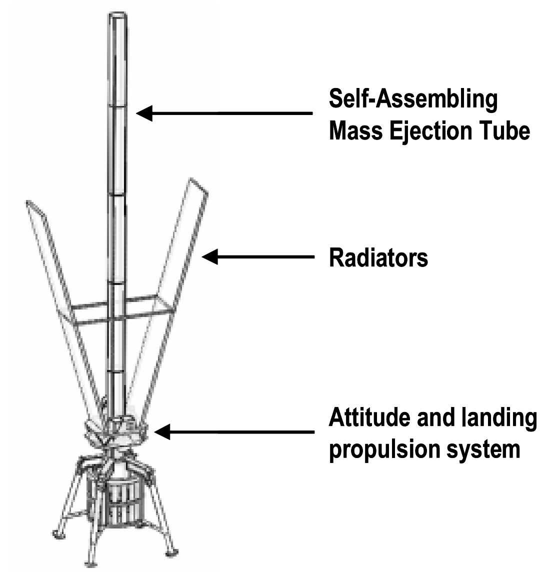


Figure 8.1. Lander Spacecraft Primary Components

The ejector itself is a mass driver system consisting of a series of electromagnets arranged axially around a central barrel. A magnetized "bucket" containing ejecta is propelled along the barrel by the alternating attractive and repulsive forces of the mass driver electromagnets. The ejecta are allowed to leave the barrel while the bucket is slowed down and some of its energy recovered via regenerative braking. The bucket can then be reused for additional firings. The high downforce from the mass driver is the primary driver for the structural weight and size. The reactor, mining system, and other subsystems must be supported, but a higher load is transferred along the mass driver's structure, through the landing struts and anchors, and into the impactor.

Additional subsystems include thermal control, data processing, navigation and sensing, and communication systems. Small attitude control system (ACS) thrusters provide the ΔV to maneuver the lander from a transfer vehicle to the impactor, land on the impactor, and anchor into place.

8.2 SYSTEM SIZING

A Reduced Order Simulation for Evaluation of Technologies and Transportation Architectures (ROSETTA) modeling process was employed to develop a parametric performance and sizing model. A ROSETTA model is a spreadsheet-based meta-model which is a representation of the design process for a specific architecture (e.g., ETO, in-space LEO-GEO, HEDS)[18]. ROSETTA models contain representations of a baseline design into which technologies can be infused. Using the baseline requirements, the mass ejector landers were designed and sized from both historical data and physical relationships[19,20] Input in-space trajectory data and in-space transfer stage (ISTS) sizing disciplines were also added to the ROSETTA model.

For the design of the MADMEN lander spacecraft various parameters needed to be examined in order to arrive at an optimum design. As seen in Table 8.1, these multiple design variables mainly included those associated with the ejecta (mass, volume, speed) and surface operation time while the constraints mostly originated from launch vehicle constraints (total volume, total mass, track length). The specific optimization objective in this design process was the minimization of the total number of spacecraft required for the particular target (see Table 8.2).

Table 8.1. Baseline Madmen Lander Design Trade-Off Parameters

Item	Main Effects
Ejection Velocity	Launch Energy Down Force Launch Power
Ejecta mass per shot	Launch Energy Core Size Total Mass Ejected
Operating Time on Target Body	Number of Landers Public Confidence
Mass Driver Track Length	Launch Power Down Force
Shot Frequency	Reactor/Capacitor Size Trade Number of Landers

Table 8.2. Baseline Madmen Lander Design Constraints and Objective

Mission Phase	Components
Constraints	Operating Time on Target Body Launch Vehicle Packaging Limit on Ejecta Size Mass Driver Track Length Launch Vehicle Mass to c_3 = 0 km^2/s^2
Objective	Minimize the total number of spacecraft required for the particular target

The ROSETTA model was optimized using a genetic algorithm from the OptWorks suite of Excel-based non-gradient based optimizers (Pi Blue Software, Inc.). Due to the complex nature of the design space, a simple gradient-based optimizer would stall on local minima. A genetic algorithm optimizer can handle such multiple local minima and better attempt to determine the global minimum within a design space. The genetic algorithm also provides the option of using discrete variables for discrete choices, such as material type, which can be used for later trade studies. For the optimization the design variables, limits, and final values are given in Table 8.3. The objective function was the total number of landers required for the particular impactor (in this case the D'Artagnon defined threat).

Specific constraints on the optimization included a surface operation time constraint of 60 days, shot frequency constraint given a 2 second delay between shots, and a launch vehicle constraint for the launch vehicle stack

(MADMEN lander spacecraft and in-space transfer stage) to be less than 9,305 kg which is the launch mass of a Delta-IV Heavy to a c_3 of 0 km^2/s^2.

Table 8.3. MADMEN Lander Spacecraft Optimization Parameters

Item	Minimum	Maximum	Final Value
Core diameter (m)	0.01	1.00	0.1935
Ejection Velocity (m/s)	1.0	2000.0	186.83
Ejecta mass per shot (kg)	0.05	2.00	2.0
Rail Length (m)	0.50	10.00	10.0
Shot frequency (per minute)	1	30	1.0
In-Space Transfer Stage (ISTS) tank diameter (m)	1	4	2.769

The minimization of lander mass generally tended towards smaller ejection velocity with a lager ejecta mass per shot in fewer shots per minute. Since power equals force times velocity, force equals mass times acceleration, and acceleration equals velocity squared over two times the rail length, then power is related to the third power of velocity. In addition, the mass of the mining system is related to the third power of core diameter. Thus these two design parameters (ejection velocity and core diameter) have a large influence on the final characteristics of the system. Rail length tended towards the maximum given the inverse relationship between power (acting as a proxy for mass and directly related to it) and rail length. The shot frequency tends towards the minimum since more shots per minute entails more power required (power also equals work divided by time) resulting in a larger reactor mass. For example, starting from the final optimization parameters listed above, an increase from one shot to thirty shots per minute results in a source power requirement increase of over 200%. An additional optimization parameter given over to the genetic algorithm was the tank diameter of the in-space transfer stage (ISTS). This parameter was constrained by the dimensions of the launch vehicle.

From these preliminary results, a working baseline of a 10 m launch rail, 2.0 kg ejecta mass per shot, and a launch velocity of 186.83 m/s (well within the capability of today's rail launchers) was utilized. For this configuration, the ejecta will undergo an acceleration of almost 178 Earth g's for a period of 0.11 seconds causing a downward force on the lander spacecraft of 3.49 kN. The mass driver power required is 0.798 kW per shot (includes 36% additional power for the bucket fraction). The energy consumed per shot is 47.5 kJ.

The mining system is the largest single lander element, requiring a large drill and tubing to core out a sufficient volume of the impactor. The rapid shot rate requires a heavy reactor and the large downforce necessitates increased structure, particularly along the ejector. The main structure of the MADMEN lander spacecraft consists of titanium for the anchors, supports, mining frames, vehicle frame, and ejector frame. The propulsion system for the MADMEN lander spacecraft was based upon a hydrazine mono-propellant system with an Isp of 220 seconds. Additional propulsive ΔV was assumed including 50 m/s for cruise stage egress and 100 m/s for impactor landing.

Given the large power requirements for the drill and mass ejection system, the baseline MADMEN lander includes a nuclear reactor. Solar power systems, even at these nominal distances from the Sun (within 1.25 AU) may still not be able to provide sufficient power required for the drill and mass ejection system. In addition, the environment around an impactor's drilling site would include many dust particles (existing and those that develop due to drilling) and cause shadowing effects due to the body's rotation, thus posing substantial hazards to a solar power generation system. This baseline design includes a small (under 100 kW) particle bed reactor (PBR) system with associated shielding, power conversion, conditioning, and rectangular heat rejection radiators[21]. Figure 8.2 relates the power schematic, showing the transfer of both power and thermal energy.

A system-wide power efficiency chain is developed that attempts to account for power requirements for the power conversion, power conditioning, mining, and mass driver systems. Reactor power conversion is based upon an advanced, lightweight thermionic system with conversion efficiency of 30% (see Table 8.4). Cabling and other losses are estimated at less than 1%. The major need for power originates from the mass driver system. The mass driver requirement of 10 kW of power leads to a need of 11.56 kW of electrical power requiring a total reactor thermal power level output of 42.23 kW (See Table 8.5). The sizing and mass estimating assumptions for the MADMEN power system are listed in Table 8.6.

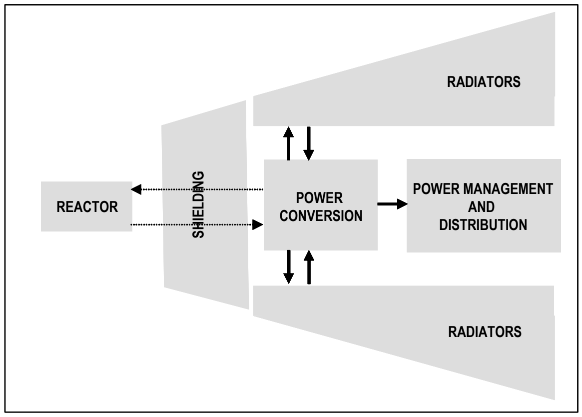


Figure 8.2. Components of Power Subsystem

Table 8.4. Power Efficiency Chain: MADMEN Lander Spacecraft

Efficiency Item	Value
η-other	99.5%
η-cabling	99.5%
η-shielding	99.0%
η-power-conversion	30.0%
η-power-conditioning	95.0%
η-propellant-feed-system	95.0%
η-mining	95.0%
η-driver	95.0%

Table 8.5. Top-Level Power Budget: MADMEN Lander Spacecraft

Item	Power [kW]
Thruster Power Required	0.010
Propellant Feed System Required	0.010
Mining Power Required	10.000
Mass Driver Power Required	0.798
Hotel Load Required	0.025
Science Load Required	0.010
Communication Load Required	0.025
Total Load Required	10.878
Total loss: other	0.537
Total loss: cabling	0.535
Total loss: shielding	0.418
Total loss: power-conversion	28.685
Total loss: power-conditioning	0.609
Total loss: propellant-feed-system	0.001
Total loss: mining	0.526
Total loss: driver	0.042
Total losses: all	31.353
Total Power Required from Reactor	42.231

Given the optimization parameters, subsystem specifications, and launch vehicle constraints, the total dry mass of a MADMEN lander spacecraft for the D'Artagnan impactor is 1,593 kg (gross mass of 1,718 kg). Table 8.7 relates the specific parameters that describe each individual MADMEN spacecraft. A two-level mass breakdown statement (MBS) is given in Table 8.8 and illustrated in Figure 8.3. For this case, the total number of MADMEN lander spacecraft required to be functioning for the entire surface operation phase of 60 days was 85 landers (to impart a ΔV to the target of 1.0 m/s).

Table 8.6. Power Subsystem Sizing Assumptions

Item	Value
Nuclear reactor core density [kg/m^3]	1500
Radiation shield density [kg/m^3]	1500
Specific mass of power conversion [kg/kWe]	2.0
Specific mass of PMAD/ power conditioning [kg/kWe]	0.70
Radiator Effective Mass/Area [kg/m^2]	4.0
Capacitor Unit Mass [kg/kJ]	0.5

The largest portions of the MADMEN lander spacecraft consists of the mining and power system. Given the small amount of propulsive ΔV required, the propulsion system (including propellants) is a relatively minor component of total gross mass. A scale comparison of the MADMEN spacecraft to both the Lunar Module (LM) and Soviet LK Energia is shown in Figure 8.4.

Table 8.7. Baseline MADMEN Lander Spacecraft Summary Parameters

Item	Value
Ejection velocity	187 m/s
Ejecta mass per shot	2 kg
Rail length	10 m
Shot frequency (per minute)	1 per minute
Total surface time of process	60 days
Total power required	42.2 kW
Dry Mass / Gross Mass	**1,593 kg / 1,718 kg**

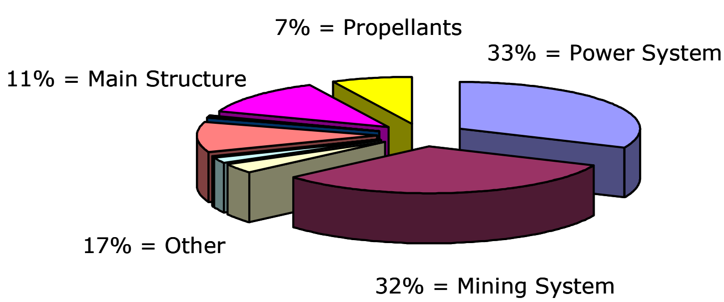
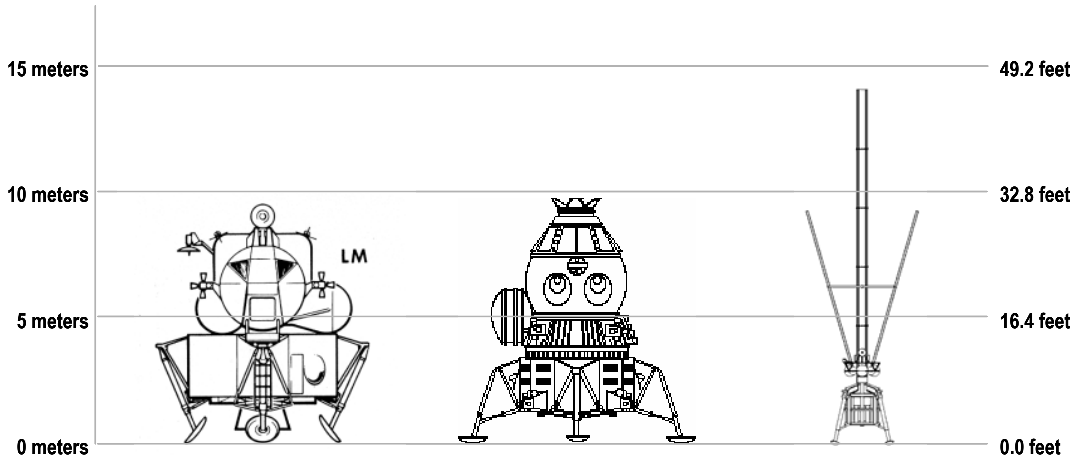


Figure 8.3. Lander Spacecraft Mass Breakdown



Apollo LM	LK Energia	MADMEN Lander	VEHICLE NAME
15,100 kg	**10,300 kg**	**1,593 kg**	**DRY MASS**
9.39 m	9.81 m	13.97 m	Length
6.37 m	9.51 m	2.54 m	Height
9.39 m	9.51 m	2.54 m	Width

Figure 8.4. MADMEN Lander Scale Comparison

Table 8.8. Baseline MADMEN Lander Spacecraft Mass Breakdown Statement (MBS)

Item	Mass (kg)
1.0 Power System	555.4
Reactor	204.4
Energy conversion	61.3
Capacitors	21.5
Power conditioning	23.7
Shielding	244.4
2.0 Mining System	517.1
Drill	193.5
Main bit	19.4
Tubing	164.2
Ore Processing	140
3.0 Ejector	54.2
Mass Driver	52.4
Bucket	1.8
4.0 Propulsion	31.2
RCS Thrusters	24.0
Tanks	4.5
Lines	2.7
5.0 Thermal Control	39.0
Radiators	39.0
6.0 Main Structure	171.8
Anchors	3.5
Anchor Drive	20.0
Supports	8.7
Mining Frame	17.5
Vehicle Frame	34.9
Ejector Frame	87.3
7.0 Data Processing	8.0
Attitude/Orbit Determination	2.0
Attitude/Orbit Control	2.0
Device Pointing	2.0
Integrated Function	2.0
8.0 Navigation Sensing/Control	4.0
Celestial/IMU	4.0
9.0 Telecom and Data	4.0
TCM Module	2.0
Command and Data Handling	1.0
Communications Payload	1.0
10.0 Margin (+15%)	207.7
Total Dry Mass	**1,592.7**
11.0 RCS Propellants	124.9
Cruise stage egress	41.3
Main	40.3
Reserves and residuals	1.0
Impactor landing	83.6
Main	81.5
Reserves and residuals	2.1
Near Earth Departure Mass	**1,717.5**

9.0 In-Space Transfer Stage (ISTS)

9.1 OVERVIEW

Given the modular nature of the MADMEN lander spacecraft, a similar philosophy was developed for the components needed to get these spacecraft to their target body. For this baseline architecture the MADMEN spacecraft are sent directly from Earth to the target body. The in-space transfer stage (ISTS) transports the MADMEN lander spacecraft to the target using conventional rocket engines performing a two burn sequence for earth escape and target capture. Once near the target body, the ISTS releases a MADMEN lander spacecraft and it performs the final maneuver towards the surface. For this initial assessment, a top level design decision was to place integer number of MADMEN lander spacecraft on each ISTS such that the entire stack would fit on currently available expendable launch vehicles (see Figure 9.1). No in-space operations segment was assumed for this analysis.

Figure 9.1. Notional Concept Illustration of MADMEN Lander Spacecraft and In-Space Transfer Stage (ISTS)

9.2 TRAJECTORY MODELING

Based upon the chosen asteroid threat, D'Artagnon, a patched-conic analysis was performed of the required in-space trajectory to the target. The approach utilized an ephemeris for the target body and a Lambert-solver to calculate the hyperbolic velocity (c_3) and ΔV values with respect to the departure and arrival bodies (Earth and the impactor). From Earth escape, a simple two-burn transfer was assumed to capture into the orbit of the target. These two burns consist of a transfer orbit/earth escape injection and an impactor capture burn. Based upon the detection information for the target listed in Table 7.1, an approximate position for D'Artagnon was determined at the date of detection.

It was assumed that the Earth-To-Orbit (ETO) launch vehicle, nominally the Boeing EELV Delta IV-Heavy, will provide the needed velocity to achieve earth escape (with $c_3 = 0\ km^2/s^2$). A Delta IV-Heavy (4250H-19 with 5m x 19.1m composite dual manifest fairing) vehicle can launch approximately 9,305 kg to a c_3 of 0 km^2/s^2 to achieve earth escape[22]. This was the launch constraint used in the ROSETTA model optimization described earlier.

This trajectory analysis attempted to find the lowest total c_3 solutions for the D'Artagnon target. The minimum total c_3 accounts for departure and arrival maneuvers. The square root of this total is approximately the total ΔV required for both maneuvers. This assumes that the launcher provides c_3 of zero at departure and that the MADMEN propulsion is not activated until the vehicle is nearly out of the gravity well of Earth. An additional assumptions if that the gravity well of the asteroid is negligible.

A cyclic pattern of minimum ΔV versus Earth launch year emerges (see Table 9.1). As the positions of Earth and the target change, the ΔV ranges from a low of about 3 km/s to a high of about 8.5 km/s over a four to five year period (see Figure 9.2). To avoid a worst case, it was assumed that the actual first launch date would occur in 2008 from a date of detection of 2004. This reflects an intermediate trajectory case, though it is perhaps unreasonable to assume that one can always wait for "perfect" alignment. For this chosen year, the minimum c_3 date for the year 2008 yielded a ΔV of 5.423 km/s.

Table 9.1. D'Artagnon - Minimum Total c_3 Trajectories (1 day steps, 365 day search from January 1 each year)

Departure Date	Time of Flight [Days]	Departure, Arrival, and Minimum c_3 [km^2/s^2]	Approximate ΔV [km/s]
10/19/2004	214	12.08,15.38,24.47	5.241
10/3/2005	177	32.64, 11.59, 44.23	6.651
12/31/2006	326	37.34, 29.48, 66.82	8.174
2/19/2007	435	16.12, 26.92, 43.04	6.561
2/25/2008	**367**	**10.51, 18.90, 29.41**	**5.423**
3/5/2009	285	9.12, 11.37, 20.49	4.526

Given these assumptions (along with a 5% margin), the total ΔV required for the in-space transfer portion of the trajectory to capture into the impactor's orbit was set to 5.694 km/s (see Figure 9.3). This will be the velocity change required by the in-space transfer stage (ISTS) to get a MADMEN spacecraft to the target. Additional ΔV required for the MADMEN spacecraft themselves for cruise stage egress and impactor landing was set to 50 and 100 m/s respectively. The in-space ΔV can be sub-divided into roughly 1.939 km/s for Earth escape and 3.484 km/s for impactor capture. Arrival at the D'Artagnon target occurs on 2/26/2009 allowing approximately less than 7 months until Earth impact. For this baseline design this is the time period wherein the swarm of MADMEN spacecraft will have to impart a velocity change of 1 m/s to the impactor. This time period of surface operations is artificially constrained to 60 days. This is a assumed to be the maximum time period society will accept for this architecture to perform.

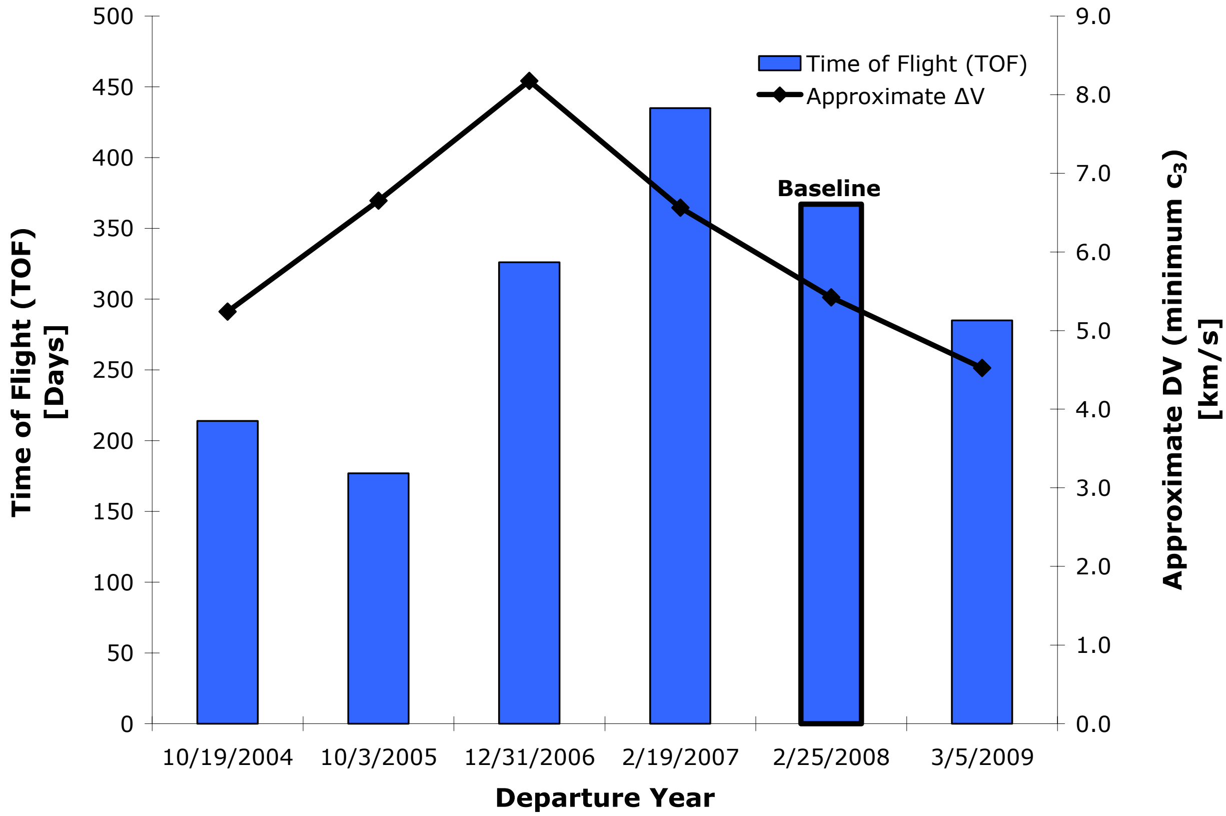


Figure 9.2. D'Artagnon - Minimum Total c_3 Trajectories
(1 day steps, 365 day search from January 1 each year)

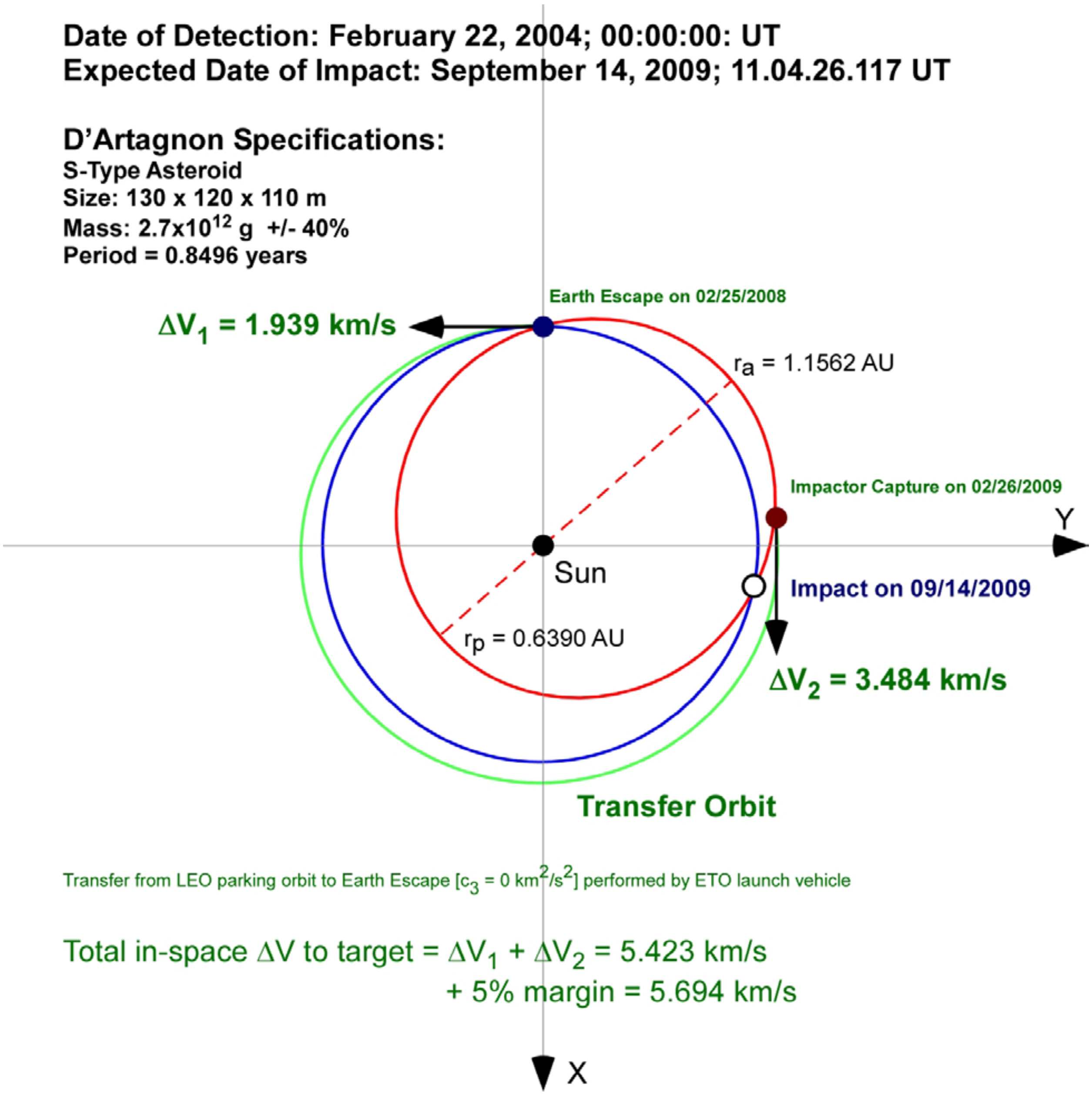


Figure 9.3. D'Artagnon Trajectory Profile

9.3 SYSTEM SIZING

The In-Space Transfer Stage (ISTS) was sized in the ROSETTA model described previously in this report. Parametric sizing models were created for the ISTS and several associated subsystems (notably telecommunications). The initial transfer stage was designed to be a liquid oxygen / liquid hydrogen propulsion system using RL-10A-4-2 engines with an Isp of 433 seconds. The ISTS include both a higher power Earth telecommunications system and independent power system. With the addition of such systems, the ISTS has been designed for possible use as an orbital relay.

The ISTS sizing process took the mass of the MADMEN spacecraft as an input. Since the mass of each of those spacecraft was relatively large (over 1 MT each) and the payload of the Delta IV-Heavy is about 9.306 MT, only one spacecraft was placed on each transfer stage. Given this payload for the ISTS, total stack mass on the Delta IV-Heavy is around 9.25 MT as seen in Table 9.2, just within the limits of the launch vehicle to a c_3 of zero (earth escape).

Table 9.2. Baseline Mission and In-Space-Transfer Stage Summary Parameters

Item	Value
Delta-V imparted to Impactor	1 m/s
Impactor Mass / Diameter	2.7 x 10^9 kg / 130 m
Delta-V to get to Impactor	5,423 m/s
Dry Mass / Gross Mass with Payload	**2,316 kg / 9,252 kg**

Unlike the MADMEN lander spacecraft, the ISTS is mostly propellant. The majority of the rest of the ISTS gross mass is the individual MADMEN lander spacecraft itself. A detailed mass breakdown is laid out in Table 9.3, while the mass contributions by percentage of the major line items are shown in Figure 9.4.

Table 9.3. Baseline In-Space Transfer Stage (ISTS) Mass Breakdown Statement (MBS)

Item	Mass (kg)
1.0 LH2 Tank Structure	103
2.0 LH2 Tank Insulation	72
3.0 LOX Tank Structure	35
4.0 LOX Tank Insulation	38
5.0 Propulsion	187
6.0 Telecommunications	4
7.0 Subsystems	40
8.0 Other Structure	48
9.0 Margin (+15%)	72
Total Dry Mass	**598**
Total Payload Mass (Lander)	1,718
Impactor Arrival Mass	**2,316**
Required Propellant	6,936
Pre-Injection Mass	**9,252**

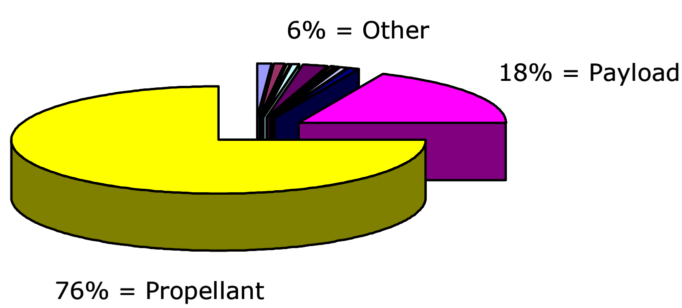


Figure 9.4. In-Space-Transfer Stage Mass Breakdown

10.0 Architecture Success and Cost

10.1 MISSION SUCCESS

There is uncertainty inherent in all aspects of interplanetary missions. Some portions of a mission may have a higher probability of success than others. The baseline MADMEN mission described above is not immune to these challenges. Multiple backups of these spacecraft are required since the ultimate objective is to achieve 100% mission success. Unlike other missions, the societal desire is for these missions to be completely successful.

A top-level determination was made of the probability of mission success given the various portions of the mission. The entire mission was broken out into three phases: transfer, activation, and operations. The transfer portion deals with both Earth launch, cruise to the impactor, and landing. Activation involves the activities to start the process of mass ejection including rail extensions and drill activation after cruise. The operations phase includes the nominal drilling and ejection process as well as communication between swarm components to coordinate mass ejection.

Qualitative assessments were made by the authors in regards to the probability of success (see Table 10.1). The transfer phase had the lowest probability given a low value for success on impactor landing. Given the inputs, the overall success rate was 0.4371. For the baseline MADMEN case, the total number of spacecraft required at full functionality for a full lifetime to perform the mission is 85, but including the chance of failure, the total number of spacecraft actually required is 195.

Table 10.1. Overall System Probability of Success Per Mission Phase

Mission Phase	Components	Value
Transfer	Launch (with stage separation)	0.9850
	In-Space Earth Assembly	1.0000
	Earth Escape Burn	0.9950
	In-Space Trajectory	0.9995
	Impactor Capture Burn	0.9800
	Transfer Stage Separation	0.9500
	Transfer Stage Egress Burn	0.9800
	Impactor Landing Burn	0.9500
	Impactor landing	0.7500
Total Transfer Success		**0.6368**
Activation	Rail extension	0.9500
	Reactor power	0.9900
	Drilling Activation	0.9800
	Driver Activation	0.9800
Total Activation Success		**0.9033**
Operations	Surface operations	0.8000
	Swarm communication	0.9500
Total Operations Success		**0.7600**
Overall Mission Success		**0.4371**

Given the particular asteroid threat for this analysis, D'Artagnon, 195 MADMEN lander spacecraft are required to change the approximately 100 m wide asteroid's velocity by 1.0 m/s during a surface operation time of 60 days. This correlates to 195 Delta IV-Heavy launches. Generally, the launch capacity of current expendable launch vehicles (fewer than 10 MT mass to earth escape) precludes any more massive systems than the MADMEN lander spacecraft shown.

10.2 ARCHITECTURE COST

A life cycle cost (LCC) analysis was performed on the baseline MADMEN architecture for the D'Artagnon impactor case. Design, Development, Testing, and Evaluation (DDT&E) and Theoretical First Unit (TFU) costs were determined. No development costs were assumed for the in-space transfer stage (ISTS) since this component is based upon already existing and well defined systems. Appropriate rate and learning curves were applied to MADMEN lander spacecraft, ISTS, and Delta-IV launch vehicle purchases. A facilities cost of $220 M was determined along with a recurring operations cost of $2 M per spacecraft. Nominally only one year of operations was assumed. Table 10.2 lists the LCC summary. A total LCC of $53.454 B (FY2004) is required for development, acquisition, launch, and operations for the 195 MADMEN lander spacecraft and associated ISTS components. This amounts to approximately a recurring spacecraft mission cost of $274 M, similar to the cost limits on current NASA Discovery-class missions. The total amount is large, but it is assumed that if a legitimate threat were to manifest, society would be willing to allocate such fiscal resources for this project.

Table 10.2. Life Cycle Cost Summary

Item	Total Cost [$M]: FY$2004	Cost / Lander Spacecraft [$M]: FY$2004
Lander Spacecraft	1,249	6
DDT&E	**1,249**	**6**
Lander Spacecraft	18,758	96
In-Space Transfer Stage (ISTS)	4,297	22
Acquisition	**23,055**	**118**
Facilities	**220**	**1**
Operations	**22,539**	**2**
Launch	**28,539**	**146**
Total	**53,453**	**274**

11.0 Baseline Sensitivity Analysis

The leading factors contributing to high required launch rates include the large mass of the individual MADMEN spacecraft, the large ΔV to be applied to the target, and the length of mass driver surface operations of the spacecraft. A sensitivity analysis can be performed to examine the nature of the architecture with regards to these parameters.

Figure 11.1 illustrates the sensitivity of the total number of required landers to the ejecta mass per shot and the ejection velocity. Meanwhile, Figure 11.2 shows the relationship between the total mass required at near Earth departure and the mass and velocity of each ejecta shot. It can be seen from these figures that while firing ejecta at higher velocities and at higher mass levels per shot results in fewer total spacecraft required, the total mass at near Earth departure increases dramatically.

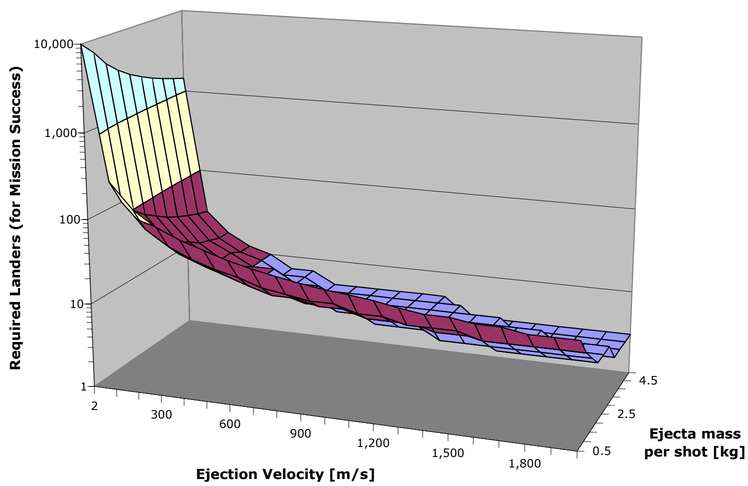


Figure 11.1. Sensitivity of Total Required Landers to Ejection Velocity and Ejecta Mass per Shot

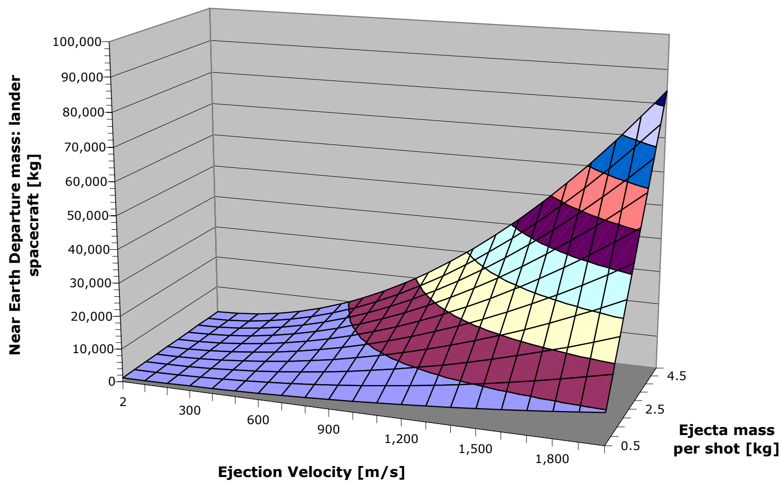


Figure 11.2. Sensitivity of Near Earth Departure Mass to Ejection Velocity and Ejecta Mass per Shot

A final sensitivity trade was conducted to examine the dependence of the total spacecraft required on the mass of the target asteroid and the total surface operation time. As seen in Figure 11.3, the greatest influence on the required MADMEN swarm size is the asteroid mass. However, the effect of surface operation time becomes increases in significance when dealing with more massive asteroids.

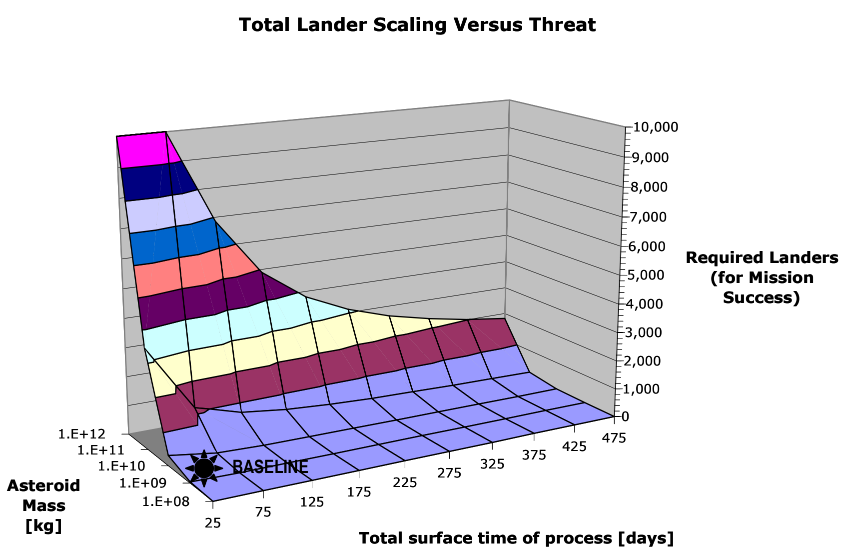


Figure 11.3. Sensitivity of Total Required Landers to Asteroid Mass and Total Surface Operation Time

These preliminary results do not reflect a complete system optimized solution to the planetary deflection issue. Additional options and architecture variations could reduce the spacecraft size and subsequent number of required ETO launches.

12.0 Future Work

Additional examination of the MADMEN concept would be beneficial in determining the optimum design of the spacecraft and architecture. The main objectives of this future work would be to examine particular aspects of the concept in depth. Another objective of these further refinements would be to attempt to reduce the total spacecraft mass required per asteroid.

One of the main aspects that should be examined would include an actual simulation of the MADMEN swarm similar to modeling of terrestrial autonomous robots. This would help to understand the interactive nature of the swarm in order to determine the true number of spacecraft required. This high fidelity, near real time simulation of the environment and spacecraft swarm around a target impactor would include input specifications for the spacecraft and impactor, time step simulation of the actual process of landing, communication, drilling, and ejecting, using both orbiting nodes (potential for use of additional use of cruise stage) and landers. Additionally, the simulation could be coupled with a Discrete Event Simulation (DES) of failures.

Future work would also include increasing the fidelity and depth of the lander and transfer stage design. This would include a more extensive examination and modeling of drill subsystem with various trade studies based upon different composition (suitability of approach to rock pile versus stony-type asteroid impactor)

Additionally, more mission related studies can be performed. A more complex and realistic transfer trajectory model could be developed that would take into account the rotation of the impactor body and attempt to model the entire MADMEN swarm and impactor system as a continuous low thrust propulsion system. The trade between delaying launch time for smaller ΔV's to the rendezvous and increased numbers of landers required for shorter times on the impactor could be developed. An examination is needed for the specific sensitivity of mission parameters to target detection time/transfer time and on-site excavation time. This could be coupled with studies on the utility of pre-deploying large numbers of MADMEN spacecraft in the Earth-Moon or Earth-Sun libration points (vs. simple rapid response from ground launchers). Other possible studies could include determining the most favorable in-space transfer approach to maneuvering the MADMEN spacecraft to the staging point or toward the target body (chemical vs. low thrust, for example). A summary of specific trade studies to be addressed during a possible Phase II study are listed in Table 12.1.

Table 12.1. Proposed Future Studies

Study Type	Options
High Fidelity Simulation	Near real time simulation of near impactor proximity operations
General detailed subsystem modeling	Enhancement of the ROSETTA design model
Pre-Deployment	Earth-Moon L2, Earth-Sun L1, Earth-Sun L2, or Ground versus simple rapid response from the Earth
Mass Driver Velocity	Variable (approx 50 m/s – 500 m/s)
Mining technology	Mechanical, Acoustic, etc.
Landing configuration	More extensive examination and modeling of landing and attachment issues, Impact landing vs. "soft-land"
Mass Driver Power (per spacecraft)	Variable (500 W – 50 kW)
Architecture Configuration:	Examination of an ANTS swarm like configuration (subsets of similar spacecraft) with subspecialties
Failure Analysis	Continued analysis of potential failure scenarios, more quantitative justification for failure rates, Discrete Event Simulation (DES) of failures
Technology Suite And Roadmap	Technology suite determination and roadmap, development of architecture roadmap
Precursor Swarm Mission	Including but not limited to precursor missions for both scouting and reconnaissance using swarms and a potential precursor mitigation mission using swarms
Telecommunications	With and without motherships
Launch Vehicle	Delta IV, Atlas V, RLV, etc.

13.0 Summary

This analysis has presented the MADMEN architecture as a novel and potentially valuable technique for NEO deflection. The potential solution described here considers not only the need to move a specific impactor's orbit, but also the need to have a highly reliable, robust, and scaleable architecture that is cost effective, easy to manufacture, easy to launch, and practical to intercept most incoming threats. This preliminary assessment has indicated that several tens to hundreds of MADMAN lander spacecraft, each with a mini mass driver system, can deflect a local/regionally-devastating incoming asteroid that is in an orbit generally close to the Earth. The general feasibility of the concept has been established. Conceptual level modeling has proved the basic tenants of the concept given assumptions about the on-board subsystems. Preliminary trajectory analysis indicates current launch vehicles can be utilized for this concept. A nominal cost per spacecraft similar to NASA Discovery-class missions has been established. Substantial reductions can be made in the total number of spacecraft and/or spacecraft mass if both surface operation time and deflection distance are traded-off in the analysis. Specific use was made of fictional threat scenarios to present a case study of this planetary defense architecture. Future efforts are still required to extend the simulation of this concept and add detail to various avenues of investigation.

14.0 Media and Public Relations Efforts

During the course of the Phase I effort, SEI presented the MADMEN swarm concept through a variety of public venues. These efforts helped increase awareness of the asteroid mitigation problem and potential NIAC-funded solutions to both the technical community and the public in general.

- 8/30/2003 - A presentation on the topic of planetary defense was delivered at DragonCon 2003 in Atlanta, GA. Entitled "Asteroids: Are They Coming for You and How to Stop Them," the presentation reviewed the progress of the international space community in the area of NEO detection, and explored the range of proposed asteroid impact mitigation options.
- 2/21/2004 - The Atlanta Journal-Constitution ran a front page story on the MADMEN swarm concept titled "Dodging a Space Bullet."
- 2/23/2004 - A paper on the MADMEN concept entitled "A Rapid and Scaleable Architecture Design for Planetary Defense" was presented at the first AIAA Planetary Defense Conference.
- 2/25/2004 - A short interview with Dr. John Olds was aired on Discovery Channel Canada's Daily Planet program. The MADMEN vehicle and its concept of operations were discussed, in addition to a general dialogue on the threat posed by asteroids and comets.
- 2/28/2004 - Washington D.C.'s all news WTOP radio station broadcast a segment called "Taking the Bite Out of an Asteroid" about SEI's developments in asteroid mitigation.
- 4/27/2004 – Dr. John Olds was guest speaker at a morning meeting of the Gwinnett County Kiwanis Club. A presentation on planetary defense and the MADMEN architecture was given.
- *Popular Mechanics* is currently preparing an illustration to accompany a "Tech Watch" article on the MADMEN.
- Space.Com has interviewed Matthew Graham for an upcoming article on the MADMEN project.
- A.C. Charania was interviewed by Flight International magazine and Associated Press Television while attending the AIAA Planetary Defense Conference in Orange County, CA.
- A forum discussion on SEI's MADMEN concept has generated nearly 500 postings on the popular technology-oriented Slashdot.org internet site.

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Appendix B: Organization and Personnel

DESCRIPTION OF ORGANIZATION

Founded by Dr. John R. Olds, SpaceWorks Engineering, Inc. (SEI) is a small aerospace engineering and consulting company located in metropolitan Atlanta. The current staff consists of several aerospace engineers and one financial officer. SEI specializes in providing products and services to enhance engineering design and analysis. In the services arena, SEI provides timely and unbiased analysis of advanced space concepts ranging from space launch vehicles to deep space missions. The firm's conceptual and preliminary level toolsets and methods can help determine feasibilities of space systems, viabilities in the marketplace, and determine the temporal impacts of technology on public and private actors. The firm forecasts future markets making determinations of future policy and media initiatives.

The firm's capabilities include conceptual and preliminary level modeling of a broad range of future space transportation and infrastructure concepts. Typical systems architectures might include 2nd/ 3rd / 4th generation single-stage and two-stage reusable launch vehicle designs (rocket, airbreather, combined-cycle), launch assist systems, in-space transfer vehicles and upper stages, orbital maneuvering vehicles, lunar and Mars transfer vehicles for human exploration missions, in-space transportation nodes and propellant depots, and interstellar missions. For these and other concepts, SEI can provide complete packaged analyses, from the initial vision to a final converged engineering concept, including: engineering design and analysis, independent concept assessment, life cycle analysis, and programmatic and technical analysis. SEI has experience with many industry-standard aerospace engineering design and analysis tools.

SEI performs much of its work in the area of computational modeling and simulation. Facilities include standard offices with no specific hardware laboratories. Equipment at SEI is limited to that hardware or software that aids in computational modeling. SEI has a broad range of software and hardware for various platforms and operating environments. In addition, SEI has experience with a variety of software languages, including C, C++, Java, Visual Basic, Fortran, Python, and Perl.

Recent projects at SEI include:

- NASA Langley: Next Generation Launch Vehicle (NGLT) architecture support
- NASA MSFC TD-03: Quickstrike ETO/In-Space trade tree concept studies
- Northrop Grumman: Hypersonic cruise vehicle demonstrator system (HCV-DS) design for DARPA/Air Force FALCON program
- AFRL-WPAFB: Innovative concept development for RLVs using combined-cycle propulsion for military applications
- NASA Institute for Advanced Concepts (NIAC): Phase I Award for asteroid planetary defense
- NASA MSFC: SPLV – Small payload launch vehicle (SPLV) assessment
- NASA KSC: Design for Operations (D4Ops) space transportation study
- NASA KSC: Facilities and Ground Operations Analysis (FGOA) tool development for future space transportation systems
- NASA MSFC: Air-launch to orbit (ALTO) study support
- NASA MSFC: ARTS dual fuel RLV concept with launch assist
- NASA MSFC Advanced Concepts Group: 3rd Gen RLV concept assessment and engineering tool development
- NASA 2nd Gen RLV / Space Launch Initiative (SLI) Program: Advanced Engineering Environment (AEE)
- NASA Headquarters: FY2002 RLV technology goals assessment
- NASA inter-center Value Stream Analysis Program: Micro and macro level technology implications for 3rd Gen RLVs
- NASA MSFC Integrated Technology Assessment Center (ITAC): Space transportation technology prioritization
- Revolutionary Aerospace Systems Concept (RASC) Program at NASA MSFC: Database and tool development

- NASA Institute for Advanced Concepts (NIAC): Phase I Award for Mars telecommunication networks
- SAIC and NAL (Japan): ATREX engine test program performance assessment
- Lockheed Martin Astronautics: Assessment of optimization codes for space transportation case studies
- DARPA: Responsive Access Small Cargo Affordable Launch (RASCAL) program subcontract for performance analysis
- NASA MSFC Program Planning Office: Heavy-lift launch vehicle configurations predicated on SLI technologies

PRINCIPAL INVESTIGATOR (PI): DR. JOHN R. OLDS

Dr. John R. Olds is President and CEO of SpaceWorks Engineering, Inc. Dr. Olds has over 15 years of experience working on advanced space transportation projects, having worked with General Dynamics' Space Systems Division, NASA Langley Research Center, NASA Marshall Space Flight Center, and the Mars Mission Research Center at North Carolina State University. Dr. Olds is also currently an associate professor in the School of Aerospace Engineering at Georgia Tech where he acts as Director of the Space Systems Design Laboratory. As Director of the SSDL, he has had the opportunity to advise numerous M.S. and Ph.D. students on space-oriented design topics. He is author or co-author of over 50 technical papers related to conceptual design of advanced space systems since 1986. He has conducted studies of advanced launch systems, RLV designs, Mars missions, LEO-based satellite constellations, lunar resource missions, and space solar power satellites. Dr. Olds is a registered professional engineer in the state of Georgia and an Associate Fellow of the American Institute of Aeronautics and Astronautics. He holds a Ph.D. in Aerospace Engineering from N.C. State University, an M.S. in Aeronautics and Astronautics from Stanford University, and a B.S. in Aerospace Engineering from N.C. State University.

Selected professional papers include:

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Mr. A.C. Charania is senior futurist at SpaceWorks Engineering, Inc. (SEI). His previous experience includes roles at Accenture (formerly Andersen Consulting), Futron Corporation, and the Georgia Institute of Technology’s Space Systems Design Laboratory (SSDL). At the first organization, projects involved formulating strategies to address future concepts of the “network” as applied to comprehensive strategic technology assessments of the terrestrial telecommunications marketplace; examining both markets (long distance, local access, Internet, Intranet, and E-Commerce) and technologies (ATM, AIN, ISDN, and xDSL). Projects at the latter two organizations included conceptual design and analysis (with a concentration on financial engineering and robust design) of future space concepts such as: Space Solar Power for NASA Marshall, Mars Orbit Basing (MOB) / Solar Clipper for NASA HQ, 3rd Gen and Bantam RLVs for NASA Marshall, space tourism for NASA Langley, Phobos landers, and Europa landers. In particular, his expertise includes far term technology / market forecasting utilizing analytical models and incorporation of robust design methods in the conceptual design process. He is a previous NASA Institute for Advanced Concepts (NIAC) fellow based on his role as Principal Investigator for the phase I study (CP-01-02) entitled: "Networks on the Edge of Forever: Meteor Burst (MB) Communication Networks on Mars." He holds an M.S. in Aerospace Engineering from the Georgia Institute of Technology (with a concentration in systems design and optimization), a B.S. in Aerospace Engineering from the Georgia Institute of Technology, and a B.A. in Economics/Mathematics from Emory University.

Selected professional papers include:

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Selected professional papers include:

- 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.
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Mr. Jon Wallace is a Project Engineer at SpaceWorks Engineering, Inc. (SEI) in Atlanta, Georgia. His background includes experience with conceptual design and evaluation of advanced aerospace systems. During his academic tenure, he served as team leader for the conceptual design of an in-space tether launch system intended to support Mars exploration. His experience includes trajectory optimization for future generation RLVs, CAD geometry automation and optimization, and historical evaluation of nuclear electric propulsion concepts. He holds a B.S. in Aerospace Engineering from the Georgia Institute of Technology.

Selected professional papers include:

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