Mission Design and Analysis for Suborbital Intercept and Fragmentation of an Asteroid with Very Short Warning Time

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Small near-Earth objects (NEOs) ~50–150 m in size are far more numerous (hundreds of thousands to millions yet to be discovered) than larger NEOs. Small NEOs, which are mostly asteroids rather than comets, are very faint in the night sky due to their small sizes, and are, therefore, difficult to discover far in advance of Earth impact. Furthermore, even small NEOs are capable of creating explosions with energies on the order of tens or hundreds of megatons (Mt). We are, therefore, motivated to prepare to respond effectively to short warning time, small NEO impact scenarios. In this paper we explore the lower bound on actionable warning time by investigating the performance of notional upgraded Intercontinental Ballistic Missiles (ICBMs) to carry Nuclear Explosive Device (NED) payloads to intercept and disrupt a hypothetical incoming NEO at high altitudes (generally at least 2500 km above Earth). We conduct this investigation by developing optimal NEO intercept trajectories for a range of cases and comparing their performances. Our results show that suborbital NEO intercepts using Minuteman III or SM-3 IIA launch vehicles could achieve NEO intercept a few minutes prior to when the NEO would strike Earth. We also find that more powerful versions of the launch vehicles (e.g., total ΔV ~9.5–11 km/s) could intercept incoming NEOs several hours prior to when the NEO would strike Earth, if launched at least several days prior to the time of intercept. Finally, we discuss a number of limiting factors and practicalities that affect whether the notional systems we describe could become feasible.

I. Introduction

Earth has a well-documented history of impact by asteroids and comets that were sufficiently energetic, in terms of mass and impact velocity, to cause significant damage ranging from local or regional devastation to mass extinctions. Such asteroids and comets, whose orbits approach or cross Earth's orbit, are designated near-Earth objects (NEOs). At present there are at least tens of thousands of undiscovered NEOs larger than 100 m in diameter, and likely hundreds of thousands or even millions of undiscovered NEOs smaller than 100 m in diameter. Any of them may be found to be on a collision course with Earth and, therefore, require a planetary defense mission to deflect

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or destroy it prior to Earth impact. Our current NEO detection and tracking programs are making significant strides in discovering NEOs and monitoring their orbits for future Earth impacts to provide advance warning of any threats.

However, detecting Earth-impacting NEOs in advance is only one part of the solution. Early detection is not of much help unless we have well-tested and proven systems ready to be used against the incoming NEO. Towards this end, we have been engaged in a NASA Innovative Advanced Concepts (NIAC) Phase 2 study entitled "An Innovative Solution to NASA's NEO Impact Threat Mitigation Grand Challenge and Flight Validation Mission Architecture Development." In this research effort we have been developing designs for the Hypervelocity Asteroid Intercept Vehicle (HAIV), along with designs for missions (to harmless NEOs) during which the HAIV could be tested, refined, and, ultimately, made ready for action against actual incoming NEOs on Earth-impacting trajectories.[1,2,3,4]

One aspect of our research has been the development of incoming NEO response timelines, with an emphasis on assessing the minimum amount of warning for which a spacecraft mission could credibly be made ready and launched to intercept a NEO. This has led us to consider ultra-short warning scenarios for which the incoming NEO is not detected until less than 24 hours before its Earth impact. The purpose of considering such an incredibly short warning times is to help us establish a lower bound for the warning time through investigation of these highly stressing cases.

Our line of inquiry has practical applications beyond the assessment of lower bounds on actionable NEO warning time, in the sense that ultra-short warning scenarios are, in fact, realistic situations that will likely have to be dealt with in practice. At present we have no space-based NEO survey telescopes, despite the fact that scientists and engineers in the NEO community have repeatedly articulated the need for one.[5] The Jet Propulsion Laboratory (JPL) has proposed its NEOCam concept[a], while the B612 foundation is attempting to fund the development of their Sentinel telescope concept with private donations[b]. A new ground-based system, the Asteroid Terrestrial-impact Last Alert System (ATLAS), is currently under development and is scheduled to commence full-scale operations in early 2016. ATLAS should be able to provide approximately 3 weeks of warning for an incoming 100 m NEO, and 2 days of warning for an incoming 10 m NEO[c].

With only ground-based observation capabilities, we cannot observe NEOs in the sunward direction and are therefore blind to NEOs on such approach trajectories. This was demonstrated spectacularly by the impact and detonation of a ∼20 m NEO approximately 30 km above the city of Chelyabinsk in Russia. There was no warning of the NEO's approach, and its airburst detonation released approximately 500 kt of energy. The resulting shock wave shattered windows throughout the city, collapsed some building roofs, and injured approximately 1500 people[d]. Prior to this event, the NEO 2008 TC_3 was the first to be detected prior colliding with Earth. The approximately 4 m size NEO was detected approximately 20 hours before impacting the Earth and exploding high in the atmosphere over Sudan[e]. The only other instance of a NEO being detected prior to Earth impact is the NEO designated 2014 AA, an approximately 2–4 m object that entered Earth's atmosphere approximately 21 hours after being discovered on January 1st, 2014[f]. Earth is typically struck by NEOs several meters in size a few times each year.

Even when NEOs are not approaching from the sunward direction, small NEOs (being very faint in the night sky due to their small size) are difficult to detect and so we may, therefore, have relatively little warning of the approach of a small NEO that is still sizable enough to cause

[a]See `http://neocam.ipac.caltech.edu/`, last accessed 2014-06-04
[b]See `https://b612foundation.org/sentinel-mission/overview/`, last accessed 2014-06-04
[c]See `http://fallingstar.com/how_atlas_works.php`, last accessed 2014-06-04
[d]See `http://en.wikipedia.org/wiki/Chelyabinsk_meteor`, last accessed 2014-06-04
[e]See `http://neo.jpl.nasa.gov/news/2008tc3.html`, last accessed 2014-06-04
[f]See `http://neo.jpl.nasa.gov/news/news182a.html`, last accessed 2014-06-04

damage to people and infrastructure on Earth's surface. Small (e.g., <100 m diameter) NEOs are far more numerous than larger NEOs, and so one of the most likely NEO threats we face is that posed by a ∼50–150 m diameter NEO on an Earth-impacting trajectory that is not detected until shortly (hours, days, or perhaps a few months) before Earth impact. We are motivated to deal with such relatively small NEOs because the energies they are capable of delivering are not trivial. The aforementioned Chelyabinsk impactor was only ∼20 m in size but delivered approximately 500 kt of energy. It detonated relatively high above the ground because of its very shallow atmospheric entry angle, but if it had entered at a steeper angle, the damage to the ground would have been much worse. Under some general assumptions for NEO density, impact velocity, impact angle, and atmospheric entry behavior, NEOs 20–85 m in diameter are expected to create 230 kt to 28 Mt airbursts, and NEOs 100–150 m in diameter are expected to impact the ground with energies between 47 and 159 Mt[g].

In this paper we focus on a hypothetical assessment of what could be done to act against incoming NEOs with only a few hours or a few days of warning time, from intercept trajectory design and launch vehicle payload capabilities viewpoints. Current anti-ballistic missile (ABM) technology could be adapted for use against NEOs. The United States has deployed Ground Based Interceptor (GBI) missiles that are launched from silos and can intercept an enemy missile in the midcourse phase of flight with an Exoatmospheric Kill Vehicle (EKV). SM-3 missiles have a similar capability and are designed to be launched from ships. The higher altitude and larger payload requirements for a suborbital NEO intercept are beyond current GBI missiles. However, Intercontinental Ballistic Missiles (ICBM)s, such as the Minuteman III, do have adequate launch mass capability. In this paper, we assume that the Minuteman III based interceptor will be able to deliver a HAIV payload into a precision intercept trajectory against an incoming NEO. The HAIV system includes a Nuclear Explosive Device (NED) for the purposes of disrupting the NEO into a large number of small fragments.

In short warning time situations, when we don't have sufficient mission lead time to achieve large orbital dispersion of NEO fragments, full neutralization of the NEO impact threat is infeasible because most of the fragments will still strike Earth. However, if the NEO is fractured or fragmented into sufficiently small pieces prior to reaching Earth's atmosphere, each of those small pieces will break up sooner and the resulting airbursts will occur at safer (higher) altitudes. Thus, one way to frame the goal of a suborbital NEO intercept and fragmentation mission is that goal is essentially to reduce the probable impact damage of a 50 m NEO to be no greater than the damage level of several Chelyabinsk-like events.

II. Ideal Optimal Intercept

II.A. Problem Formulation

NEOs are on hyperbolic trajectories with respect to the Earth because they occupy heliocentric orbits and are not in captured orbits about the Earth. For a given hyperbolic NEO trajectory that intersects the Earth and a fixed launch site on Earth, the goal is to determine the optimal suborbital intercept trajectory. The performance of the missile is limited to its available total ΔV. The criterion for optimal intercept is defined as the maximum altitude of intercept from a fixed launch position on Earth. Because the NEO is on a hyperbolic Earth encounter trajectory, maximum altitude is equivalent to earliest intercept.

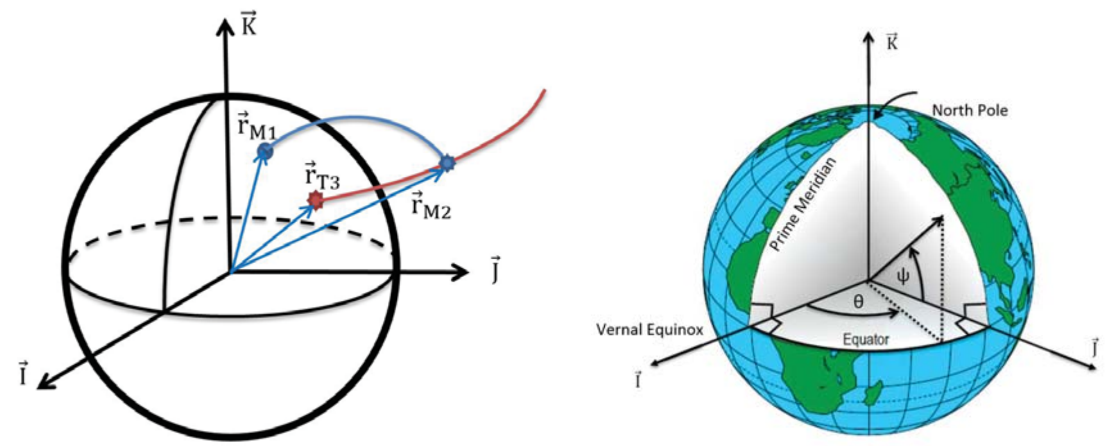
Our preliminary conceptual NEO intercept study consists of the following elements and assumptions:

[g]See `http://en.wikipedia.org/wiki/Impact_event`, last accessed on 2014-06-04

- Orbital elements of the target NEO at acquisition are assumed to be exactly known;
- The interceptor missile begins its flight from a few known locations on Earth's surface (defined by latitude and longitude relative to the Earth-Centered Inertial (ECI) frame);
- Each interceptor missile's performance is simply characterized by its available ΔV;
- The Earth is assumed to be a rotating sphere with negligible atmosphere;
- Each missile launches from Earth's surface with a single impulse (a multi-stage rocket model may be used in a future study)[h];
- Restricted two-body orbital dynamics are assumed.

II.A.1. Coordinate System

The NEO trajectory, interceptor trajectory, and positions on Earth's surface (of launch sites) are defined with respect to the ECI reference frame, shown in Figure 1(a). The time, distance, and speed units used here are seconds (s), km, and km/s, respectively. For nomenclature purposes, a subscript T refers to the target NEO, and a subscript M refers to the missile.



(a) ECI reference frame. (b) Earth's surface relative to the ECI frame.

Figure 1. Relevant problem parameters in the ECI frame.

Because the vectors are defined with respect to the ECI frame, it is not necessary to correlate the problem to a specific sidereal time. Instead, we assume that the prime meridian is aligned with the vernal equinox direction at the moment of interceptor launch. This makes it convenient to map the surface longitude to the ECI frame without having to calculate sidereal time. Figure 1(b) shows the orientation of the Earth's surface with respect to the ECI frame. The latitude and longitude positions are transformed into ECI position vectors as follows:

$$\vec{r} = R\cos\theta\cos\psi\vec{I} + R\sin\theta\cos\psi\vec{J} + R\sin\psi\vec{K} \tag{1}$$

where θ is longitude, ψ is latitude, $R = 6378.15$ km is the radius of the Earth, and $\vec{I}$, $\vec{J}$, and $\vec{K}$ comprise the ordered orthonormal basis vector triad for the ECI frame axes.

[h]As an example, the burn time for the optimal example shown herein is 187 seconds, while the total time of flight is 1394 seconds. Thus, the impulsive ΔV assumption used in this study is justified.

II.A.2. *Time Frame*

The NEO intercept scenario is independent of time. The timing of the major events such as target acquisition, missile launch, and intercept are all relative to an arbitrary time of impact. Each point in time is measured in seconds until impact. The times are identified using subscripts as described in Table 1.

Table 1. Event time nomenclature.

Symbol	Definition
t_0	Time of Target Detection
t_1	Time of Intercept Missile Launch
t_2	Time of Intercept
t_3	Time of Predicted Earth Impact

II.A.3. *Target Orbit*

The NEO's trajectory, hyperbolic with respect to the Earth, is defined in terms of the geocentric orbital elements at the time of acquisition (i.e., when the NEO is detected and its state is known). The acquisition time is arbitrary for this problem, but we assume that at the time of acquisition the NEO is beyond the range of any missile but inside the Earth's gravitational sphere of influence ($\sim$1,000,000 km). For the example problem in this paper, the target NEO orbit is designed such that it impacts the east coast of the United States with an incidence angle of 53.73° (measured from vertical) and an impact velocity of 14.933 km/s; these values are typical for Earth-impacting NEOs.

II.B. Technical Approach

II.B.1. *Equations of Motion*

The interceptor missile and the NEO are treated as point masses for which the governing dynamics are:

$$\dot{\vec{r}} = \vec{V} \tag{2}$$

$$\dot{\vec{V}} = -\frac{\mu\vec{r}}{r^3} \tag{3}$$

where $\vec{r}$ and $\vec{V}$ are the position and velocity vectors of the point mass, and μ is the gravitational parameter of Earth (3.986×10^5 km^3/s^2). The position and velocity are related to the semi-major axis of the orbit through the vis-viva equation and the equation of orbit, according to

$$\frac{V^2}{2} - \frac{\mu}{r} = -\frac{\mu}{2a} \tag{4}$$

$$r = \frac{a(1-e^2)}{1+e\cos\nu} \tag{5}$$

where a is the semi-major axis, e is the eccentricity, and ν is the true anomaly.

II.B.2. Optimization

The optimal suborbital intercept of an NEO from a fixed launch site is found by maximizing the altitude of intercept. The high altitude will limit any effects on Earth due to the nuclear explosion and give fragments more time to disperse. The optimal orbit will utilize the full ΔV available to the missile, and account for the additional speed provided by the Earth's rotation. There are only two free variables that determine the suborbital trajectory: the time-of-intercept (TOI), measured as seconds until impact, and the time-of-flight (TOF) of the missile. We will show that there is a unique optimal solution for typical intercept scenarios.

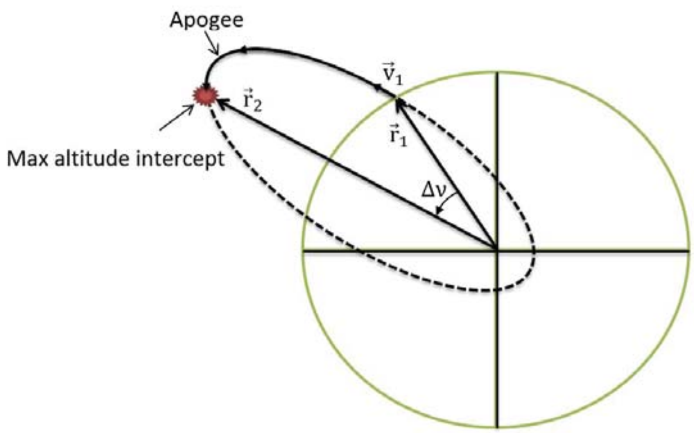


Figure 2. Intercept orbit diagram.

II.B.3. Accounting for the Rotation of Earth

The Earth's eastward rotation essentially provides a ΔV boost to the interceptor, allowing it to reach a higher altitude. The speed at the equator (in units of km/s) is estimated as

$$V_E = \frac{2\pi R}{24 \times 3,600} = 0.4638 \tag{6}$$

The speed of the Earth's surface in the ECI frame is dependent only on the latitude and longitude of the launch site. The inertial velocity vector of the launch site, $\vec{V_L}$ is found as

$$\vec{V_L} = -V_E \sin\theta \cos\psi \vec{I} + V_E \cos\theta \cos\psi \vec{J} \tag{7}$$

The velocity vector due to rotation at the launch site ($\vec{V_L}$) is added to the burn-out velocity vector of the booster ($\vec{V_{\text{bo}}}$) to obtain the initial velocity of the missile ($\vec{V_1}$) as

$$\vec{V_1} = \vec{V_{\text{bo}}} + \vec{V_L} \tag{8}$$

II.B.4. Determining Target NEO Position and Velocity as a Function of Time

The position of the NEO in the ECI frame must be known at any given point in time. From the orbital elements of the NEO at the acquisition point, the initial perifocal frame position and velocity vectors, $\vec{r}_1$ and $\vec{v}_1$, are calculated and transformed into the ECI frame. For a given TOF of the NEO, the position at that time can be found by solving Kepler's TOF equation in reverse.

$$t_2 - t_1 = \sqrt{-\frac{a^3}{\mu}}\left[(e \sinh H_1 - H_1) - (e \sinh H_2 - H_2)\right] \tag{9}$$

where H is the hyperbolic eccentric anomaly and is related to the true anomaly (ν) by

$$\tanh \frac{H_i}{2} = \sqrt{\frac{e-1}{e+1}} \tan \frac{\nu_i}{2} \tag{10}$$

II.B.5. Lambert Solver

Using the known positions of the launch site and the NEO at a given time t_2 (the time of intercept), along with the missile TOF, the required initial velocity vector for the missile can be found by solving Lambert's problem. The Lambert solver used in this paper is a slightly modified version of the universal variable method described in Ref. 6. This represents the open-loop guidance that is central to the optimization scheme.

II.B.6. MATLAB® Optimization Routine

The solver used to find the optimal solution is the `fmincon` function available in the MATLAB® Optimization Toolbox. `fmincon` is a constrained nonlinear multivariable minimization routine. There are two independent variables involved: TOF and TOI. Both variables are given upper and lower bounds to keep the solver from testing unreasonable points. An interior set point is chosen within the search window as the starting point. A constraint is placed on the solution such that the required ΔV is not permitted to exceed the maximum ΔV available to the missile. The NEO's altitude decreases monotonically with time because the NEO is on a hyperbolic trajectory with respect to the Earth. Thus, maximizing TOI is equivalent to maximizing intercept altitude. A graphical representation of an example search window is presented in Figure 3. Each of the contours is a line of constant ΔV required for intercept. For each ΔV curve, there is one TOF at which TOI is a maximum. This is the unique solution point for the ΔV at which intercept altitude is maximized. The locus of the optimal altitude intercepts is shown on the graph, as well as the set of intercepts for which the interceptor reaches the target NEO at apogee. It is interesting to note that the optimal intercept solution follows a nearly linear trend.

II.C. NEO Intercept Examples

Several example intercept scenarios and solutions are presented herein. For these examples, the NEO is discovered heading toward the east coast of the United States less than 11 hours before impact. The orbital elements of the target NEO are provided in Table 2. Interceptors based on the Minuteman III and the SM-3 Block IIA will be launched from silos at Minot Air Force Base (AFB), North Dakota. The maximum intercept altitude for each vehicle will be compared. Because the smaller SM-3 can be launched from a ship, an SM-3 will be launched from a position in the Gulf of Mexico. This is intended to show how positioning the launch site directly beneath the NEO's path can increase the intercept altitude.

II.C.1. Interceptor Characteristics

The Minuteman III and SM-3 Block IIA have a ΔV capability of 6.6 km/s and 5.5 km/s at burnout, respectively[i]. They are launched from a silo field in North Dakota with coordinates 48.5°

[i]Estimate of SM-3 Block IIA performance obtained from `http://www.fas.org/pubs/pir/2011winter/2011Winter-Anti-Satellite.pdf`, last accessed on 2014-06-12. Estimate of Minuteman III performance ob-

N, 101.4° W. For clarity, these interceptors will be referred to as interceptors A and B, respectively. Interceptor C will be a SM-3 Block IIA launched from a ship located at 25° N, 90° W.

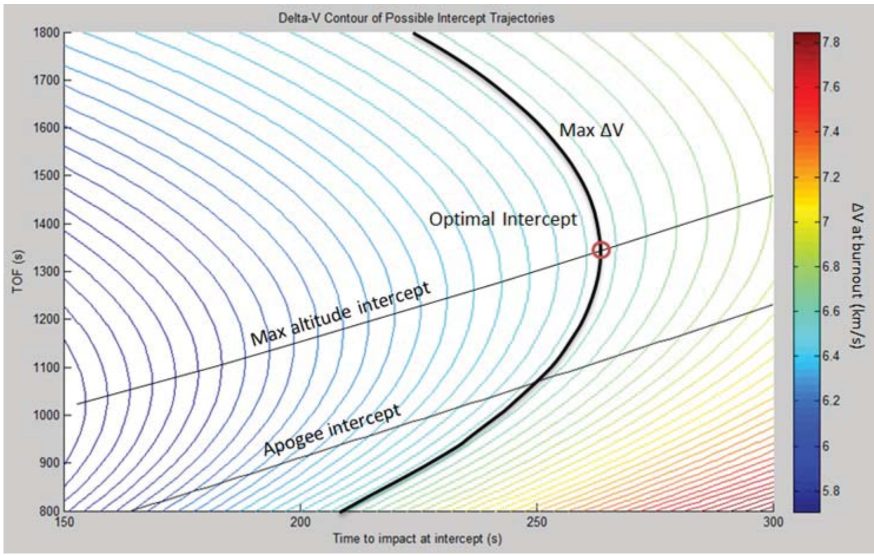


Figure 3. ΔV contour across interceptor launch window.

Table 2. Target orbital elements.

Orbital Element	Value
a	-4067.1 km
e	2.154
i	59°
Ω	256°
ω	100°
ν_0	243.4°

II.C.2. Results

Interceptor A reaches the highest intercept altitude of 2,625 km. Both of the smaller SM-3 missiles are able to achieve intercept, but at lower altitudes. Details of the intercept results are presented in Table 3. Figure 4 shows the NEO's path and the three interception trajectories relative to Earth. Interceptor C is launched from a point nearly directly beneath the NEO's path and, therefore, can reach a higher intercept than the same vehicle launched from further away. Due to the unpredictable nature of NEO impacts, however, it would not always be practical to move the launch site on short notice or have many launch sites around the country. Increasing the ΔV performance of the booster vehicle is much more effective, as will be discussed presently. The Minuteman III has 16.7% higher ΔV than the SM-3 used in this example, yet it can achieve intercept at 50% higher altitude when launched from the same location at the same target.

tained from `http://www.fas.org/nuke/guide/usa/icbm/lgm-30_3.htm` and `http://www.astronautix.com/lvs/mineman3.htm`, both last accessed on 2014-06-12. We derive conservative performance estimates from those sources.

Table 3. **Optimal Intercept Parameters**

Interceptor	A	B	C
Vehicle	Minuteman III	SM-3 IIA	SM-3 IIA
ΔV (km/s)	6.6	5.5	5.5
Launch Site	48.5°N 101.5°W	48.5°N 101.5°W	25°N 90°W
Impact Altitude (km)	2,625	1,269	2,044
Time Until Impact at Intercept (s)	264	133	209
Time of Flight (s)	1341	971	817
Intercept Closing Speed (km/s)	14.2	14.4	13.7

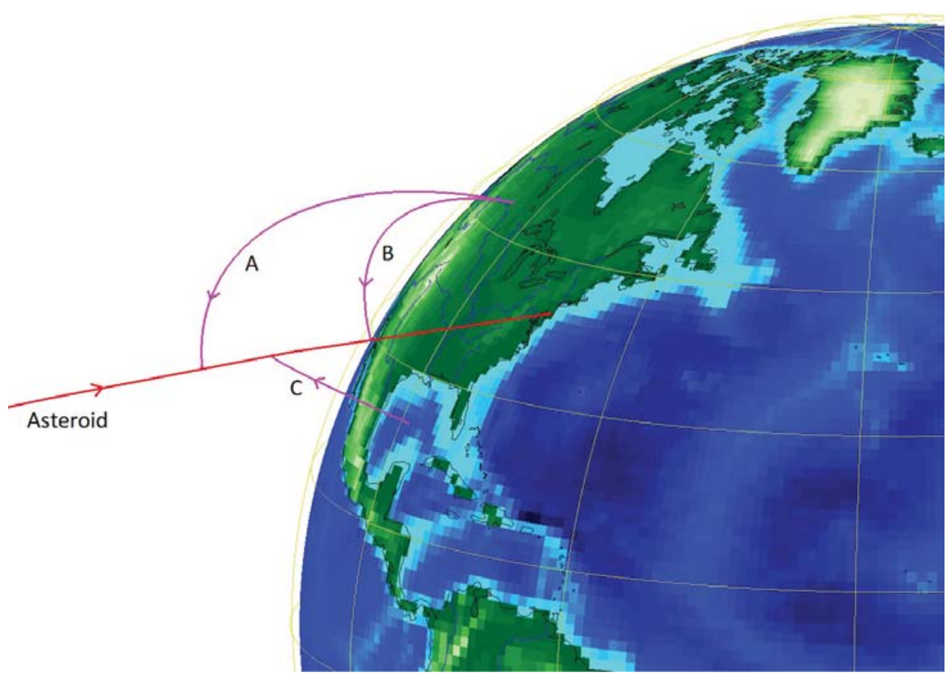


Figure 4. Ideal optimal intercept trajectories.

III. Planetary Defense Domes

For the purposes of planetary defense planning it is important to choose launch sites that maximize coverage of critical areas. The Minuteman III has sufficient range to protect most of North America if the silo location is chosen carefully. Assuming that intercept must occur above 1,000 km altitude to be reasonably safe, Figure 5 shows example defense coverage areas for the following three launch sites: Minot AFB in North Dakota, Vandenberg AFB in California, and Cape Canaveral in Florida.

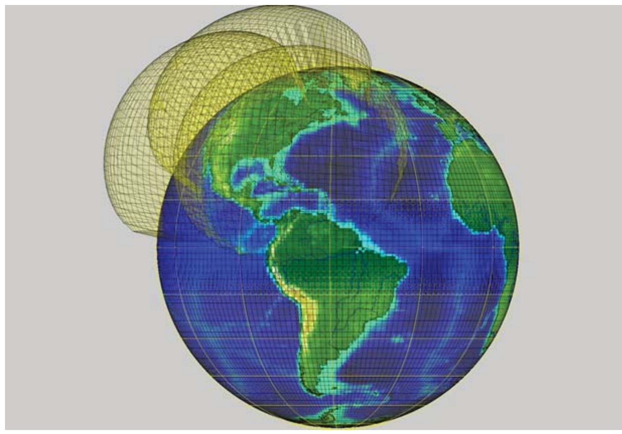


Figure 5. Planetary defense domes.

All of these sites are already used for testing/deployment of missiles, and together they create a fairly even spread of coverage across the entire continent. It should be noted that a simplified model was used to compute the defendable area for each site. The domes in Figure 5 are terminated at the apogee of the interceptor, creating a more conservative estimate of their range. This is roughly equivalent to being within the magenta ellipse in the 2-D special case shown in Figure 6, which will be discussed presently. The actual useful range of each missile site is thus larger than shown in Figure 5.

IV. Special Cases

IV.A. Uniform Gravity Case

An example with uniform gravity is included here to show that a unique optimal solution is possible in a simple case, and that the optimal trajectory does not necessarily involve NEO intercept at the interceptor's apogee. The missile is launched from a flat surface in an airless uniform gravity field in the same plane as the NEO. Thus, both the NEO and interceptor have parabolic trajectories. The NEO trajectory, missile launch site, and missile ΔV are fixed. The launch angle is varied to show all possible trajectories for the interceptor. Figure 6 shows the possible paths of the missile. The magenta ellipse represents the locus of the apogees for the interceptor paths. The inset in Figure 6 clearly shows that the maximum altitude intercept occurs outside of the magenta ellipse, meaning the missile will reach apogee before coming down to meet the NEO. This may seem counter-intuitive, but similar situations occur in most cases of Keplerian orbits as well.

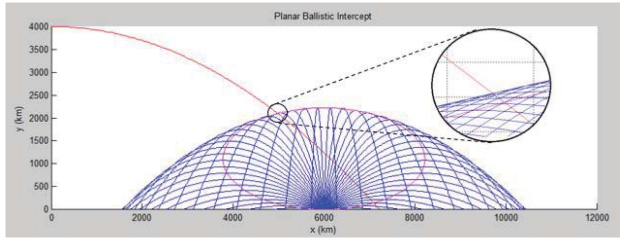


Figure 6. Planar ballistic intercept for uniform gravity case.

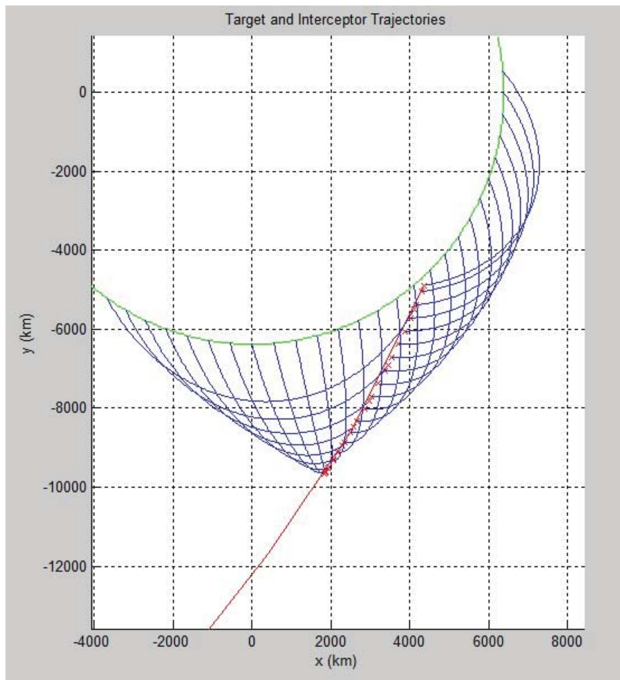


Figure 7. Two-body planar ballistic intercept.

IV.B. Coplanar Intercept

A slightly more realistic case than the uniform gravity case is presented here. Two-body Keplerian dynamics are used, but the NEO's path and the missile's trajectories are restricted to Earth's equatorial plane. As with the uniform gravity case, the NEO's path is constant and the ΔV capability of the missile is fixed. Figure 7 displays the optimal trajectories for several launch sites. The data in Figure 7 show that intercept at apogee case is not necessarily optimal.

IV.C. Late Intercept Solutions

After the optimal launch time for a certain target and interceptor configuration has passed, it is still possible to achieve intercept, albeit at a lower altitude. The launch time window for post-optimal solutions is bounded by the optimal launch time and the latest possible launch time for which intercept is still possible. The latter bound is equivalent to the minimum-TOF ballistic trajectory between the launch site and the target impact site. For every post-optimal launch time, t_1, there is a unique intercept trajectory that maximizes altitude. It can be shown that the maximum altitude possible for intercept decreases monotonically with later t_1. Therefore, the best time to launch after the optimal t_2 has passed is as soon as possible. Because t_1 is considered fixed for each trajectory calculation, we must vary TOF to obtain the earliest possible intercept. Figure 8 shows the locus of solutions across the post-optimal solution window. This plot was generated using Interceptor A from the above example. The maximum time to impact at intercept on the in Figure 8 represents the actual optimal solution. This calculation also serves to validate the original algorithm described in this paper, as both calculations result in the same optimal trajectory for the Interceptor A example. Figure 9 shows a sampling of post-optimal trajectories. The leftmost trajectory is the optimal solution, and the rightmost trajectory is the “last chance” solution.

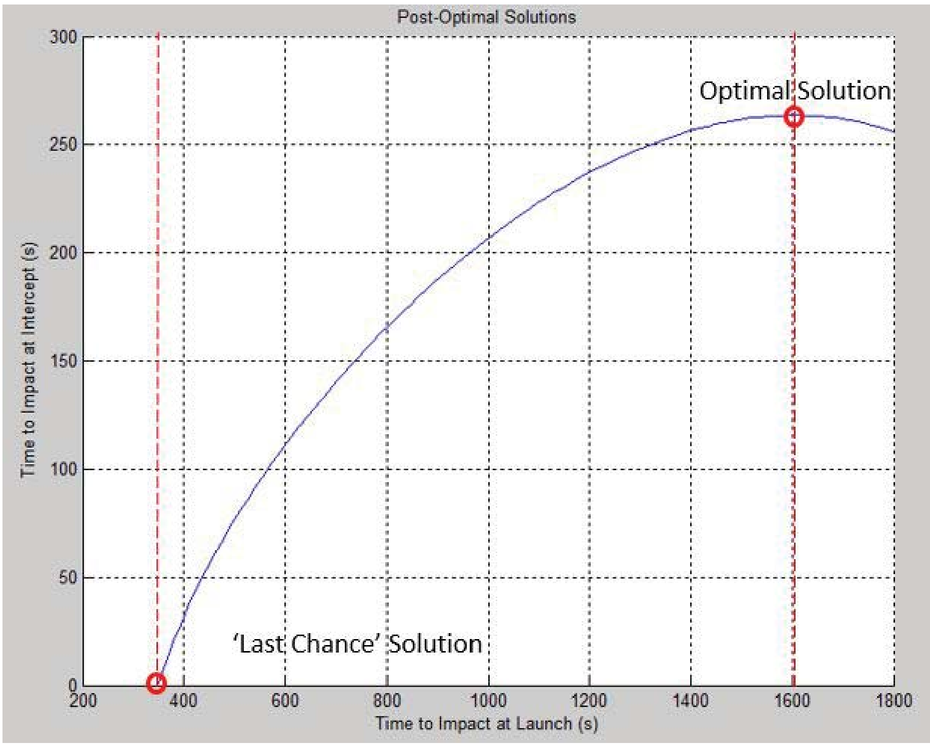


Figure 8. Late intercept solution window.

V. High ΔV Interceptors

In this section, interceptors with higher ΔV performance are considered as summarized in Table 4. Firstly, the Minotaur V launch vehicle with five solid fueled stages can launch a 300 kg payload with a ΔV of 9.5 km/s. This is much greater than the Minuteman III considered earlier, however the Minotaur V must be assembled on a launch pad and cannot be launched from a silo. The second case to consider is a fictional booster vehicle that can deliver the interceptor to nearly the moon’s mean orbit radius of 384,000 km. The NEO’s trajectory and launch site are kept the same as in the previous example.

It should be noted that the fictional booster approaches a parabolic escape orbit, although it remains a suborbital trajectory. Because of this, the results are very sensitive to small changes in ΔV. The Minotaur V can reach the NEO at an altitude nearly 5 times higher than the Minuteman III, and the intercept altitude increases exponentially with increasing launch ΔV. The time of flight is a limiting factor here. For the case of the fictional booster, intercept occurs 10 hours before impact, but the interceptor must be launched nearly 5 days before impact. The important point

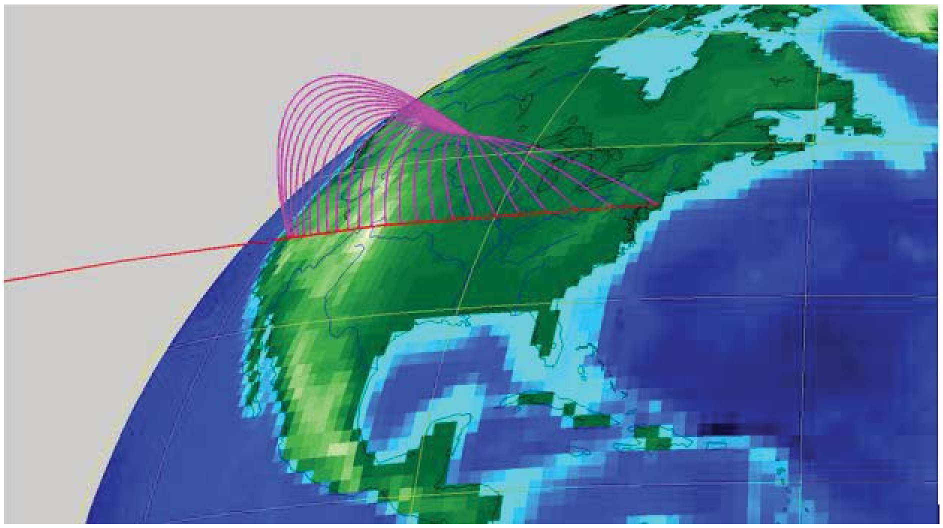


Figure 9. Late intercept trajectories.

Table 4. High ΔV intercept scenarios.

Vehicles	Minotaur-V	Fictional Booster
ΔV (km/s)	9.5	11.12
Launch Site	48.5°N 101.5°W	48.5°N 101.5°W
Impact Altitude (km)	15,101	393,620
Time Until Impact at Intercept (s)	1,388	38,623
Time of Flight (s)	5,779	414,030
Time of Flight	1.6 hrs	4.79 days

illustrated by these examples is that a small improvement in ΔV leads to a proportionately large increase in intercept altitude. The ΔV improvement can be achieved by using a larger booster or reducing the payload mass.

VI. Other Launch Vehicle Options

Although the Minuteman III ICBM is the primary example considered in this paper, it does not represent the only viable option for last-minute suborbital asteroid interception. This section looks at some of the alternatives provided in Tables 5 and 6. The list is limited to active or recently deactivated boosters that can launch at least a 300 kg payload into LEO, and large ICBMs. Liquid fueled launch vehicles are excluded from the list, as they require a more complicated and time-consuming fueling procedure that may not be compatible with a short warning launch. It is important to note that if there is enough time to assemble a large rocket on a pad before launching the interceptor, sending the interceptor into a parking orbit will generally be more effective than a purely suborbital intercept mission.

Table 5. Non-ballistic missile options.

Vehicle	Stages	Country	Platform	Payload to LEO (kg)
Minotaur I	4	US	Launch Pad	580
Minotaur IV	4	US	Launch Pad	1735
Minotaur V	5	US	Launch Pad	532 (GTO)
Pegasus	3	US	Air Launch	443
Shavit	3	Israel	Launch Pad	350
Start-1	4	Russia	Launch Pad	532
Taurus/Antares	4	US	Launch Pad	1320

Table 6. Ballistic missile options.

Vehicle	Stages	Country	Platform	Burnout Velocity (km/s)	Throw-Weight (kg)
Minuteman III	3	US	Silo	6.6	1150
Peacekeeper	4	US	Silo	6.7	3950
Trident II	3	US	Submarine	6.3	2800
R-36	3	Russia	Silo	$\sim$7.0	8800
GBI	4	US	Silo	6.0	$\sim$ 100
SM-3	4	US	Ship	5.5	$\sim$ 100

Both conventional launch vehicles and ballistic missiles are listed in Tables 5 and 6, along with a comparison of performance. An estimate of the payload to LEO is given for each launch vehicle, and an estimate of the burnout velocity and throw weight is given for the ballistic missiles. While not specific to a suborbital intercept mission with a 300 kg payload, these numbers provide a rough comparison of performance between the vehicles.

VII. Practical Considerations and Limitations

VII.A. Fragmentation and Airbursts

For any scenario in which suborbital intercept is the only option, it is unlikely that such an intercept will result in complete neutralization of the NEO. In such close proximity to the Earth, most, if not all, fragments will still strike the Earth. Similarly, any attempt to completely disrupt a large NEO would require a prohibitively large nuclear payload that would itself be dangerous to the Earth. For those reasons, the method of defense described herein is only effective against smaller ($\sim$50–150 m) NEOs. In the scope of the problem considered herein, however, we have assumed that large NEOs are more likely to be discovered relatively far in advance of their Earth impact dates; consequently, small NEOs are the most probable threat and are, therefore, focused on in this paper. NEOs much smaller than 50 m may break up in the atmosphere, depositing at most a shower of less dangerous fragments. However, the events at Tunguska in 1908 and Chelyabinsk in 2013 both provide evidence that large airbursts over land are capable of causing significant damage. On the other hand, a fragmented or fractured asteroid will tend to break up at a higher altitude, which would limit the damage caused by low-altitude airbursts.

VII.B. EMP Effects

The US and USSR both experimented with high altitude (400 to 500 km) nuclear detonations during the 1960s. It has been found that smaller yields and sufficiently high altitudes limit the effects on the ground. Additionally, we may be able to appropriately shape an NED explosion such that most of the explosion energy is directed toward the target NEO and away from the Earth. However, the possible EMP effects on both the ground and Earth-orbiting satellite infrastructure must be investigated further.

VII.C. Launch Vehicle Mission Planning Issues

The entire ascent flight of a launch vehicle from lift-off to the final target point in space basically consists of two phases: the atmospheric (or endoatmospheric) ascent and the vacuum (or exoatmospheric) ascent. Most launch vehicles are operated in open-loop guidance mode (but, obviously, in closed-loop flight control mode) during the atmospheric ascent flight. That is, launch vehicle guidance commands for achieving optimal flight trajectories are pre-computed in pre-mission planning. They are updated using the day-of-launch wind profile prior to launch, loaded into the launch vehicle guidance computer, and then used as pre-programmed guidance commands in actual ascent flight. Trajectory optimization tools are used to pre-compute optimal ascent trajectories for various flight conditions and path constraints.

Once a launch vehicle reaches an altitude of approximately 50 km or above, where the atmospheric effects can be ignored, the vehicle is then operated in closed-loop guidance mode for its exoatmospheric ascent. For example, the Space Shuttle was operated in open-loop ascent guidance mode for the powered first stage (ascent flight with the solid rocket boosters). The powered second stage (after solid rocket booster jettison) utilized a closed-loop guidance algorithm for its exoatmospheric ascent.

The open-loop guidance during the atmospheric ascent is not capable of autonomously adapting to significant off-nominal flight conditions. Pre-mission planning for generating optimal ascent trajectories has been known to be an extremely time-consuming and labor-intensive process. Consequently, rapid generation of optimal ascent trajectories and autonomous/adaptive closed-loop atmospheric ascent guidance have been a research topic of practical interest for many decades.[7,8,9,10,11]

Advanced ascent guidance technology needs to be further developed for operationally responsive launch vehicles required for planetary defense with very short warning times (e.g., 1 to 24 hours).

VIII. Future Work

As described previously, the primary purpose of this work is to begin understanding the most rapid launch response that might be possible against an incoming NEO, from a trajectory optimization perspective. However, the investigation undertaken towards that end has also led us to identify an array of topics that ought to be pursued in future work, if further development of the suborbital NEO intercept concept is pursued; such development is not currently within the scope of our Phase 2 NIAC study. The following is a summary of the key future work items identified:

- NED yield sizing for properly fragmenting ∼50–150 m NEOs of various types, structures, etc.
- Limiting cases for the sizes/types of NEOs for which a suborbital disruption attempt would be likely to be effective in reducing the negative effects on Earth. That is, if the incoming NEO is larger than a certain size, it may be that a disruption attempt would not be effective, produce an undesirable outcome, or require a NED that is too energetic to detonate near Earth.
- Examination of all effects on Earth (and Earth-orbiting satellite infrastructure) due to NED detonation at altitudes of 2500 km or higher.
- How quickly a dedicated launcher (e.g., silo-based) could actually be made ready for deployment during a very short warning time incoming NEO scenario, accounting for all logistical and programmatic factors.
- Precision guidance for ascent and the terminal phase of intercept. As noted in the results presented herein, the velocity of the interceptor relative to the NEO at intercept is on the order of 14 km/s, which poses a very challenging hypervelocity guidance, navigation, and control problem, especially when aiming at a relatively small NEO.
- Effects of realistic navigation and orbit determination errors (e.g., unavoidable errors in knowledge of the NEO's orbit and knowledge of the interceptor's state as a function of time).
- Assessment of NEO interception performance achieved when the interceptor begins in Earth orbit rather than on Earth's surface, and comparison to the results we have presented herein.
- Analysis of the minimum time required (starting from NEO threat detection/confirmation) to prepare an interceptor for launch via a spacecraft launch vehicle (instead of a silo-based booster), including interceptor vehicle development/preparation, testing, launch vehicle preparation and integration, etc. This analysis would be timely and interesting, because typical spacecraft development schedules require 4 to 6 years from the start of development until launch.

IX. Conclusions

In this paper we have examined suborbital intercept of small (∼50–150 m) NEOs with very short warning time. The ideal optimal trajectory calculation results presented herein may be thought of as a simple open-loop trajectory model for an NEO intercept mission design study. Within the assumptions and limitations of our analysis framework, we find that current silo-launched booster vehicles have sufficient burnout velocities to deliver a payload to intercept a NEO approaching

Earth impact, when the NEO is is very near Earth. Also, the performance of a modified ICBM could easily be improved by carrying a smaller payload for NEO interception and disruption than the larger warhead it was originally designed to carry. Such a system could serve as a last-minute defense tier. We again emphasize that if more warning time than several hours is provided (e.g., >1 week), then an interplanetary (i.e., far from Earth) intercept/fragmentation becomes feasible, but will require an interplanetary launch vehicle. While this paper is limited in scope and represents a preliminary examination of the problem, it lays the foundation for any further research and development that may be performed. Perhaps most importantly, the results in this paper demonstrate that, in principle, preparing for and executing Planetary Defense missions need not be prohibitively expensive (i.e., some existing hardware may be directly applicable).

When a hazardous NEO on a collision course with Earth is discovered we will not have the luxury of designing, testing, and refining our systems and plans. We will need to be fully prepared at that time to take effective action on relatively short notice with a high probability of succeeding on the first try, because we may not have a second chance. That level of adeptness and preparedness can only be achieved through proper design and testing of systems now, so that we are comfortable with carrying out planetary defense test and practice missions well before we need to deploy such a mission in response to an actual threat.

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References

[1] Wie, B., "Hypervelocity Nuclear Interceptors for Asteroid Disruption," *Acta Astronautica*, Vol. 90, 2013, pp. 146–155.

[2] Kaplinger, B., Wie, B., and Dearborn, D., "Nuclear Fragmentation/Dispersion Modeling and Simulation of Hazardous Near-Earth Objects," *Acta Astronautica*, Vol. 90, 2013, pp. 156–164.

[3] Pitz, A., Kaplinger, B., Vardaxis, G., Winkler, T., and Wie, B., "Conceptual Design of a Hypervelocity Asteroid Intercept Vehicle (HAIV) and its Flight Validation Mission," *Acta Astronautica*, Vol. 94, 2014, pp. 42–56.

[4] Barbee, B., Wie, B., Mark, S., and Getzandanner, K., "Conceptual Design of a Flight Validation Mission for a Hypervelocity Asteroid Intercept Vehicle," *Proceedings of the IAA Planetary Defense Conference 2013*, Flagstaff, AZ, April 15-19, 2013.

[5] Barbee, B. W., ed., "Target NEO: Open Global Community NEO Workshop Report," 2011, downloadable from URL `http://targetneo.jhuapl.edu`.

[6] Bate, R., Mueller, D., and White, J., *Fundamentals of Astrodynamics*, Dover Publications, 1971.

[7] Hanson, J. M., Shrader, M. W., and Cruze, C. A., "Ascent Guidance Comparisons," *The Journal of the Astronautical Sciences*, Vol. 43, No. 3, 1995, pp. 307–326, Also, NASA-TM-112493.

[8] Dukeman, G. A., "Atmospheric Ascent Guidance for Rocket-Powered Launch Vehicles," AIAA 2002-4559, August 2002.

[9] Picka, B. A. and Glenn, C. B., "Space Shuttle Ascent Flight Design Process Evolution and Lessons Learned," AIAA 2011-7243, 2011.

[10] Lu, P., Griffin, B., Dukeman, G., and Chavez, F., "Rapid Optimal Multiburn Ascent Planning and Guidance," *Journal of Guidance, Control, and Dynamics*, Vol. 31, No. 6, 2008, pp. 1656–1664.

[11] Chavez, F. and Lu, P., "Rapid Ascent Trajectory Planning and Closed-Loop Guidance for Responsive Launch," *Proceedings of the 7th Responsive Space Conference*, Los Angeles, CA, April 27–30, 2009.