SHIELD—A Comprehensive Earth Protection System

A Phase I Report to the NASA Institute for Advanced Concepts

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SHIELD–A COMPREHENSIVE EARTH PROTECTION SYSTEM

1.0 EXECUTIVE SUMMARY

The greatest natural threat to the long-term survivability of mankind is an asteroid or comet impact with the Earth. Throughout its history, the Earth has continuously been bombarded by objects ranging in size from dust particles to comets or asteroids >10 km in diameter. Although the probability of the Earth being hit by a large object is low, the effects of an impact are so catastrophic that it is essential to prepare a defense against such a threat. SHIELD is a comprehensive Earth defense system designed to discover, catalog, calculate the orbits and impact probability of near-Earth objects (NEOs) and, to deflect, disperse, or otherwise alter the orbit of a potential impactor. This study provides an architectural concept for an overall system suitable for implementation within the next 10 to 40 years.

Although much work still needs to be done, studies exist on many of the various aspects of Earth protection. A survey of the work performed to date has helped to understand the threat and refine the SHIELD design. SHIELD ties together the various aspects of this research into a comprehensive Earth protection system design, and provides selective investigations into crucial individual areas of detection and deflection of NEOs.

The SHIELD system as defined consists of three components: Sentries, Soldiers, and the Earth Control Center. Sentries are spacecraft designed to search and locate NEOs of all types, including near-Earth asteroids (NEAs), short period comets (SPCs), and long period comets (LPCs). Each Sentry spacecraft carries a visible imager specifically designed for this task. The imager takes multiple images separated in time of a location in space, registers the images, subtracts the fixed background (stars), and analyzes the remainder for objects in motion. When enough images are taken, the orbit of each object is calculated and the potential for each orbit to be Earth crossing is determined. If an object is determined to be an Earth crosser, the onboard database is queried to determine if it is a known NEO. If not, the orbit is propagated forward in time to understand the impact risk. The results are stored in the onboard database, downlinked to the Earth Control Center, and transmitted to the other Sentries.

The Sentries are the least "futuristic" component of the SHIELD system. Indeed, very capable Sentries can be launched today. The architectural characteristics of the Sentry design trade off the number, location, and search patterns of the Sentries against both the time to complete the catalog of NEAs and the warning time of LPCs. The ultimate goal of the Sentries is to provide the maximum lead time for a potential impact, which simplifies the task of the Soldier to deflect or disperse the object.

The results of this study provide the spacecraft characteristics required to support the Sentry design, imager specifications, justification for space-based vs. ground-based imaging, and a discussion of the tradeoffs of the number of sentries, their location and search patterns. A set of three or four Sentries in heliocentric orbits at the radius of Venus orbit is baselined.

The Soldier component of SHIELD consists of a series of spacecraft designed to mitigate the threat of an impact by deflecting or dispersing the potential impactor. The efficiency of the mitigation types is a function of the object size, velocity, physical properties, and warning time to impact.

Since the specific Soldier design is highly dependent on the mitigation method chosen, the detailed design of a Soldier is not attempted in this report.

Several potential mitigation methods are discussed, including chemical propulsion, kinetic energy impact, mass drivers, electric propulsion, solar sails, directed energy, and nuclear. A comparison of their capabilities by normalizing their specific impulse is presented. Also, calculations of the size of the object that can be moved an Earth radius vs. the warning time are given for each technique considered. Each has its own engineering challenges and requires some development to be feasible. The state of development, potential for threat mitigation, and engineering issues of each is discussed. The mitigation methods can be broadly categorized into two types: rendezvous in which the Soldier physically lands on the NEO (e.g., chemical and electrical propulsion, mass driver, nuclear surface detonation, etc.) and intercept (e.g., kinetic energy impact, nuclear standoff, etc.). The system design of rendezvous- and intercept-type Soldiers are discussed as are the instrumentation required for each. An estimate of the number of Soldiers required and a parking orbit location that will enhance the chances of reaching most NEOs is discussed. A multi-tiered space and Earth-based Soldier complement with four Soldiers placed in high-energy Venus-return orbits and two additional Soldiers on Earth in a standby, launch-on-demand mode are baselined.

Soldiers using intercept-type methods require a separate remote-sensing spacecraft for information regarding the physical characteristics of the body, its composition, center of rotation, etc. The system requirements of a scout concept and instrumentation requirements are discussed.

The Earth-based component of SHIELD must receive the data from the Sentries and Soldiers, verify the calculations of impact potential, maintain the NEO database, communicate with the ground-based telescope surveys for prior and/or follow-up observations, recommend to the global authority that a Soldier asset be committed to mitigate a potential threat and direct the Soldier activities. Portions of the Earth-based SHIELD component may be integrated with existing facilities such as the Minor Planet Center. This component of SHIELD is not addressed in this report but will be addressed in detail in Phase II.

The Phase I study has clearly shown that a SHIELD Earth-protection system is practical and that a full system could be built within a few years. Some important features of SHIELD and a full evaluation of its performance must await the completion of a Phase II study.

2.0 BRIEF SUMMARY OF HAZARD DUE TO COSMIC IMPACT

The potential for globally catastrophic effects from larger diameter NEO impacts (≥1 km) has only recently been recognized. *Alvarez et al.* [1980], linked the K/T extinction of the dinosaurs 65 million years ago to a cosmic impact. The Comet Shoemaker–Levy 9 impact of Jupiter in 1995 has also focused attention on this issue. In the last two decades, much work has been and continues to be done on many of the individual aspects of the impact threat and its mitigation including but not limited to:

- Expected impact frequencies vs. size and energy of the impactor
- Number of asteroids and comets vs. size in near-Earth orbits
- Effects on the global ecosystem of impacts of various sizes
- Effect of a deep water (tsunami) vs. land impact
- Expected number of deaths per event
- Relationship of risk to size of impactor (annual probability of death from a cosmic impact)

- Physical properties of NEOs as related to the effect of impact and the ability to modify their orbits
- Various nuclear and non-nuclear methods to deflect, break up, or otherwise modify the orbit of asteroids or comets enough to miss the Earth

The first step in the SHIELD study was to survey the work that has been performed to date in an attempt to understand the threat and its effect on the system design. The results of this survey indicated that the vastly different orbital and physical parameters of NEAs, SPCs (period <20 yrs), and LPCs (period >20 yrs) would have a significant effect on the SHIELD. NEAs and LPCs in particular represent a bi-modal problem for SHIELD. Although the technology can be developed to locate and mitigate a threat from NEAs particularly given the long warning times available after the catalog is complete, LPCs—with their inherent short warning times, higher orbital velocities, and random (often retrograde) inclinations—will be much more difficult to locate and defend.

Since NEAs are thought to represent 90 to 95% of the Earth impactors, their effect on the SHIELD design is addressed first.

Near-Earth Asteroids

Several studies exist quantifying the number of deaths, environmental and economic effects, etc., of a large diameter NEA impact. A curve of expected fatalities per event vs. diameter and energy of a NEA is given in Fig. 1 [*Morrison et al.*, 1994]. At an object diameter threshold of 0.5 to 2 km, called the global catastrophe threshold, the expected number of fatalities increases by several orders of magnitude and exceeds 25% of the world's population. Various sources compute the object diameter necessary to trigger a global catastrophe to be from 0.5 to 2 km with 1 km generally considered the reference threshold. The actual global catastrophe threshold varies as a

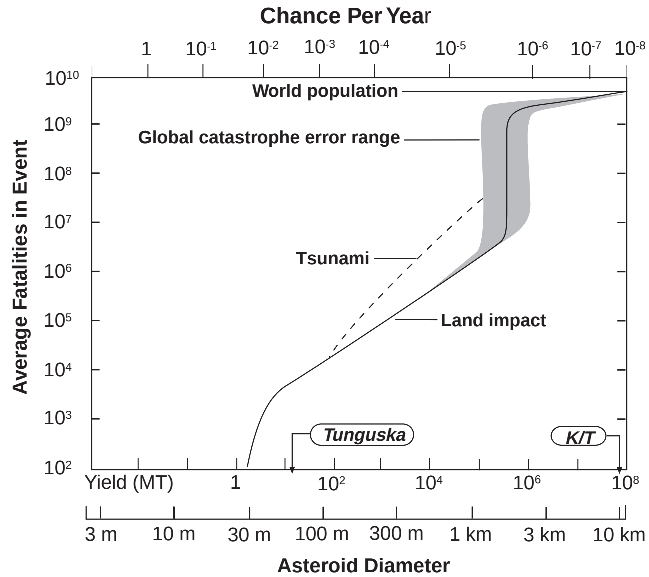

Figure 1. Average mortality as a function of energy, asteroid diameter, and probability of occurance for the current population of the Earth.

function of relative object velocity and mass, impact angle, location of impact, etc. For this curve, a relative velocity and density of 20 km / s and 3 gm / cm^3 are assumed.

Energy is given in megatons (MT) of TNT, where 1 MT = 4.2×10^{15} joules. An object travelling at a relative velocity of 3 km / s has the same energy as the equivalent mass of TNT. The average asteroid and cometary velocities are 21 km / s and 55 km / s (Hughes), which represent KE multipliers of 49× and 336× the equivalent mass of TNT, respectively. To put things in perspective, the K / T extinction impact caused by an object 10 to 15 km in diameter had a kinetic energy (KE) of 10^8 MT, a factor of 5×10^9 more powerful than the bomb dropped on Hiroshima (20 kilotons). The 1-km-dia. global catastrophe causing object has a KE of 10^5 MT.

The probability of a large diameter NEO impact with Earth, as shown in Fig. 1, is small but not insignificant. For example, Figure 1 shows a 2×10^{-6} probability of a globally catastrophic impact (>1 km object) occurring in a given year or a 1×10^{-4} chance in the average lifetime. As a result of the low probability of occurrence, the relatively recent realization of the issue, and the lack of a relatively significant recent impact, NEO impacts have not yet come to the forefront of public policy. The lack of public concern for the issue has been accompanied by a lack of funding to locate objects and study the mitigation techniques.

As a result of this and other similar studies, the prime focus of SHIELD Sentries (and of terrestrial-based telescopes) is to locate, catalog, and determine the impact probability of NEAs ≥1 km in diameter.

Figure 2 [*Morrison et al.*, 1992] gives the estimated number of Earth-crossing asteroids larger than a given diameter. This curve and other similar curves are based on analyses of ground-based telescopes' search patterns (biases) and statistics, the lunar cratering record, etc. Figure 2 shows that ~2100 NEAs larger than 1 km in diameter are thought to exist in near-Earth orbits of which ~10% have been located to date.

Recent advances in ground-based telescope technology (primarily large area CCDs and automated software) have significantly increased the rate at which NEAs are found. However, it will still be a century before the catalog of >1-km-dia. asteroids are 90% filled at the current rate of discovery [David Morrison, Asteroid and Comet Impact Hazards Web Site, NASA Ames Space Science Division]. Once an asteroid is located, and its orbit parameters defined, its orbit can be propagated out for up to two centuries [*Yeomans et al.*, 1994] providing decades of advanced warning of a NEO impact. The longer warning time also results in a lower velocity change, ΔV, and total impulse requirement to deflect the object as shown in Figs. 3 [*Gurley et al.*, 1994] and 4, respectively. A critical goal of the SHIELD system, therefore, is to locate and catalog all

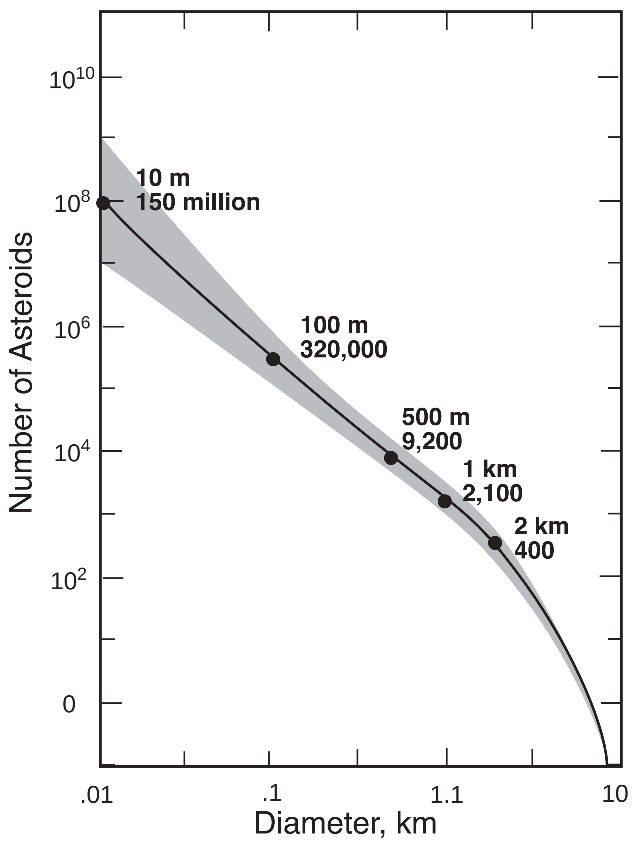

Figure 2. Size distribution of Earth crossing asteroids.

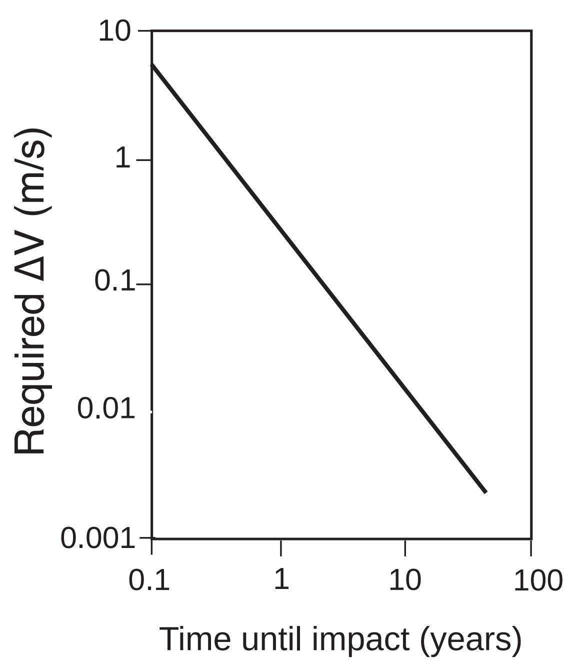

Figure 3. Required deflection velocity as a function of time before impact for a typical asteroid.

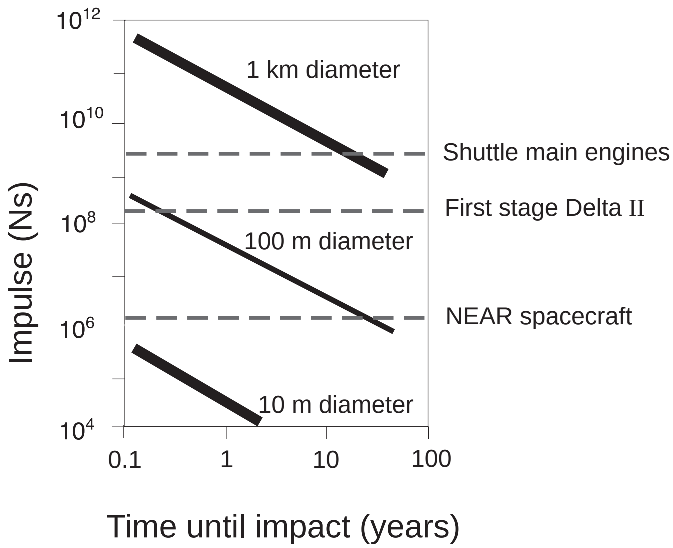

Figure 4. Required impulse as a function of time before impact. Shown are capabilities of the Space Shutle's main engines, the first stage of the Delta II, and the NEAR spacecraft.

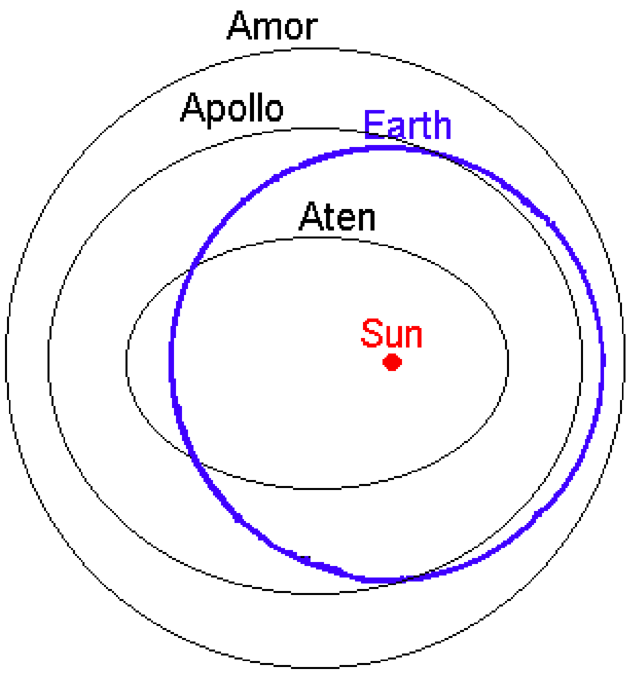

Figure 5. Orbits of typical Aten, Apollo, and Amor type asteroids relative to the Earth

NEAs ≥ 1 km in diameter as soon as possible to provide this warning.

As shown in Fig. 2, the number of NEOs varies with size roughly as a power law with an exponent of –2. Thus, there are >9,200 and 320,000 NEAs larger than 500 and 100 m in diameter, respectively. Although there are no recent examples of an impact in the 500 m to 1-km-dia. range, studies have shown that impacts by objects in this size range can devastate whole states and small countries [*Morrison et al.*, 1992]. An actual example of an impact of a ~50-m-dia. asteroid is one that exploded over Tunguska, Siberia in 1908 devastating an area over 40 km in diameter (larger than the Washington, D.C. beltway). An event of this size is expected to occur every 300 years although the probability of it occurring over a populated area is much smaller. As a result, SHIELD will extend the NEA catalog down to smaller diameter NEAs after the ≥1-km-dia. catalog is complete.

Three classes of NEAs are defined: Atens, Apollos, and Amors, the orbits of which are shown in Fig. 5. Aten-type NEAs have a semi-major axis, $a < 1$ and an aphelion, $Q > 0.983$ AU (Earth perihelion). Apollos-type NEAs have $a > 1$ and perihelion, $p < 1.017$ AU (Earth's aphelion). Amors, with $a > 1$ and a perihelion of $1.017 < p < 1.3$ AU, have the potential to be perturbed into an Earth-crossing orbit. Due to the very dim nature of the targets, all telescope NEO search patterns are centered at solar opposition. Aten-type NEAs in particular are very difficult to locate from Earth as they spend very little time outside of Earth's orbit. As a result few of these have been found, and the total contribution of Aten-type asteroids to the NEA population is not well understood. The need to locate Aten-type NEAs will affect the choice of location of the SHIELD Sentries.

Figure 6 (JPL Solar Systems Dynamics Web Site, http://ssd.jpl.nasa.gov/a_distrib.html) represents the number of main belt

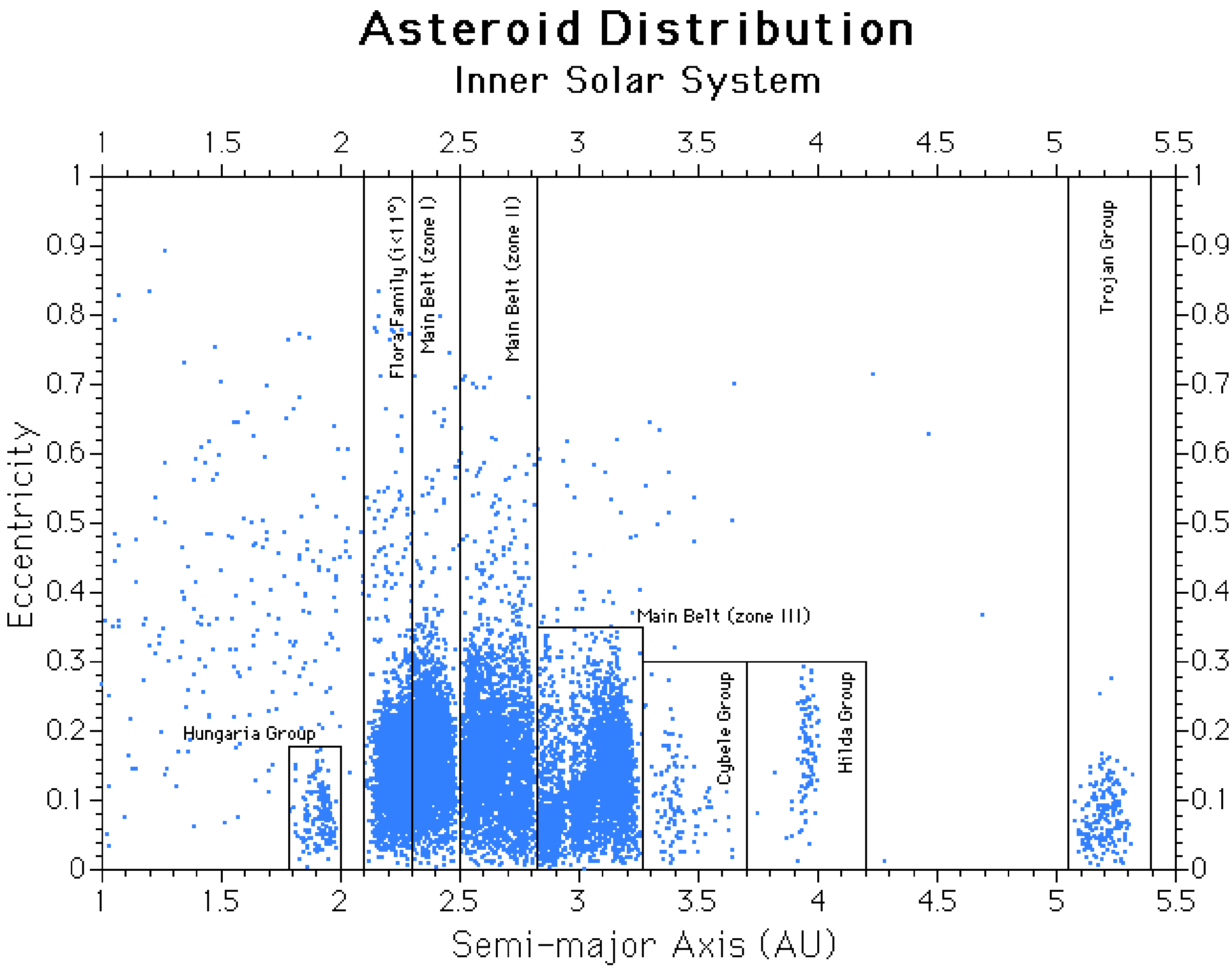

Figure 6. Number and families of known main belt asteroids vs heliocentric distance

asteroids with known orbits and their distance from the Sun in AU. An estimated 40,000 currently have approximately known orbits, most of them lying in the main asteroid belt between Mars and Jupiter. Most of them have low-inclination, low- eccentricity orbits with semi-major axes that range from 2.1 to 3.3 AU. Although most of the large main belt asteroids have been found (~682 objects with d >50 km), less than 10% of the ~10^5 objects with d > 10 km and an even smaller proportion of the ~10^8 objects with d > 1 km have been found (Bailey and Napier, 1999). Compare this to the estimated 2100 NEAs with d > 1 km and it is obvious that if the Sentry imager has enough sensitivity to reach the main asteroid belt, that an overwhelmingly large fraction of the asteroids detected will be main belt asteroids that are not Earth crossers. For example, the LINEAR system as of March 2, 1999 had detected 170,841 asteroids of which 16,613 were new discoveries (~10%) and 171 (~0.1%) were NEOs (Linear Project Home Page). Sentries must be designed to handle the number of main belt asteroids (computational load, size of database, downlink capability) or alter search strategy to filter out slower motion of main belt asteroids.

The typical orbital periods of NEAs are 2 to 5 years, allowing them to be detected on more than one apparition and providing more accurate orbit determination, propagation, and impact probabilities. The prograde orbits and low inclinations bound or at a minimum allow for an optimization of the search area and strategy. The lower mean velocities (~21 km / s), generally low inclination, prograde orbits, and potentially long warning times, allow more time to reach the object and flexibility in the choice of mitigation techniques.

Earth-Crossing Comets

Earth-crossing comets are divided into two classes: SPCs (<20 yrs) and LPCs (>20 yrs). SPCs, which represent 10 to 20 % of all comets [*Bailey and Napier*, 1999], are similar to NEAs in terms of average velocity, inclinations relative to the ecliptic, and orbital periods. For any given size, SPCs contribute an additional 1% to the NEA population [*Morrison et al.*, 1992].

LPCs are thought to contribute ~5 to 10% of the flux of impactors in the >1-km-dia. range [*Morrison et al.*, 1992]; however, because of their greater impact speeds, LPCs account for 25 to 50% of the craters >20 km in diameter. As shown in Table 1, LPCs are significantly different from NEAs in terms of their physical and orbital parameters. In fact, NEAs and LPCs represent a bi-modal problem for the SHIELD system. For example, the periodicity of the NEA, SPC, and the Halley family of LPC orbits allow for orbit knowledge, propagation, and predictions of future impacts leading to impact warning times on the order of decades. NEAs orbits can be relatively accurately propagated forward for up to two centuries [*Yeomans et al.*, 1992]. Since no *a priori* knowledge of Oort Cloud LPC orbits exists, the maximum impact warning time is from the time that it is first discovered. LPCs generally become active between 5 and 10 AU but

Table 1. Orbital and Physical Parameters of Near Earth Objects.

Parameter	NEAs	SPCs Jupiter Family	LPCs Halley's Family	LPCs Oort Cloud Comets
Orbital period	2 to 5 yrs	<20 yrs	20 < T_p 200 yrs mean 5.05 yrs	>200 yrs
Mean velocity relative to Earth	21 km / s	19.9 to 26 km / s	52.3 km / s	55 km / s
Inclination relative to the ecliptic	Most <40°	Uniformly populated between ± 31°	Random between 0 to 180°	Random between 0 to 180°
Density (gm / cm^3)	3.65	0.2	0.2	0.2
Impact warning time (T_w)	$0<T_w<200$ yrs Decades *after*	$0<T_w<200$ yrs Decades *after catalog complete*	Decades with *a priori* orbit knowledge	Up to 2 yrs
Fraction of to NEO population	~90–95%	~1%	~5 to 10%	

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are seldom observed at solar distances >5 AU [Morrison et al., 1992]. LPCs take approximately 16 months to travel from the orbit of Saturn (9.5 AU) to that of Jupiter (5.2 AU) and a little more than an additional year to travel to perihelion from Jupiter. As a result, the warning time for an impact from a LPC will typically be on the order of months and almost certainly, no more than 2 years. NEAs have low inclinations and prograde orbits. LPCs have a random inclination distribution and often have retrograde orbits. This random inclination distribution requires that Sentry search areas cover the entire sky. The retrograde orbits also significantly affect the search patterns used which, for Earth-based telescopes, are optimized for prograde motion.

The higher relative velocities, coupled with the high inclination/retrograde orbits and the short impact warning time, combine to make LPCs difficult for Soldiers to reach. In fact, given a two-year impact warning and a Soldier ready to launch, it is not clear that, with today's technology, a Soldier could reach the LPC in time to mitigate the threat. For this reason, Soldiers must be placed in holding orbits to optimize the chances of reaching a LPC. Assuming that the Soldier can reach the object in time, the choice of mitigation techniques is limited. Rendezvous-type mitigation techniques usually require too much ΔV or too much time. Intercept-type techniques are usually required. The differences in average density of LPCs relative to NEAs have the potential to alter the effectiveness of intercept type mitigation techniques and must be accounted for in the design.

In summary, the greater kinetic energy, differences in the physical characteristics, shorter warning time to impact, and high inclinations and/or retrograde orbits relative to the ecliptic, combine to make LPCs much more difficult to defend against relative to NEAs. Many deflection/dispersal techniques that have the potential to be effective with NEAs (which will typically have decades of warning time when the catalog is complete), will not work with LPCs.

Several important SHIELD-related conclusions have been reached from researching the current literature.

- It is extremely important to complete the survey of >1-km-dia. NEAs as soon as possible to obtain the decades of warning time that the complete catalog affords.
- Once all of the 1-km-dia. NEAs have been located, the catalog should be extended to smaller diameter NEAs, which have the capability of destroying entire states and small countries.
- The location of Sentries must take into account Aten-type NEAs, which cannot easily be detected from heliocentric orbits ≥1 AU.
- The Sentry design must be capable of handling the number of main belt asteroids or some method must be used to filter them out of the location process.
- The out of plane orbital parameters, high-relative velocities, and non-repetitive nature of the LPC orbits, result in short warning times and long intercept travel times. As a result, Soldiers will have to be placed in parking orbits to maximize the chance of intercept.
- Due to the high-relative velocities of LPCs, rendezvous-type mitigation techniques require too much ΔV. Intercept-type mitigation techniques are required.
- Sentries must have expanded coverage area (deg^2/month) and modified search patterns relative to existing ground-based telescopes and even relative to the proposed Spaceguard survey.
- The optimal number and location for both Sentries and Soldiers may be different for NEAs and LPCs.

3.0 SENTRY

The Sentry component of the SHIELD system will consist of multiple spacecraft each with a visible imager specifically designed to detect NEOs. Each Sentry will have the processing power to register, background subtract, and difference several images of the same location spaced in time. When enough detections of a specific NEO are collected, the orbit of the object is calculated, and it is compared to the database to determine if it is a previously located object. If it is a new object, the potential for the object being an Earth crosser is evaluated, and, if the potential is high enough, the orbit is propagated forward in time onboard to provide a detailed determination of the chance of an impact. Finally, the object, its orbital parameters, and the impact risk are stored in the onboard database, and forwarded to the Earth Control Center and to the other Sentries. If the object is thought to have a probability of impact above a predetermined threshold, entire images containing the object will be forwarded to the Earth Control Center. This operation is fully autonomous.

Sentry Imager Design

The imager consists of a large aperture telescope with a large format, cooled CCD detector. The telescope parameters will be similar to existing ground-based telescopes. The telescope will require a 1-m-diameter aperture and will have a $1.6 \times 1.6°$ FOV. If launched today the CCD would be 4096×4096, 15 mm pixels with four quadrant readout. The angular resolution of 1.4×1.4 arcsec would be sufficient considering the spacecraft pointing errors and jitter. To minimize dark current and to take into account the radiation effects on charge transfer efficiency, the CCD will be passively cooled to <–70°C, and would have to be shielded to keep the total radiation dose below 10 krad. The sampling scheme would have to be capable of discriminating against cosmic ray hits to the CCD.

The CCD selected would be a frame transfer device to eliminate the need for a mechanical shutter and provide more efficient use of search time. To maximize sensitivity, the CCD would be a high QE, back-thinned device with low-read noise. The imager sensitivity would be $V_m = 22$ with <100 s of integration time. The specific parameters of the imager will be optimized for the orbit and search pattern selected.

Sentry Spacecraft Design

The spacecraft subsystem designs are straightforward with attitude control and determination having the most stringent requirements. Attitude requirements include 3-axis control with arcsec pointing and knowledge over the maximum integration time of the CCD (~100 s), and subarcsec stability and jitter. Since the imager will nominally be pointed in solar opposition, the solar arrays are always in full sun providing a favorable power situation. One side of the spacecraft is therefore always looking away from the Sun, providing an ideal surface for passive cooling of the CCD. For Sentries launched in the near term, telemetry will be downlinked once every day to two weeks to minimize the disruption of the scan sequence, allow for the high-gain antenna to be pointed toward Earth, and to minimize ground station costs. For Sentries launched further out in time, telemetry will be downlinked by Sentry request when a predetermined number of NEOs or downlink data size has been reached, a potential Earth impactor has been located or if certain autonomy rules have fired.

Communication between Sentries is required for the autonomous operation of the SHIELD system and to provide a method for Sentries on the opposite side of the Sun to communicate with Earth. The Sentries must be able to transmit and receive the object orbital parameters, the object catalog, and the attitude and orbit data of the other SHIELD spacecraft. If an NEO is located that has a significant impact probability, each of the relevant images will be transmitted to the Earth Control Center for further analysis and impact assessment. The transmission and reception of data between SHIELD spacecraft will be accomplished via an optical communication system. The European Space Agency is currently developing a standard optical crosslink design.

The orbital accuracies required for the Sentries depend on the minimum distance to the target asteroid and the angular accuracy with which asteroids and stars can be measured. The objects of most interest are those that cross the Earth's orbit, which is about 0.2 AU from the aphelion of the Sentry orbit, assuming the Sentries are in heliocentric orbits near the orbit of Venus. The 1.4-arcsec-pixel angular resolution results in a spatial resolution of 200 km at 0.2 AU; therefore, the orbital position at any given time needs to be determined to that accuracy or better. That is well within the current capabilities of interplanetary spacecraft orbits determined from Doppler and range measurements, with the accuracy of the ephemerides generated from the determined orbital elements being less than 50 km.

The timing accuracy is determined by the positional accuracy discussed earlier divided by the maximum spacecraft heliocentric velocity of not more than 40 km/s, so it must be better than about 5 s, compared with currently possible spacecraft timing accuracies of less than 0.01 s.

If a Sentry were to be launched today, the only requirements that may not be easily met with current technology are the onboard computational capability and the Sentry-to-Sentry optical communications. Computations of the type required have been performed on spacecraft. However, they may not be easy to implement on a low-cost spacecraft. Most of the computations described previously could be performed on the ground at the expense of higher mission operations costs. The image registration, background subtraction, differencing, and NEO detection must still be performed onboard to minimize the downlink data rate. Several organizations have built optical communications hardware for spacecraft-to-spacecraft communications. However, most current designs are not optimized for the very long distances between Sentries. Sentry-to-Sentry communications could be accomplished using standard RF techniques with limited data rates and higher input power.

Detection techniques, computational and storage requirements

Asteroids and comets are distinguished from the background star field by their motion over multiple images. Therefore, images must be taken and stored of the same sky arc at different times. The basic steps in processing an image include A/D conversion, image registration, background subtraction (stars) and normalization, cosmic ray filtering, and velocity matched filtering. The minimum angular motion that can be seen is determined by the time between subsequent images; therefore, the search pattern selection inherently provides motion filtering.

After the image manipulation is complete and moving objects have been identified, each object's orbit is determined and the orbits of new objects are cataloged. The differentiation of new versus known objects will be facilitated using an autonomously maintained object catalog. This catalog will contain the objects orbit and error estimates associated with each asteroid and comet. Each time an object is identified and an orbit determination is made, the catalog will be searched and if that object is determined not be in the catalog, a new entry will be made. The object catalog might eventually contain approximately 10^8 identified objects that may require storage for 20 double precision numbers per identified object, representing the orbital parameters and error estimates. The object catalog will be updated and simultaneously kept by all Sentries and transmitted to the ground periodically. Autonomous spacecraft-to-spacecraft communication will facilitate the catalog update and maintenance. Repeated observation and orbit calculation by multiple SHIELD Sentries will continue to improve the orbit determination and error estimates.

In the determination of an asteroid orbit, a series of steps will be followed by each Sentry [*Yeomans et al.*, 1994]. First the trajectory of the object is to be integrated forward, taking care of Earth and moon perturbations separately. General relativistic equations of motion and perturbations by all planets at each integration step are also included. For short and long period comets whose motions are affected by the rocket-like ice vaporization, a non-gravitational force model is used. When a close approach to the Earth is sensed by the numerical integration software, an interpolation procedure is used to determine the time of the object's closet approach and the minimum separation distance at that time. For those objects making an approach to Earth within specified distances, a full perturbation analysis is conducted to determine whether or not the object's error ellipsoid at the time of closet approach includes the Earth's position. This screening process can be achieved in a straightforward fashion using efficient and proven software with no intervention from the ground. Upon determination that an impact cannot be ruled out the stored images, orbit determination, perturbation analysis results, and catalog information will be sent to the ground. This method for computing impact probabilities is only an approximation; however, it will significantly reduce mission operation costs. A more precise computation of impact probabilities can be obtained from a Monte Carlo approach, which requires a great deal more computation and is best preformed at the Earth Control Center.

The image and object database memory requirements, assuming that the search patterns are repeated at a cadence of less than once per day and a catalog size of 10^8 objects, are 29 GB and 4 GB, respectively (see list). The calculations of orbit determination, projection, and impact probability can be accomplished with the use of an equivalent modern day desktop processor with approximately 32 Mb of RAM.

Storage requirements for *images*:

- 24-hour storage capacity
- Image ~ every 100 s
- 4096×4096 pixels
- 2 bytes / pixel
- 33.56×10^6 bytes / image * 864 images / day = 29.0 GB

Storage requirements for *object catalog*:

- 10^8 objects
- 20 double precision numbers
- 40 bytes / object * 10^8 objects = 4.0 GB

Tradeoffs of Space-Based vs. Ground-Based NEO Detection Systems

There are several advantages and disadvantages of space-based imagers relative to their Earth-based counterparts. The advantages include additional observing time, ability to optimize location, and greater sensitivity. Additional observing time and increased sensitivity allows for greater sky areal and volume coverage per month, thus increasing both the number of NEOs discovered and providing more complete discovery of smaller diameter NEOs. The ability to optimize their location can, for example, allow the discovery of Aten-type asteroids and provide up to nine months of additional warning time for LPCs. The disadvantages include cost and the inability to downlink the raw image for future reference due to data rate constraints. A short discussion of each of these is given.

Several excellent ground-based imaging systems used to detect NEOs are in operation today. The observation time per month of each of these is limited by several factors, listed below, due to being an Earth-based platform.

- Ground-based imagers can only operate at night roughly between 3 hours after astronomical twilight to 3 hours before astronomical dawn [*Pravdo et al.*, 1999]. Therefore observing time is limited and biased higher in the winter relative to the summer.
- The moon must be below the horizon; therefore, 10 to 12 nights per month centered on the full moon are not available.
- Poor weather conditions.
- Limited available usage due to borrowing time on non-dedicated telescopes. Two of the three most productive ground-based imagers borrow time on Air Force-owned telescopes. The NEAT program time allocation, for example, was reduced to six nights per month starting in January 1997 [*Pravdo et al.*, 1999].

Without the reduced time allocations, there is approximately 100 hours of available "dark time" per month [*Harris*, 1998]. Space-based imagers can operate 24 hours per day, seven days a week except for interruptions for downlinking of data giving a factor of six to seven more observing time per month than the Earth-based imagers. This additional observing time allows for full hemispherical sky coverage, the ability to perform follow-up observations on new discoveries, longer integration times, and thus more sensitivity.

The most important effect of the reduced observing time on Earth-based imagers is the reduction in sky coverage. Additionally, the latitude of the observatory together with the telescope design can also limit the available declinations [*Pravdo et al.*, 1994]. All-Earth based coverage zones are centered on solar opposition and because of the coverage limitations, are optimized for NEAs at the expense of LPC detection. The suggested Spaceguard survey coverage area, for example, is ±30° in longitude and ±60° in latitude centered on solar opposition. Given this search pattern, LPCs in certain orbits will not be detected at all and many will have warning times only on the order of weeks to months [*Marsden and Steel*, 1994]. Adding telescopes for Earth-based observations does not help this problem much. Adding Sentries equally spaced throughout the orbit, allows full sky coverage over a period of 1 to 6 months (depending on the number of Sentries, their orbital period, the search pattern chosen, and the search rate in deg^2/ month) as opposed to the year that it takes Earth-based detection systems to travel the full orbit. Having three Sentries located in a Venus-type orbit (225 day period), for example, would provide Sentry location repetition every 75 days. Assuming that each Sentry could search its solar opposition centered hemisphere during that time, the vulnerable "back side" of Earth would be covered at least every 75 days or sooner depending on the hemispherical search time. Increasing the number of Sentries or their telescope apertures can help to reduce the maximum revisit time for any area of the sky.

The ability to select the Sentry location allows the system to optimize the detection of NEOs. Using the previous example, a Sentry located at the same heliocentric distance as Venus would easily detect Aten-type asteroids. The Sentries would still be capable of detecting Apollo-type NEAs as well. Figure 7 gives the detection ellipse for an imager with V_m = 22 sensitivity assuming 1-km-dia. NEAs from a heliocentric Venus-like orbit. Sentries at Venus orbit can still detect 1-km NEAs in the main asteroid belt.

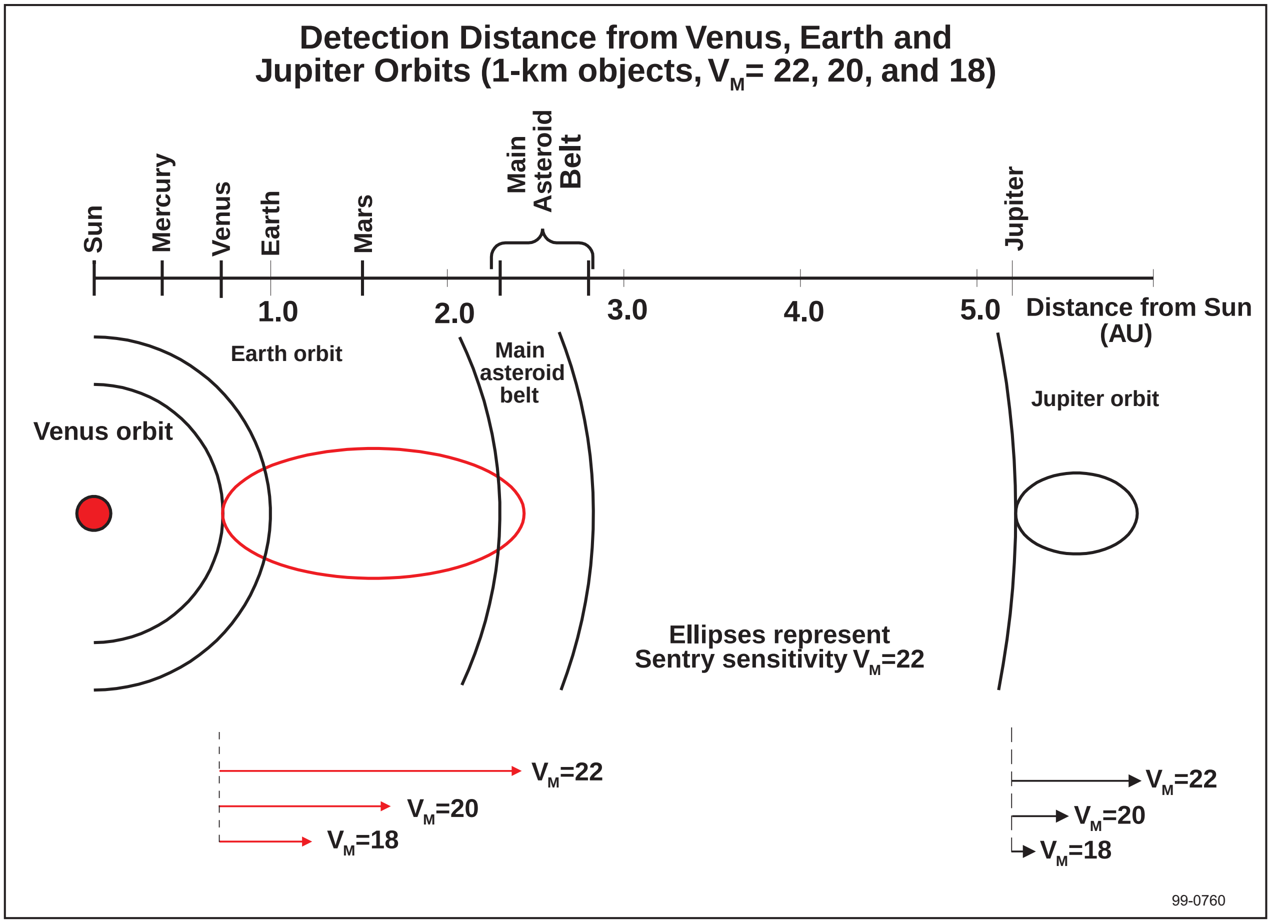

Figure 7. Detection distance from Venus, Earth, and Jupiter orbits assuming 1 km diameter object, for V_m = 18, 20, and 22.

Having space-based imagers, with the greater observing time and optimal locations, allows the Sentry to observe NEAs over a larger fraction of their orbit and gives greater flexibility in determining the search strategy. For example, the search pattern can be defined to only locate asteroids with an angular motion relative to the Sentry greater than a specific threshold. This flexibility can be used to filter most of the main belt asteroids, which, because of their distance from the Sentries, move at a relatively slow rate.

Another advantage in a space-based Sentry system is greater sensitivity. Space-based systems do not have the problem of "seeing" or atmospheric extinction or the general background light encountered on the Earth. "Seeing" is caused by atmospheric disturbances that tend to spread the asteroid image over several pixels, thus reducing the amount of light per pixel. Atmospheric extinction reduced the sensitivity by 0.1 to 0.2 visual magnitudes while the background light level reduces the S/N of the system and thus the sensitivity. The most sensitive ground-based telescopes normally operate at ~V_m = 20 but are capable of operating at V_m = 22 (tradeoff of sensitivity vs. coverage).

The improved limiting magnitude, full sky coverage, optimal orbits, and increased observing time of the space-based Sentry system result in much more efficient detection and cataloging of both the globally catastrophic and the smaller diameter NEAs, which will provide the decades of warning time of an impending impact much sooner. Proper selection of the orbit allows for detection of Aten-types NEAs as well. The space-based Sentry architecture also plugs the holes in the current Earth-based LPC search strategy (which is clearly optimized for NEA detection) and allows for observations of LPCs on the opposite side of the Sun, thus providing up to nine months of additional impact warning time. Given the difficulty in reaching LPCs discussed earlier, this additional warning time may be the difference in making an uncorrectable disaster into a potentially correctable one.

Suggested Number and Location of Sentries

Several conclusions can be obtained from previous discussions regarding the number and location of Sentries:

- Sentries must have a heliocentric distance of <1 AU to efficiently locate Aten-type asteroids.
- Sentries with a sensitivity of V_m = 22 located at the heliocentric distance of Venus can still image 1-km-dia. NEAs located in the main asteroid belt.
- Multiple Sentries will result in completing the catalog of 1-km NEAs sooner and extending the catalog down to the 0.1- to 1.0-km-diameter range.
- Multiple Sentries will enable detection of LPCs on the Earth's blind side providing up to nine months additional impact warning time.
- Multiple Sentries will provide full spherical sky coverage every 30 to 75 days.
- Even with several Sentries in a <1 AU orbit, the warning time for LPCs is extremely short.

On the basis of these conclusions, the baseline is a set of 3 Sentries at the heliocentric distance of Venus (0.72 AU). A typical Sentry orbit, including the transfer from Earth, is shown in Fig. 8. The Sentry fields of regard, assuming a set of three Sentries, are shown in Fig. 9. The actual time that a Sentry takes to revisit a particular region in its

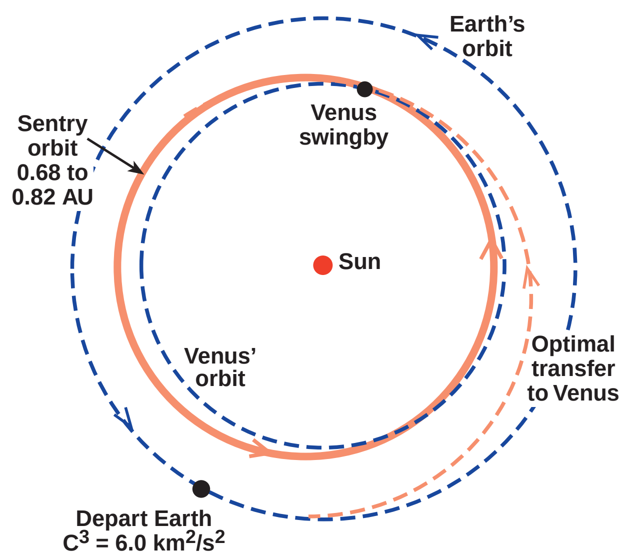

Figure 8. Typical Sentry orbit at a similar heliocentric distance as Venus including transfer trajectory from Earth.

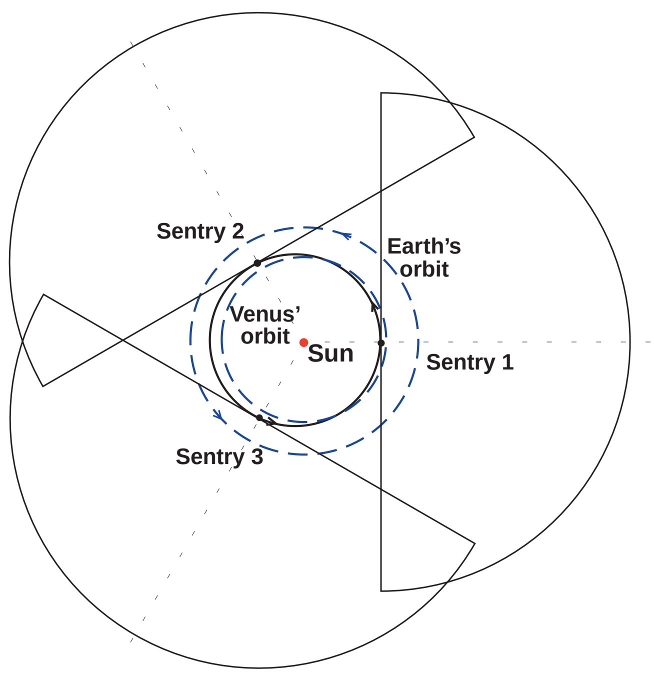

Figure 9. Sentry coverage with three Sentry spacecraft near the heliocentric distance of Venus observing 90° from the opposition direction.

coverage area depends upon the required exposure time and the search pattern. The Sentry fields of regard will rotate with the orbital motion of the Sentries.

A calculation of the time to complete the catalog of 1.0 and 0.5-km-dia. NEAs based on this Sentry complement and a plausible extrapolation of the distribution of NEA orbital elements will be refined in the Phase II study with Monte Carlo simulations. A similar analysis of warning times for LPCs will be performed in Phase II.

Sentry orbits closer to Jupiter would allow for earlier detection of LPCs. After nearly 100% of the Earth-crossing Aten (and other) asteroids have been cataloged, a more distant Sentry system might be developed to provide additional warning time for the continuing threat of LPCs. More of the distant Sentries would be needed to monitor all directions, as indicated in Fig. 10. An architecture study for an LPC warning system will be developed in the Phase 2 proposal. Figures 9 and 10 neglect phase angle effects and the probable brightening of LPCs as volatiles start to sublime as they approach the Sun. The detection "pies" would be larger for LPCs larger than 1 km, or with albedoes greater than 0.05.

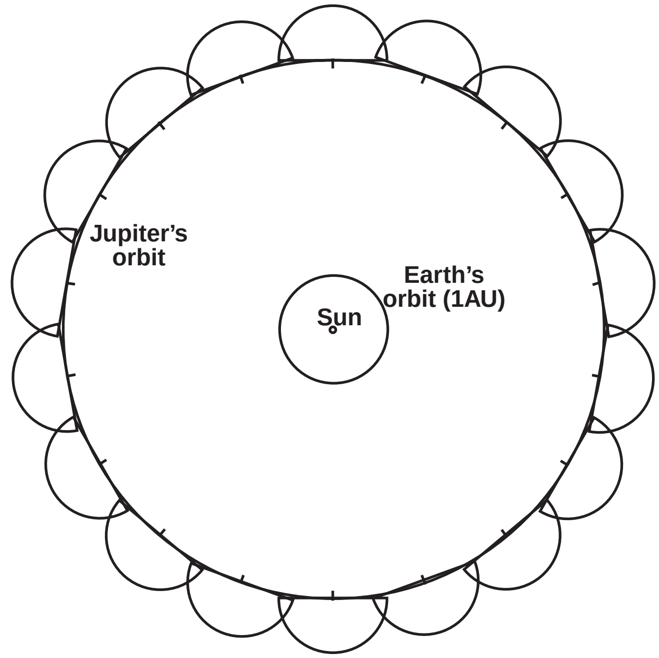

Figure 10. Long period comet distant early warning system. Ecliptic-plane coverage for long period comets 1km in diameter can be provided by 18 Sentry spacecraft observing 90° from the opposition direction from Jupiter's orbit. The Sentries could be placed there with Jupiter swingbys. Many out-of-plane Sentries would also be needed to give full-sky coverage.

4.0 HAZARD MITIGATION

Several technology options exists that ultimately could be used to defend the Earth against the threat of an impact from an asteroid or rogue comet. Each requires some engineering development to be viable. A qualitative analysis of the mitigation technologies is presented to compare the theoretical performance of each, both in terms of normalized I_{SP} and time to divert an object the radius of the Earth. The current state of readiness, type of mission parameters, comparative effectiveness, and the advantages and disadvantages of each option are also presented. The mitigation options that will be analyzed include: kinetic impact, chemical propulsion, electric propulsion, solar sails, directed energy, mass drivers, nuclear detonation, and nuclear propulsion. This section also includes a discussion of the engineering development that needs to occur to make the more credible, near-term diversion technologies work.

Kinetic Energy

Kinetic impact delivery systems are the simplest of the options to be discussed. To provide an impulse to an incoming threat, an interceptor is accelerated and rammed into the threat to provide a characteristic change in the velocity of the object, ΔV. For best results, the change in velocity must be applied in the orbit direction of the threat preferably at its perihelion [*Ahrens and Harris*, 1994]. The magnitude of the ΔV depends primarily on the production and integrated velocity of excavated debris or ejecta produced by the impactor. The mass and velocity of the ejecta provides an impulse, which in addition to the momentum of the incoming projectile, must balance the change in momentum felt by the threatening asteroid or comet [*Ahrens and Harris*, 1994]. Kinetic impact delivery are worth considering because on Earth, cratering efficiency due to impacts (ratio of the mass displaced relative to the projectile or explosive mass) may be 4 orders of magnitude greater than that due to chemical explosives [*Ahrens and Harris*, 1994]. As shown below, this high efficiency results in a far greater specific impulse $I_{SP_{KE}}$ relative to currently available deflection technologies other than nuclear.

The production of ejecta on asteroids depends primarily on two cratering regimes. For small craters, the strength of the target controls both its diameter and the momentum of its ejecta. On asteroids, this strength regime extends up to projectile diameters ranging from 6 [*Nolan et al.*, 1996] to 10 m [*Greenberg*, 1996] given projectile densities ~3000 kg / m^3 and impact velocities ~5 km / s. For impact velocities ranging from ~40 km / s to 5 km / s, the limit of the strength regime is defined by projectile masses ranging from ~5×10^5 kg to 2×10^6 kg based on cratering efficiencies rule [*Holsapple and Schmidt*, 1982; *Housen et al.*, 1983; *Holsapple*, 1993]. Most achievable intercept impact velocities will be on the order of 10 to 12 km / s. Larger craters are controlled by the gravity of the asteroid, which determines the escape velocity of the asteroid and, hence, the magnitude of the impulse generated by ejecta.

Ahrens and Harris [1994] define the ΔV imparted by a kinetic energy impact event velocity for an impact event as

$$\Delta V = \frac{M_i V_i + p}{M_{neo}}, \qquad (1)$$

where p is the impulse of the projectile, M_i and V_i are the mass and velocity of the kinetic impactor, and M_{neo} is the mass of the Earth-threatening asteroid. Using scaling relationships for strength controlled cratering [*Holsapple and Schmidt*, 1982; *Housen et al.*, 1983; *Holsapple* 1993], *Harris and Ahrens* [1994] show that the impulse p_{KE} is given by

$$p_{KE} \approx 0.155 M_i \sqrt{\frac{Y}{\rho}} \left(\frac{\rho}{\rho_i}\right)\left(\frac{\rho V_i^2}{Y}\right), \qquad (2)$$

where Y is the yield strength of the rock comprising the asteroid, ρ is the density of the asteroid and ρ_i is the impactors density. The value of Y can range from about 1MPa (soft rock or ice) to 100 MPa (hard rock). Using gravity scaling relationships, *Harris and Ahrens* [1984] find that in the gravity regime the impulse p is given by

$$p_{KE} \approx 0.16 M_i \frac{V_i^{1.22}}{V_{esc}^{0.22}}, \qquad (3)$$

where V_{esc} is the escape velocity of the asteroid. All variables are in SI units.

The magnitude of ΔV as function of projectile mass is shown in Fig. 11 for impact into 1-km-dia. asteroid. The projectile mass required to deflect this asteroid by 1 cm / s ranges from 1.2×10^5 to 1.3×10^6 kg for impact velocities ranging from 40 km / s to 5 km / s.

Fragmenting rather than deflecting a potential asteroid threat may have serious implications for the survival of humanity on Earth. For example, impact scaling relationships [*Holsapple and Schmidt*, 1982] indicate that two smaller bodies will crater the Earth more efficiently than just one body. Furthermore, several authors [*Schultz and Gault*, 1982; *Melosh et al.*, 1990] have shown that a significant cloud of fine debris impacting the Earth's atmosphere may significantly increase the surface temperature, causing global wildfire as well as disastrous effects on animal life. Thus, when considering the kinetic impact technique for deflecting an asteroid, it is important to consider the risk of fragmentation. Several authors [*Housen and Holsapple*, 1990; *Davis and Ryan*, 1990; *Ryan and Melosh*, 1998] have shown that small asteroids become weaker to fragmentation as their size increases up to D < 1–3 km. This is because there are many more pre-existing weak flaws in a large asteroid versus a smaller one. When the asteroid size increases beyond 1 to 3 km, gravitational self-compressions deters fragmentation and strengthens larger asteroids. Asteroids, therefore, are their weakest around 1 to 3 km in diameter.

Holsapple [1993] indicates that the critical specific energy required to disrupt a 1-km asteroid is ~10^6 ergs/g. A 40 km/s projectile that achieves a ΔV of 1 cm/s would probably disrupt such an asteroid (see Figs. 11 and 12). Slower projectiles traveling at

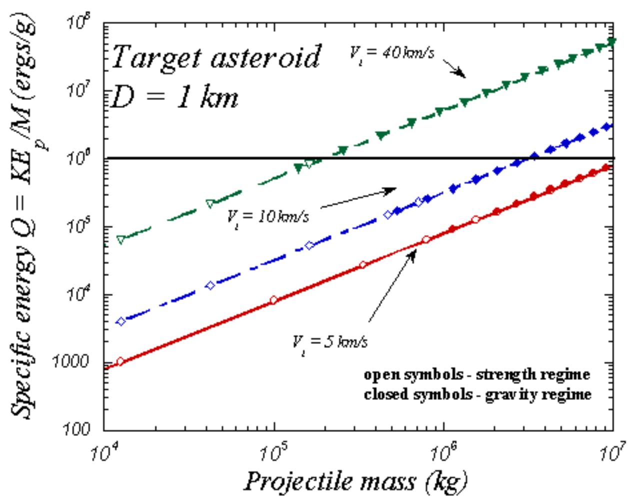

Figure 12. The specific energy of the impact (defined as the ratio of the projectile kinetic energy to the asteroids mass) as function of the projectile mass responsible for deflecting a 1-km-dia. asteroid by ΔV shown in Fig. 11. The horizontal line indicates the approximate critical specific energy that would suffice to disrupt a 1-km-dia. asteroid [see *Holsapple*, 1993].

5 to 10 km/s (Figs. 11 and 12) are about an order of magnitude below this specific energy. However, this critical specific energy may decrease with decreasing impact velocity because of the dependence of tensile fracture on strain-rate [*Housen and Holsapple*, 1990; *Davis and Ryan*, 1990; *Ryan and Melosh*, 1998]. As a result, when considering the kinetic energy impact for deflecting a few kilometer diameter asteroid, a disruption analysis must be undertaken. However, the kinetic energy impact deflection technique is probably viable for stronger smaller bodies (D < 1 km) without concern for fragmentation.

The specific impulse $I_{SP_{KE}}$ defined in Equation 4 is used to compare the impact technique relative to other mitigation options. Some values for this $I_{SP_{KE}}$ are shown in Fig. 11.

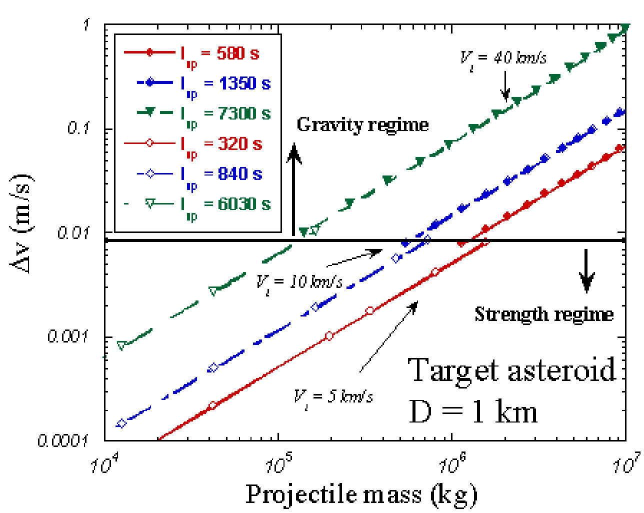

Figure 11. The deflection velocity ΔV achieved of a 1-km-dia. asteroid as a function of projectile mass for three different impact velocities (upside down triangle–40 km/s, diamond–10 km/s, circle–5 km/s). Open symbols are for impacts in the strength regime while closed symbols are for impacts in the gravity regime.

$$I_{SP_{KE}} = \frac{p_{KE}}{Mg} \tag{4}$$

The key technologies that would need to be developed to deflect asteroids by impact are: the targeting and terminal intercept system used for final maneuvers of the interceptor (or improved determination of projectile and asteroid orbits to within tens of meters a long time prior to impact), a delivery scheme that could provide the most mass and velocity for the least cost, and an accurate way to ensure that the impact occurs along the center mass of the threat in its orbit direction (to avoid momentum transfer tangential to the orbit direction). Other factors that could significantly influence the production and velocity of ejecta during cratering, but which have not been included in the above analysis, are the physical characteristics of the target asteroid (beyond its mass and strength) such as its porosity and volatile content. Initial studies [*Asphaug et al.*, 1998] indicate that an increase in the macroscopic porosity of an asteroid will increase the velocity of ejecta excavated, although the total amount of ejecta excavated should be reduced [*Cheng and Barnouin-Jha,* 1999].

Oblique impacts may also be viable for deflection but require more study. Such impacts increase the production and velocity of ejecta excavated early in the cratering process [*Schultz and Gault*, 1985; *Schultz*, 1999], but reduce the total production of ejecta [*Gault and Wedekind*, 1978] and the chances of disrupting an asteroid [*Cheng and Barnouin-Jha,* 1999].

Chemical Engines

All forms of propulsion, solar sails, and mass drivers in practice use the same delivery system and perform basically the same function. In all cases these objects must be soft-landed onto the threat, effectively "bolted on," and then perform a propulsion maneuver over a period of time to produce a large enough ΔV to change the course of the body. From this common base, the technologies then diverge somewhat in their efficiency and performance.

Traditional chemical propulsion systems perform a series of short, "impulsive" burns to provide the needed ΔV. These devices typically have an I_{SP} of 200 to 500 s, and provide a large amount of thrust in a short period of time. This is an advantage because the asteroid will most likely have an arbitrary rotation that would make continuous thrust difficult. Before this maneuver can be performed, however, it is important to know the location of the center of mass of the body, and provide most of the thrust through this point. Otherwise much of the energy used by the thruster will provide rotational, not translational, acceleration. Chemical propulsion systems are well developed and are only limited in the ΔV they can provide by the amount of fuel they carry, and the rated life of the engine. Cryogenic fuels such as liquid oxygen / liquid hydrogen should not be used because of the long mission times involved.

The velocity change that a chemical thruster can apply to a system, ΔV_{CE}, is dependent on the size of the system, the amount of fuel burned, and the efficiency of the engine.

$$\begin{aligned}\Delta V_{CE} &= C^* \ln\left(\frac{M_o}{M_f}\right)\\ &= I_{SP_{CE}} \cdot g_o \ln\left(\frac{M_{NEO} + M_E + M_f}{M_{NEO} + M_E}\right).\end{aligned} \tag{5}$$

In Equation 5 C^* is the effective exhaust velocity of the fuel, where $C^* = I_{SP_{CE}} \cdot g_o$. Also, M_E is the mass of the chemical thruster payload without fuel in kg, M_f is the mass of the fuel, and g_o is the Earth's gravity at sea level.

Electrical Thrusters

Electrical propulsion systems differ from chemical ones in that they are much more efficient, but provide less thrust. Therefore, electrical systems must burn for days, months, or years to provide the same

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ΔV as chemical. In addition, the continuous nature of their thrusting would require thrust vector control and power cycling to account for the rotation of the body. A large power source would also be needed to provide energy to the thrusters. I_{SP}'s for these systems can range between 1000 to 10000 s; therefore, less fuel would be needed for them. Finally, the technology to build large electrical propulsion systems is not well developed at this time, so some technology development must be performed to make this a viable option.

As with chemical thrusters, electric thrusters provide velocity changes as a function of initial and final system masses and exit velocity.

$$\Delta V_{ET} = C * \ln\left(\frac{M_o}{M_f}\right) = I_{SP_{ET}} \cdot g_o \ln\left(\frac{M_{NEO} + M_E + M_f}{M_{NEO} + M_E}\right). \tag{6}$$

Here, the mass of the engine, M_E, is much more significant than for chemical systems since a large portion of mass is consumed by the engine power plant. The more power put into an electrical thruster, the higher the I_{SP}. This relationship is presented in Equation 7, where τ_{EP} is the thrust produced by the engine, η is the dimensionless thrust efficiency of the engine, and P_{in} is the power put into the system in Watts:

$$I_{SP_{ET}} = \frac{2\eta P_{in}}{g_o \tau_{EP}}. \tag{7}$$

Solar Sails

Of the options available for impact hazard mitigation, solar sails would require the most development to achieve an acceptable state of readiness. To date, no satellite has used solar sails as a primary form of propulsion. However, in the future solar sails could provide an extremely efficient use of mass to mitigate the hazard of impacts.

The two main problems associated with solar sails are the complex harness that would be required to attach the sails to an arbitrary rotating body and the large size needed to produce any significant acceleration. A solar sail needs to maintain a certain angle of incidence to the Sun to provide translational thrust in the proper direction. If the body were not rotating, this would be a workable problem. The weight of a gyroscopic, low friction harness might negate any mass efficiency effects that a sail would provide. In addition, the area of the sail would need to be extremely large to provide any significant amount of ΔV to the threat, making the chance that a rip in the thin sail could cause system failure that much more probable.

The change in characteristic velocity that a sail can provide is a linear function with respect to time, simply the acceleration of the sail-payload system multiplied by time. I_{SP} for a solar sail, also a function of time, is presented in Equation 8. Thus, the more lead time available the more efficient a sail would be. Sails could theoretically move virtually any size body given enough time and sail area. However, these are probably not feasible options unless a risk reduction development plan is initiated well in advance of their use.

$$I_{SP_{SS}} = \frac{\int_0 \tau_{SS} dt}{W_{ps}} \tag{8}$$

$$\tau_{SS} = p_o A \cos^2 \theta \frac{1}{U} \tag{9}$$

Here τ_{SS} is the thrust produced by the sail, W_{ps} is the weight of the sail and its supporting structure in kg, U is the distance of the sail from the sun in astronomical units (AU), θ is the angle of incidence of the Sun's rays, A is the area of the sail in meters, and p_o is the solar radiation pressure at the Earth's solar radius, defined to be 4.6E-06 N / m^2.

To determine the mass of the sail, a dimensionless variable called the lightness factor, λ_{SS} is specified. The lightness factor is used as a primary indicator for the performance of a solar sail [*MacNeal, R. H.*, 1972]. When λ_{SS} = 1.0, then the gravitational attraction of the Sun is exactly balanced by the radiation pressure. The equations for the mass of the sail, M_{SS}, and the lightness factor are presented in Equations 10 and 11.

$$M_{SS} = \left(7.675 \times 10^{-4} \cdot A\right) / \lambda_{SS} \tag{10}$$

$$\lambda_{SS} = \frac{p_o A}{F_G} \tag{11}$$

In Equation 11, F_G is the gravitational attraction of the Sun on the sail at 1 AU. Some example lightness factors include: ~0.5 for state-of-the-art (0.08 mil polycarbonate with 1500 A^o aluminum coating), ~1.0 for modest improvements over state-of-the-art (15-year development time), ~2.0 for significant improvement over state-of-the-art (40-year development time).

Directed Energy

Directed energy options include lasers or solar collectors. Both options beam energy onto an incoming threat to create outgassing, pressure, or vaporization of parts of the body to create either ΔV or fragmentation of the threat. This method is more effective against icy comets than metallic asteroids. Whereas solar collectors (reflects concentrated sunlight onto the body) would need to be in close proximity to the hazard, a laser could be ground based.

The engineering challenges of both are significant. Large solar collectors would have to be developed that maintain their orientation to the Sun and concentrate the solar energy at the same location on the rotating object. Powerful lasers do not currently exist to properly illuminate the target. Energy sources would have to be developed to power the laser again on a rotating object. Both systems would require an extremely accurate tracking system to keep a beam focused on the threat. Equations for the use of these technologies can be found in *Melosh, H. J. et al.* [1994].

Mass Drivers

Mass drivers use the threat body's own mass to provide a propulsive thrust. Once landed, pieces of the asteroid are mined and accelerated into escape velocity by the driver in a particular direction to provide thrust. This technology is not well defined, and would require complex low-gravity mining techniques that are not yet known. One advantage of this system is that the longer a mass driver operates the higher its I_{SP}, since no fuel should be used in the process. It is unclear how the large amounts of energy required to mine and accelerate the material will be produced.

Nuclear Detonations

For large hazards or short lead times, nuclear detonations are the only possible mitigation option in the near future. In terms of sheer efficiency, a nuclear detonation will provide the most ΔV for the least amount of weight of any system proposed here, given reasonably short mission times. No other device currently proposed can release the kind of energy per mass that a nuclear device can. Isp's, while varying sharply as the yield of the device increases, are on the order of 1×10^6 to 1×10^7s. For large hazards 1 km and over, this might be the only option available in the near future.

For hazard mitigation using detonation nuclear weapons, the device could be delivered in three ways. A weapon could be detonated in a standoff position close to the threat, causing portions of the body to melt and shear off providing change in the body's orbit. A device could be soft-landed and detonated causing cratering and ejecting mass into an escape velocity to produce a thrust. Finally, a device could be buried to either cause cratering or fragmentation of the threat into smaller pieces. This final option would either require a complex lander and low-g mining operation, or a penetrator with enough closing velocity to bury the device to sufficient depth before detonation.

A technological threshold that must be achieved before implementing any nuclear strategy for hazard mitigation is to understand how nuclear devices may fragment an asteroid. As discussed in the impact section, blowing a threatening asteroid apart while trying to deflect it may prove to be as grave a hazard as the original non-disrupted asteroid. Thus, when considering a nuclear strategy, methods to analyze the risk of disruption must be developed which ensure that, if an asteroid disrupts, the ΔV of most fragments avoids the Earth.

Standoff Detonation

Determining the change in velocity a standoff detonation will provide is difficult to determine because the geometry of the hazard plays such a large part in the equations, and this is generally not well defined until an imager can perform a detailed observation of the asteroid or comet. Another difficulty is getting the irradiated shell to blow off the body in a controlled direction. *Ahrens, T. J. and Harris, A. W,* [1994] goes into great detail about the equations used in determining the impulse created by an explosion. From these equations the specific impulses shown in Fig. 11 were derived.

Surface Detonation

In terms of efficiency, the surface detonation option provides the most ΔV for the least amount of mass delivered. As mentioned earlier, standoff detonations have a large amount of uncertainty in total impulse since the geometry and material composition of the body helps to determine the amount of energy imparted through the system. Although the mass of the buried device is equivalent to that of a surface detonation, the delivery system would be more complex and massive.

The change in velocity imparted by a surface detonation is equal to the momentum impulse, p, of the explosion divided by the mass of the hazard minus ejected mass, shown in Equation 12.

$$\Delta V_{SurD} = \frac{p}{(M_{NEO} - M_{ej})} \tag{12}$$

The velocity change is in km/s and M_{ej} the amount of material ejected from cratering caused by the explosion in kilogram.

Buried Detonation

Whereas buried detonation can be used to cause cratering effects similar to surface detonations, they are primarily used for fragmentation of a body. The size of the explosive charge needed to fragment a body is dependent on the size of the hazard, its material composition, and the depth of burial. Analyses of buried detonations are discussed in *Ahrens, T. J. and Harris, A.W,* [1994]; S*imonekno, V. A. et al.*, [1994], and *Shafer, B. P. et al.*, [1994]. If it is determined that a body is too large, or too little time is left to deflect it away from Earth using any of the techniques mentioned here, fragmentation is the last option. Though many critics state that fragmenting an object could cause more damage than the single body would, the increased surface area created by fragmenting a body allows atmospheric friction to erode more of the threat before impact, thus lessening the energy released from the bodies at impact.

Nuclear Engines

Another option that uses nuclear technology is that of nuclear engines. Nuclear engines have a higher I_{SP} than that of chemical engines (around 800 to 2500 s), thus requiring less fuel. A nuclear thermal rocket simply heats propellant in a reactor, then expands it through a nozzle to produce thrust. The same equations for ΔV and I_{SP} used for chemical thrusters can be used for nuclear engines.

Comparison of the Various Mitigation Techniques

The non-nuclear technologies discussed provide an array of options for smaller bodies. Propulsive technologies such as chemical, electric, solar sails, directed energy, and mass drivers would only be effective against smaller bodies. For larger bodies the amount of fuel needed would become too large to launch into orbit. Some key advantages of the propulsive technologies are that a low risk of fragmentation of the threat exists, and chemical systems are already well developed. However, the delivery system that would carry the thrusters to the hazard, soft land them on the body, strap them on, and control their operation would require a lengthy and costly development program.

Using the equations already presented, two figures have been generated to compare the efficiencies of the differing technologies as a function of mission duration and the diameter of hazard that can be deflected for a given time until impact. In Fig. 13, both the instantaneous and time-dependent specific impulses for each mitigation technique is presented.

Nuclear propulsion has been classified as continuous thrust, as have electric thrusters. However, both can become pseudo-impulsive since they have a limited supply of fuel. This figure assumes that the I_{SP} of the electric case

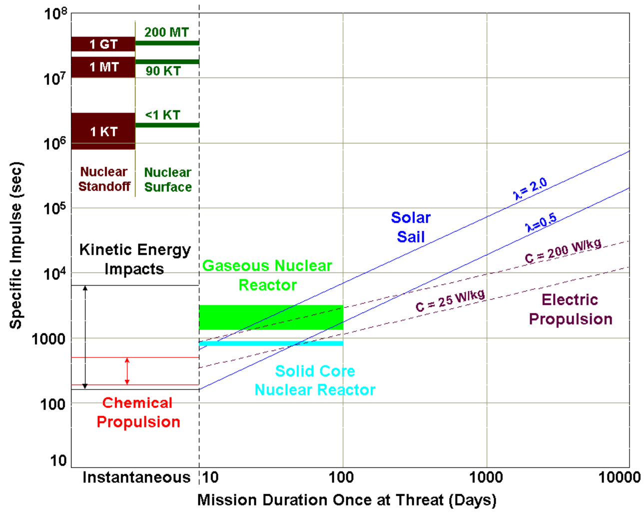

Figure 13. Specific impulse of divert technology versus mission duration. Assumes mitigation technology already at hazard. C is the figure of merit for electric propulsion. λ is the lightness factor for solar sails.

is variable as a function of mission duration, but is constant for nuclear engines.

As this figure shows, the most efficient way to divert an incoming threat is by nuclear detonation. Both have normalized I_{SP}s 2 to 3 orders of magnitude greater than their closest non-nuclear alternative. Kinetic energy impacts, chemical propulsion, and nuclear propulsion have a constant I_{SP}s whose value is a function of the equations presented earlier. They tend to have low I_{SP}s compared with the other options available. For short mission lead times, electric thrusters are more efficient than solar sails; however, this changes quickly as lead time increases. Mass drivers and directed energy were not included in this figure because equations to normalize their I_{SP}s could not be computed.

Figure 14 compares the time it takes to move an object of a given diameter the radius of the Earth for each different mitigation technique. The assumptions used to generate Fig. 14 are given in Appendix A.

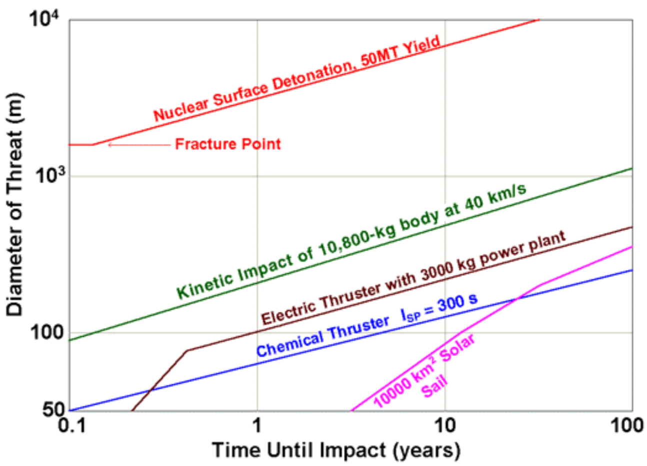

Figure 14. Diameter of incoming hazard versus time until impact. Assumes that mitigation technique is already at hazard.

As with Fig. 13, the most promising technique is shown to be nuclear detonations for large diameter objects or short lead times. Very high-speed kinetic energy impacts also seem to be highly desirable, but an impact at 40 km/s might cause a disruption of a body. Lower speed impacts, while safer, would not be able to divert as large of objects as shown in the figure. The break in the electric thruster line is the point at which the engine's fuel runs out.

Engineering Development Programs

For all of the deflection techniques already discussed in this section, some form of technology development program must occur before a technique can be considered a viable option to diverting an impact. The amount of development needed varies greatly between the different options. Some, such as kinetic impacts, only require the development of delivery and terminal tracking systems while others, such as solar sails, require a lengthy advances in materials, guidance systems, support structures, etc. A discussion of the engineering technologies required to develop some of the more credible near-term diversion technologies into working systems follows.

With the exception of ground-based lasers, all of the techniques presented in the hazard mitigation section require some form of payload to be delivered to the threat. The dynamics of orbit insertion, landing, or high-speed impact depend on the technique used and orbit geometry of the hazard. At present, however, no spacecraft has ever had a rendezvous with a low-gravity body. Several spacecraft, such as the Near Earth Asteroid Rendezvous (NEAR) mission and Galileo, have performed close flybys of asteroids and comets, but none have entered orbit or landed on a low-gravity body. Thus regardless of the option chosen, a great deal of research, development, design, testing, and implementation must occur before a working system can be fielded.

With the exception of a nuclear option, the most feasible near-term mitigation technique appears to be low-speed kinetic impacts. No orbit insertion or landing is required for this technique. Preliminary technology development studies should include example ballistic impacts on small orbital bodies, as well as tests of high-speed targeting and maneuver systems. A prospective asteroid or comet would have to be scouted ahead of time to determine some basic physical properties, such as shape, size, density, and material composition. Finally, a full test of a working system should be performed on a non-threatening asteroid to validate the hardware and analyze the results.

Even though kinetic impacts are the most basic way of imparting ΔV to a hazard, they are not necessarily the most efficient or reliable. As shown in earlier equations, to improve I_{SP} the closing speed of the interceptor must be increased. However, even though higher speeds increase the amount of energy imparted to the asteroid or comet, they also increase the risk of fragmenting the body. The proposed technique of landing a chemical or electrical thruster on the surface of the body would provide a more gentile impulse to provide the same ΔV without risk of fragmentation. However, this technique requires orbit and landing capabilities on a very low gravity body. Already programs are in place for demonstrating orbit insertion techniques. The NEAR satellite should enter orbit of the asteroid 433 Eros in February 2000. Other proposed missions such as Deep Space 4, ESA's ROSETTA mission the Near Earth Asteroid Prospector (NEAP) will also validate the technologies needed to rendezvous and orbit a small celestial object.

Once the technologies that allow a rendezvous with an asteroid or comet have been demonstrated, the technology to attach the spacecraft to the object should be devised and an attempt to soft land on a low-gravity body should be undertaken. This is a difficult undertaking because the rotational properties and material composition of the body may not be known until the satellite is already on station. Thus it would be hard to predict ahead of time the fuel levels and structural requirements of a lander. The next problem that arises is how to strap onto a rocky or icy body in a low-gravity environment. Although driving spikes into the surface at first seems like the most reasonable method, it is unclear whether this will provide enough strength and stability to secure the platform when the main engine is firing. Again, validation tests should be run ahead of an actual mission to answer these questions.

The final technology development that could be done in the near future is to develop engine systems that would be used once a satellite has landed and attached itself to the hazard. Storable chemical engines have already been proven in the Apollo lunar lander decent engine. This engine has been thoroughly tested and could be modified to fit onto a landing platform and provide the impulse needed to divert an asteroid or comet. It is also feasible to develop highly efficient electric propulsion engines in the next 20 to 40 years. The main problem with electric propulsion is not the engine itself, but providing the large amounts of power needed to provide adequate thrust and exit velocity to make the engine an effective alternative to chemical systems.

Assuming the design and launch of a simple demonstration satellite to perform a relatively slow 5 km/s impact on a already known near-Earth object were to start today, it would take a minimum of 5 years to build, launch, impact, and analyze test data. For the more complicated systems, validation of technology is decades away. Fielding an operational system would take even longer. Figures 15 to 17 present our preliminary estimates of defection capabilities 10, 20, and 40 years into the future. As with Fig. 14, time until impact does not account for the length of time required to travel to the hazard. It is assumed that it will take less than two years, though in reality delivery might take much longer. Even though a last resort defense for objects detected with lead impact times of less than 20 years from present could theoretically be fielded, such a system would be

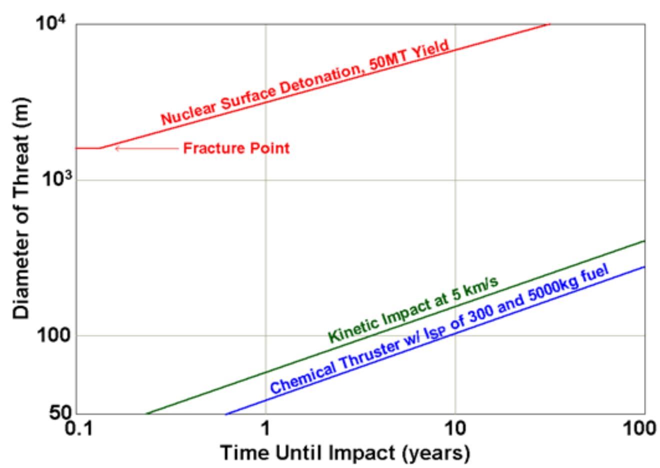

Figure 15. Feasible diversion options 10 years from present.

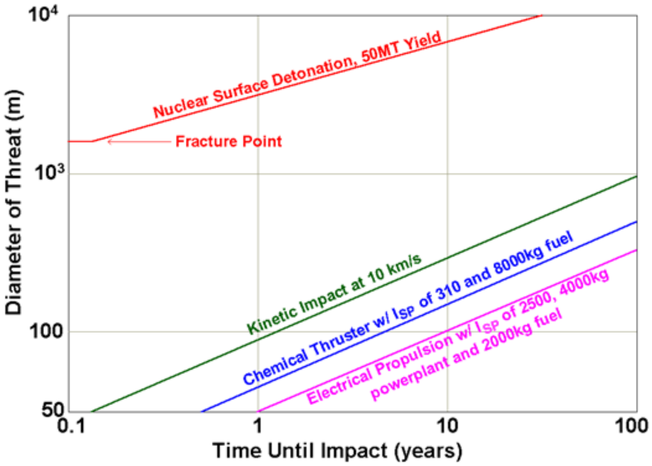

Figure 16. Feasible diversion options 20 years from present.

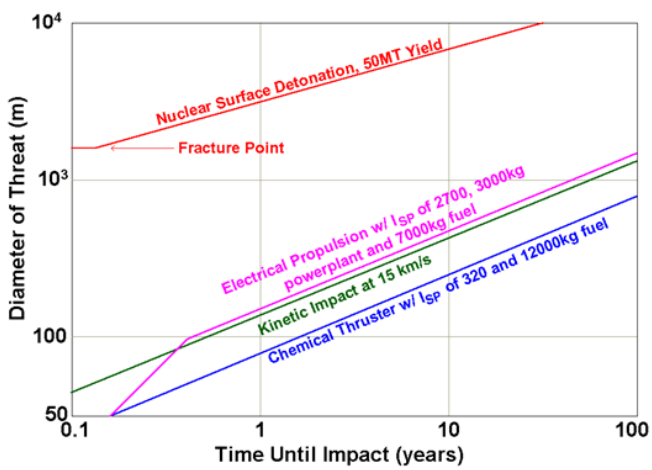

Figure 17. Feasible diversion options 40 years from present.

untested, most likely nuclear in nature, could only affect smaller bodies (less than 500 m in diameter), and would probably have to rely on fragmentation of the hazard rather than diverting it from impact. This capability must be augmented, well in advance of any direct threat, with studies and tests to help assure that such a last-resort defense will actually minimize the risk of fatalities on Earth.

Hazard Mitigation Conclusion

The non-nuclear technologies discussed provide an array of options for smaller bodies. Propulsive technologies such as chemical, electric, solar sails, directed energy, and mass drivers would only be effective against smaller bodies. Anything larger than this, and the amount of fuel or lead time needed, would become too large to launch to orbit. Some key advantages of the propulsive technologies are that there is a low risk of fragmentation of the threat, and chemical systems are already well developed. However, the delivery system that would carry the thrusters to the hazard, soft land them on the body, strap them on, and control their operation would require a lengthy and costly development program. By far the biggest disadvantage to nuclear detonations is the political and social ramifications of developing such a large, accurately targetable, highly reliable nuclear weapon. This might infringe on several treaties, and a launch failure could cause extreme amounts of destruction. However, because of the current state of technology, not a lot of modifications would be needed to upgrade current weapons from their military role to one of hazard mitigation. As with the other technologies presented in this analysis, the most development would be required in the area of the targeting and delivery system.

This section was not meant to determine which of the mitigation options is the best, but to present a basic understanding of the function of each. In terms of near-term technology, however, disregarding political and social factors, and assuming a reasonably high assurance of launch success, nuclear weapons would be the most effective device to use against bodies with a diameter greater than 500 m. For objects smaller than 500 m each option has its own effectiveness for a given set of mission parameters, and should be studied in more detail on a case-

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by-case basis. Ground-based lasers, mass drivers, or solar sails were not included as feasible technologies because at this time it is not clear that enough research and development can be performed to make these systems viable alternatives in 20 to 40 years. However, these options have vast potential and should continue to be studied.

5.0 SOLDIERS

Asteroid Properties

In the following section, the various physical and chemical properties of an asteroid that must be known to effectively implement any deflection technology and the type of instrument payload that can be used to determine these properties are discussed. The instrument payload would become an integral part of the Soldier or Scout spacecraft. In the first section, the most important physical properties and the instrumentation needed to measure them are discussed. In the following section, the lesser but equally important physical and chemical properties of an asteroid that must be known to best achieve the stated goal of deflection are addressed as is the instrumentation needed to determine these factors.

Critical Asteroid Properties and Instrument Payload

The most important characteristics of the asteroid required for deflection include its mass, density, and rotation rate. The asteroid mass determines the minimum mass and velocity of the kinetic energy impactor, the minimum fuel required by the chemical thrusters, the minimum size of the nuclear device, etc., required for deflection. The density of the asteroids provides a first order view of the internal structure of the asteroid, helping to define whether or not the asteroid is either a porous or volatile-rich rock, or a solid rock. This density is important for implementing the most effective deflection strategy. For example, if the asteroid possesses a low density and many large craters whose diameters approach that of the asteroid (e.g., 244 Mathilde [see *Veverka, et al.*, 1997; *Yeomans et al.* ,1997]), this asteroid is probably quite porous and / or volatile rich. As with 244 Mathilde, such an asteroid probably has a better ability to withstand fragmentation during either an impact or a nuclear event. A large impactor or nuclear device could, therefore, be used to deflect this body without risking fragmentation. Knowledge of the rotation rate is equally important in implementing any of the deflection techniques. For example, landing a thruster on the surface of a very rapidly rotation asteroid will be significantly more challenging than on a slow rotating body.

The mass and density of an asteroid can be obtained by a simple flyby using a visual imager and some tracking technique (e.g., Doppler tracking) of the spacecraft. Such tracking provides a measurement of the mass of the asteroid while the imager determines its size and shape. Combining this information allows one to compute the asteroid's density. NEAR used such a technique to determine the density of 244 Mathilde [*Yeomans et al.*,1997] and 241 Eros.

The rotation rate of the asteroid can be determined from the space-based Sentries by observing the periodic variation of the area and / or average albedo of the visible surface [e.g., *Harris and Lupishko*, 1989]. This technique is already extensively used from Earth-based observations of asteroids [e.g., *Binzel et al.* ,1989].

Second Order Physical Parameters and Instrument Payload

Although such cursory information provides a good first order understanding of the physical properties of an asteroid, a more complete picture of the physical characteristics of the asteroids is required by all the deflection techniques discussed. A thorough understanding of the physical nature of the asteroids is required at two scales: (1) the upper tens of meters, and (2) the deep internal structure. In the upper tens of meters, the presence or lack of a soil or regolith layer that covers the asteroid must be determined. In the eventuality that such a layer is present, the physical characteristic of this layer must be known, including its dominant grain size, porosity, cohesiveness, and depth. The deep internal structure of the asteroid must address the questions: Is the asteroid a rubble pile or a single rock? If it is a rubble pile, it is important to determine from the deflection view point what holds the various rubble pieces together.

Determining the physical characteristic of the upper tens of meters influences the implementations of many of the technologies discussed. For example, a thick porous and non-cohesive layer covering the bedrock of the asteroid would require very impressive explosive anchors to attach a chemical thruster to the surface of the asteroid. Of equal importance, knowledge of the deep internal structure determines where on the asteroid the impulse responsible for deflecting the asteroid will be applied. It would be unfortunate to land a thruster on a piece of rubble and have it float off the rest of the asteroid when the thruster, intended to displace the entire asteroid, is fired.

Some chemical information on the threatening asteroid must also be known. For any of the shock-based techniques (kinetic impact and nuclear), the presence of volatile-rich materials can significantly influence the production of vapor. Although the exact consequence of vaporization are not known, several studies indicate for impacts and certainly also for nuclear devices, that significant vaporization alters cratering efficiency, ejecta production, and ejecta velocity [*Vickery*, 1986]. The impulse generated by these deflection techniques would be, therefore, influenced by the volatile content of the asteroid threat.

Several different instruments can determine the physical characteristic of the upper tens of meters of the asteroid. From orbit, a ground penetrating sounder will provide the necessary information. Such an instrument was flown on Apollo and will be flown in the future at Mars. Using a range of radar bands, the physical properties of the surface layers can be determined from appropriate models for the dielectric constants. The rock or boulder coverage can also be estimated. The sounder results can be complemented with a thermal infrared instrument (such as TES on Mars Global Surveyor) that allows determining the porosity at the surface of the regolith, as well as the mean or dominant grain size present.

If a lander is used, ground-penetrating radar will very effectively determine the stratigraphy and physical properties of the upper 10 to 30 m of the surface providing high-resolution data. As with the sounder, the properties of the regolith are determined via dielectric constants inverted from the radar data. This kind of radar can be dragged along the surface behind a rover.

To investigate the deep structure of the asteroid, two instruments are worth considering. From orbit, a laser altimeter or range finder can be used to determine variations in the local gravity field, by accurately measuring the asteroid shape while tracking the spacecraft's orbit. Such an instrument is currently being flown on NEAR. A radar sounder should be able to perform many of the same tasks discussed previously for the laser altimeter, but at a lesser resolution. The best possible measurements of the gravity field are ensuring low orbits relative to the surface of asteroid (>km for 1-km-dia. asteroid) to obtain the highest degree possible for spherical harmonic model of the shape of the asteroid. Although not unique, appropriate models of the asteroid that reproduce both the observed topography and gravity field provide excellent information on the interior structure of an asteroid. Such data will be able to distinguish, for example, between zones in the asteroid of significant density differences, possibly identifying individual rubble pieces. These models, however, do not allow locating the position of these zones at depth. Furthermore, the presence of faults and fractures can only be identified at the surface and inferred at depth.

A laser with sufficiently high-topographic resolution (<1 m) will be able to detect directly the flexure of an asteroid due to solar tides, thereby indicating the asteroid's strength. The NEAR laser range finder (NLR) hopes to measure such flexure once in orbit at Eros.

A seismic array is composed of at least three seismometers but preferably more, would significantly complement the gravity and topography data of the laser or radar to provide extremely accurate view of the deep interior structure of an asteroid. These would be placed at many different locations across the surface of the threatening asteroid, along with a seismic source, namely a Vibroseis. The Vibroseis would thump the asteroid's surface, and the seismic array would record the resulting signals. Both the Vibroseis and seismometers would best operate if attached to bedrock. As on the Earth, the various seismic stations could detect a variety of features: location of faults and fracture within the asteroid, regions of competent rock, regions of brecciated rock, regions of layering, and so on. With sufficient seismic stations, and by placing the vibroseiser at several sites on the asteroid, the seismic data obtained could be collated to the gravity and topography data to obtain a full 3-D tomographic map of the asteroid's interior. With the ground penetrating radar and/or sounder data, a detailed picture of the entire physical structure of the asteroid would be established, including global zones of weakness, and a rubble pile structure.

The presence of volatiles at depth is difficult to determine. However, the presence of such materials at the surface can be easily determined using a hyperspectral spectrometer. An instrument that is sensitive to wavelengths of light ranging from 0.4 to 3 mm will be able to detect H_2O ice, CO_2 ice, carbonates, and other clay minerals indicative of volatiles [e.g., *Gaffey et al.*, 1993]. If such materials are detected at the surface, and the density of the asteroid is low, it is likely that some fraction of the asteroid possesses volatiles. If either one of the previously mentioned radar, and the seismic data do not reveal an obviously porous structure for a low-density asteroid, then, regardless of the spectrometer results, it is very likely that the asteroid is volatile rich.

Soldier Function

The two functions of the Soldier spacecraft are sequentially the surveying/evaluating of the target asteroid or comet threat (target), and then the alteration of the target's orbit away from Earth intercept. A soldier will carry varying equipment depending on the diversion strategy to be used. A suite of selected scientific instruments will carry out survey or scouting functions. Integral to the Soldier design, or independently dispatched as a dedicated "Scout," the survey package characterize the essential physical properties, rotation, composition, strength, and interior structure of the target body. The time scale for this survey phase is on the order of days to weeks. After this evaluation the Soldier, depending on its mode of operation, will dock, grapple, or intercept the target and impart a ΔV to it to divert it from a collision trajectory with Earth.

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Soldier Design

Soldier design will rely on the method chosen for diverting the target, but will generally fall into one of two broad categories: rendezvous and intercept. In the case of a rendezvous-type diversion, the Soldier arrives in the vicinity of the target, matches orbital velocities, surveys, and then lands on or docks with it, and executes its diversion function. The prime example of a rendezvous diversion is a chemical rocket used to push the target. In the intercept case, the Soldier arrives at high velocity, spending little time in the vicinity of the body before executing its diversion action. The main examples of intercepts are impacts and nuclear devices. Best coverage of the variety of asteroid and comet threats may require a mix of both Soldier types. Soldier equipment by function is summarized in Table 2.

Intercept Diversion

The intercept Soldier is actually a pair of spacecraft. The first spacecraft is an independent "Scout" craft that carries the survey and analysis instruments (see Table 2) and rendezvous with the target to perform its assessments long before the diversion parts, or the Soldier proper. The survey mission profile calls for an orbit that intercepts the target with small relative velocity so that the Scout may settle into orbit around the target or a standoff position close to the target. In this position, the Scout conducts its survey operations, studying the asteroids physical properties with the goal of determining optimum location and other parameters for the diversion operation. Ideally, the characterization phase would take no more than a matter of weeks, as full scientific characterization and of the asteroid (cf., NEAR) is not warranted. Impact target, or warhead detonation site selection is the crucial product of this phase, and will be constrained by the surface morphology, regolith properties, rotation axis, interior structure, and center of mass of the target. As a stand-alone survey craft, the Scout may be able to draw on significant heritage from early asteroid missions like NEAR or MUSES-C.

The Soldier component of the intercept diversion would be composed of an inert mass (in the impact case) or a nuclear warhead (or number of warheads) with steering, targeting, and guidance and control systems. The Soldier would be launched independent of the Scout on a trajectory that would intercept the target at a desired time, place, and relative velocity to maximize the diverting effect. Depending on the situation, the Soldier could be launched at the same time or after the Scout is dispatched, or after it has completed its assessment of the target. The most essential components of an intercept Soldier are highly accurate targeting and guidance systems. They will be have to ensure delivery of the payload to the proper place at relative velocities of up to several tens of kilometers per second.

Rendezvous Diversion

The rendezvous diversion could be performed either by a pair of spacecraft with separate Scout and Soldier functions, much like the intercept diversion or by a single Soldier craft that integrates both functions.

The single-craft rendezvous Soldier carries both the survey package and diversion package together in an all-in-one design to the target. The survey mission profile and goals are the same for the independent Scout, except that the site selection is dependent on the diversion method being carried. In the case of a chemical rocket "pusher," the search is for a Soldier landing / docking site on the target.

Once the survey is complete, the Soldier maneuvers close to the asteroid for landing / docking at the selected site. The Soldier will use anchors or grapples to secure itself to the target surface. Anchor design must be

Table 2. Soldier Equipment

Package	Equipment	Purpose	Engineering issues
Scout/ survey*	Imager/spectrometer	Composition, regolith characterization, shape, rotation	Optimization for survey function
	GPR radar	Interior structure, regolith	Optimization
	Seismic network	Interior structure, composition	Design of operable and deliverable seismic network and signal source
	Lidar	Shape	Optimization
	Radio science	Gravity field, interior structure	Optimization
	Power/computer system**	Power/control instruments	Optimization
Intercept diversion	Inert mass or warhead	Provide ΔV for diversion	Mass or warhead strength tradeoffs
	Targeting system	Provide rapid response information for high velocity approach to target	Targeting at tens of km/s
	Guidance control and thrusters	Course corrections duringhigh velocity approach	Course adjustment to ensure intercept at tens of km/s
Rendezvous diversion	Anchors	Secure soldier to target surface	Materials, mechanics, strength, ability to anchor to various potential target materials
	Gimbals	Reposition soldier to align thrust direction for maximum effect	Mechanism, control, strength
	Thrusters	Provide ΔV for diversion	Chemical, ion, or other propellants. Thruster design. Restartability. Sector firing
	Structure		Must withstand docking and diversion activities
	Power system	Power in-situ equipment	Type (Solar/RTG/other), required power
	Communications		Autonomy issues
	Controller electronics	Enable synchronized or sector firing of thrusters	Synchronization of thruster and other functions, docking capabilities

* Scout/survey package can be an integral part of the rendezvous design, but must be separate, stand-alone spacecraft in the intercept design

** Can be part of all-in-one design of rendezvous soldier flexible enough to allow for secure placement on surfaces ranging from metallic to rock to ice to deep regolith. Anchoring is essential for stability of a thruster-type design, especially if the applied thrust axis is to be off the spacecraft axis. The anchor must hold the Soldier so that the thrust is applied in the correct direction.

Direction of thrust will be controlled both by synchronized application of ΔV (e.g., thruster firing timed as required by the rotation of the target), and if necessary, by rotating the thrusters or the entire soldier by a set of gimbals.

Design Diversity

More than one Soldier design may be necessary. Assuming the Sentry surveys and its predecessors do not encounter imminent threats in their early years, the three main threat reservoirs will be long lead-time asteroids and short lead-time comets, and imminent unexpected threats. The first category will be amenable to mitigation by either rendezvous or intercept diversion methods. The long warning will likely allow a choice of methods and timing. The second category is unlikely to provide enough warning to perform a full rendezvous mission. An intercept diversion can potentially take less time to execute, especially if essential survey functions can be accomplished by a flyby Scout rather than rendezvous. The final category will use whatever method is most expedient with whatever information can be obtained quickly.

The different threats suggest that both intercept and rendezvous Soldier designs (including integrated Soldier/Scouts and independent spacecraft) be utilized and kept on station. This will also help to enable functional redundancy of either the scouting or diversion missions.

Engineering Issues

Of necessity, Soldier design and engineering is treated here in a cursory manner. A large number of engineering issues that need to be addressed are recognized for both Scout and Soldier development. Some of these are summarized in Table 2.

Suggested Number and Location of Soldiers

At least two Soldier spacecraft need to be ready for almost immediate launch from Earth, preferably within 5 days, one capable of rendezvous with a threatening object, the other able to intercept and deflect or destroy it. However, most threatening objects would approach the Earth on trajectories that can only be reached with a high-velocity intercept trajectory, precluding rendezvous except in the case where there are many years between the discovery of the object and the predicted impact. The possibility for rendezvous, and for lower-velocity intercept, trajectories is greatly increased by distributing Soldier spacecraft in "parking" orbits around at least the inner Solar System.

The CONTOUR mission [*Farquhar et al.*, 1997] uses a high-energy orbit with multiple Earth swingbys to intercept three comets. The Earth swingbys allow flexibility in the mission, permitting a change of comet targets after launch, with a high probability that even a LPC could be reached. Figure 18 shows CONTOUR's nominal trajectory projected in the ecliptic plane in a rotating reference plane, with fixed Sun-Earth line. The heliocentric orbits are inclined by as much as 12° to the ecliptic, resulting in much larger excursions in the rotating frame. Nonetheless, Soldiers in such orbits would be slower in reaching a new object than one that could be launched soon after discovery; one would always need to wait until the

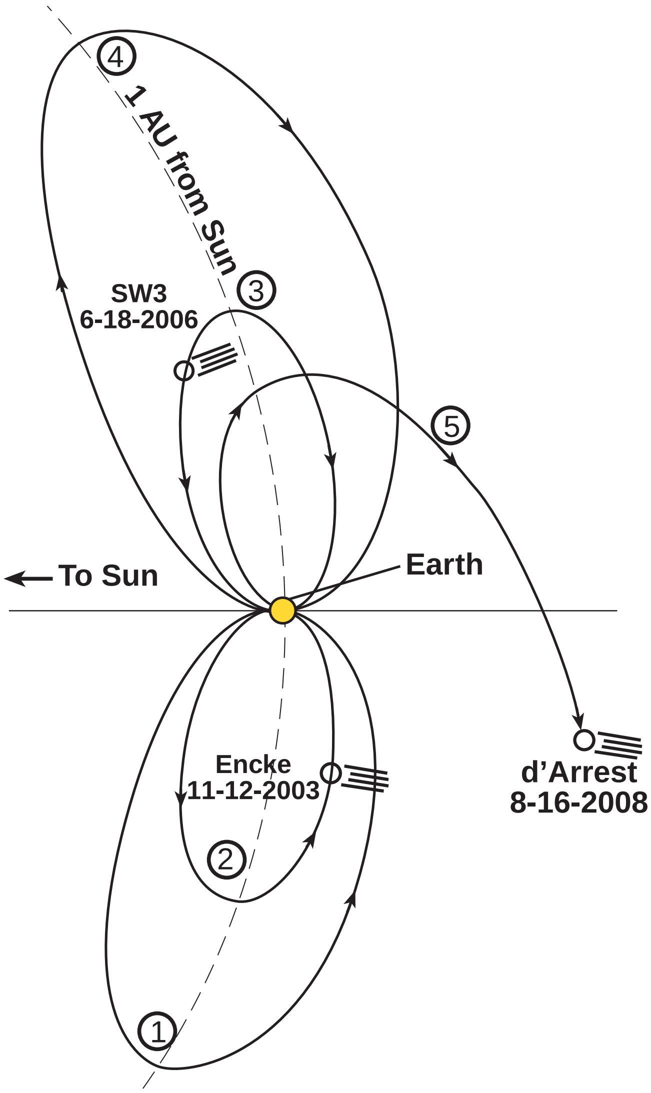

Figure 18. CONTOUR's nominal trajectory projected in the ecliptic plane in a rotating reference plane, with fixed Earth-Sun line.

Soldier returned to the Earth. However, Soldiers in orbits like this about other planets could reach many approaching objects much more quickly than those ready for launch from the Earth, and the possibilities for rendezvous with threatening objects also increases. The best planet for Soldiers would be Venus, since it (and the Soldiers accompanying it) has a shorter period of revolution (225 days) than the Earth, yet has nearly as much mass and similar size, allowing large bend angles at the Venus swingby to reach objects approaching from many different directions. Four Soldiers would permit pairs of spacecraft to return to Venus every 112 days, and in each pair, one would approach from the north, facilitating southern departure asyptotes, while the other would approach from the south, facilitating northern departure asyptotes. More Soldiers could be used to have more frequent returns, or to have different types of Soldiers (for example, rendezvous and intercept) in similar orbits.

Half the time, Venus is on the opposite side of the Sun as the Earth, giving good coverage of that region, which is much less accessible from the Earth. Soldiers in Mars and / or Mercury return orbits could help fill the gaps when Venus is on the same side of the Sun and relatively close to the Earth, but those planets have weaker gravity, restricting the achievable departure asymptote directions. Of course, Jupiter's gravity is very strong so that Soldiers in Jupiter-return orbits could reach virtually any part of the Solar System. However, with Jupiter's 12-year period, the time scales would be long, precluding intercepts as quick as those that could be arranged from inner Solar System Soldiers. If there is time, Earth-based Soldiers, or inner Solar System Soldiers in high-energy orbits, could be sent to Jupiter and even put into retrograde heliocentric orbits that, in conjunction with a good low-thrust propulsion system (solar-electric, nuclear-electric, or even solar sail), would allow rendezvous with most threatening objects. An optimum multiple Soldier architecture to minimize intercept and / or rendezvous times will be developed in our Phase 2 proposal.

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APPENDIX A
ASSUMPTIONS FOR FIGURE 14

For Fig. 14 the I_{SP} is held constant for the electric case. Other assumptions used to generate Fig. 14 include:

1. Assume hazard is a spherical body with an average density, ρ, of 3000 kg / m^3. Thus, the object's mass, in kg, is determined by the equation:

$$M_{NEO} = \pi D_{NEO}^3 \rho / 6 = 1.57E12(D_{NEO}^3) \qquad (13)$$

2. Three equations are used to determine the relationship between diameter and time until impact depending on the type of technology and the amount of lead time. For short duration periods (less than one year) the equation used is:

$$T = \frac{\delta}{\Delta V}, \qquad (14)$$

where T is the time until impact and δ is the amount of deflection needed (for all cases assumed to be 7000 km). For long duration (greater than one year) impulsive options the equation used is:

$$T = \frac{\delta}{3\Delta V}. \qquad (15)$$

This equation is used to account for the mean rate of drift imparted by an impulse applied tangentially to the orbit, in the orbit plane.

For long duration continuous thrusting maneuvers the equation used is:

$$T = \sqrt{\frac{2\delta M_{NEO}}{3\tau}}, \qquad (16)$$

where τ is the thrust of the mitigation technology in kg-km / s^2.

3. A Boeing Delta IV launch vehicle is assumed to be used to deliver the mitigation package. The current estimate is that this vehicle has the capability to place 15000 kg into a geostationary transfer orbit. From this, it is assumed that the launch vehicle can place 12000 kg of useable payload into a rendezvous orbit with the threat. Of this 12000 kg, it is assumed that 90% can be used for kinetic energy impacts (the rest is consumed in midcourse burns, final attitude adjustment, and other equipment needed to target the body). For technologies that require standoff or surface deliverable payload, it is assumed 75% of the 12000 kg is usable payload. For those technologies involving soft landings only 50% of the payload can be used for deflection.
4. To generate the kinetic impact line, a 10800-kg body impacting a hazard at a terminal velocity of 40 km / s is assumed. This analysis assumes the hazard would remain intact, all energy was imparted tangential to the orbit in the orbit plane, and the mass of ejected material was negligible compared to the mass of the body. Using Equations 1 and 2 in conjunction with Equation 13 for periods less than 1 year and Equation 14 for periods greater than one year produced the green line shown in Fig. 14.
5. For chemical thrusters Equation 4 was inserted into Equation 13 for periods less than one year and 14 for those greater than one year. Using the assumptions that only 6000 kg of usable payload could be soft landed on the hazard, the I_{SP} for the engine is 300 s (approximately the I_{SP} of the Apollo lunar decent engines), and that all of the landed mass was fuel (M_E = 0), the blue line in Fig. 14 was created. Again, all energy imparted to the system was translational tangential to the orbit in the orbit plane.

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6. For electric thrusters the thrust efficiency of the engine was assumed to be 0.5, typical for state-of-the-art systems. I_{SP} was assumed to be 2500 s. Using a power specific mass of 10 kg / kW (the best that solar arrays are theoretically capable), a 3000-kg power plant can produce 300 kW. From Equation 6 the thrust of the engine is able to be calculated. Because of the size of the power plant, the thruster only has 3000 kg of fuel left to use. While the engine is burning, Equation 15 is used to determine the deflection of the craft since acceleration is continuous. For times much greater than the burn time Equation 14 is used, since all ΔV has been imparted to the engine-hazard system. This accounts for the change in slope of the grey line in Fig. 14.
7. The square solar sails have a length and width of 100 km. In combining Equation 8 with Equation 15, it is assumed that the angle of incidence is always normal the incoming sun rays ($\theta = 0$), and the distance from the array to the Sun is approximately the radius of the Earth's orbit ($U = 1$). Because Equation 15 is a nonlinear function the slope should be curved. The analysis performed however just looked at 4 distinct hazard diameters and connected the points using lines, thus accounting for the changes in slope in the magenta line.
8. The surface nuclear detonation scenario involves a 9000-kg device (~50MT). Backwards calculations of numbers stated in *Ahrens, T. J. and Harris, A. W.* [1994] resulted in the device producing a momentum impulse of 4.9E9 kg · km / s. Using Equation 11 in Equations 13 and 14 again assumes that all energy and momentum applied to the hazard is transferred tangential to the orbit in the orbit plane. This analysis also assumes that, during cratering, the body remains intact if the ejected mass is less than 35% of the original mass of the body. The change in slope in the red line in Fig. 14 is this fragmentation point.

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