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Master thesis report
Asteroids deflection using state of the arts European technologies

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Abstract: In public opinion, protection against asteroids impact has always been on the agenda of space engineering. Actually it started from 1994 when Shoemaker Levy stroke Jupiter. This protection works in two steps: detection of threat and deflection. Some space agencies and foundations monitor the sky and set up scenario. Although the sky is nowadays well monitored and mapped, there is no global plan nowadays against this threat. This paper focuses on the deflection step, and aims at forecasting which variables are involved and their consequences on the deflection mission. In fact the result depends on several factors, like the time before hazardous moment, the accuracy of detection tools, the choice of deflection method, but the most unpredictable are human factors. This study shows a strategy and so tries to give some new response parts to the global deflection problem.

Key-word: deflection, NEO, asteroid, low thrust, ion beam shepherd, gravity link.

Nomenclature:

- \( AU \): Astronomic Unit
- \( a \): semi major axis (AU, m or km)
- \( e \): eccentricity
- \( i \): inclination (degree)
- \( \omega \): Argument of periapsis
- \( \Omega, RAAN \): right ascending node
- \( M \): mean anomaly
- \( SC \): Spacecraft
- \( HET \): Hall Effect Thruster
- \( NEO \): Near Earth Object
- \( \mu \): standard gravitational parameter
- \( \rho \): density
- \( \Delta V \): velocity increment (m/s)
Introduction:

According to a majority of scientists’ opinion, dinosaurs’ extinction occurred after an asteroid impact [1]. This gives an idea of potential effect of asteroid impact between asteroid and Earth. Protecting mankind against such threat has always been on the agenda of space engineering, and climbed even higher since comet Shoemaker Levy stroke planet Jupiter in 1994.

Protection against asteroid is a two steps endeavor: first step is to detect and track this Potential Hazardous Asteroids (PHAs) while second step consists in chasing and deflecting those PHAs in case of danger. As of today the single deflection method experienced was kinematic impact, as performed by the Deep Space probe Tempel1 in 2005 [2], providing a huge amount of energy to a comet. A lot of studies have been carried out about deviation mission using kinematic impactor. This method lacks however accuracy in the velocity provided to the target.

Deflection of a PHA is often designed as a one-of-a-kind mission, requiring specific hardware and extraordinary efforts. While this can certainly work from an engineering standpoint, deciding and setting up such unique mission may be difficult, even in the case of proven threat. The aim of this study is to go the opposite way, by assessing what can be achieved using available or near-term available assets for deflecting PHA. Should deflections be within reach of more or less routine means, this would offer a cheap and easy way to implement mission profile against PHA.

With this goal in mind, this study aims at setting up a PHA deflection end-to-end scenario/mission using devices likely available in Europe by 2025, namely Ariane6 launcher teamed with Solar Electrical Propulsion (SEP) purpose space tug. Last but not least, one must keep in mind that the results shall be accurate and measurable, to avoid increasing accidentally the impact probability.

This study will split in three steps, a first step where the asteroids and Near Earth Objects (NEOs) are studied to understand the characteristic of this threat. A second part where deflection methods are studied and one is chosen because it is versatile, and a typical mission analysis is done. Lastly a mission example is performed, starting by presenting the tools used for what follows by mission example.
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# 2 WHAT DO WE KNOW ABOUT POTENTIAL HAZARDOUS ASTEROIDS

Protecting us against something requires to know a minimum of information about this threat. This section contains a state of the arts on objects in space that could be a potential threat for Earth. It shall help to understand the specificity of asteroids hazards, their destruction power and what information and incertitude mankind has.

## 2.1 WHAT IS A NEO?

Near-Earth Objects (NEOs) are comets and asteroids that have been nudged by the gravitational attraction of nearby planets into orbits which allows them to enter the Earth's neighborhood. It can include asteroids, meteorites and comets.

Anything smaller than ten meters could be called meteorites and aren’t hazardous enough to be the target of deflection mission. Near Earth Comets are further restricted to include only short period comets. The vast majority of NEOs are asteroids (NEAs). NEAs are divided into different groups summed up in the next table 1, according to their perihelion and aphelion distance (respectively q and Q) and their semi major axes (a). Hazardous Asteroids for Earth are Atiras and Atens ones. Figure 1 helps to visualize the main groups of NEOs, their orbits and the hazardous one.

<table>
<thead>
<tr>
<th>Group</th>
<th>Description</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>NECs</td>
<td>Near-Earth Comets</td>
<td>q&lt;1.3 AU, P&lt;200 years</td>
</tr>
<tr>
<td>NEAs</td>
<td>Near-Earth Asteroids</td>
<td>q&lt;1.3 AU</td>
</tr>
<tr>
<td>Atiras</td>
<td>NEAs whose orbits are contained entirely with the orbit of the Earth (named after asteroid 163693 Atira). Also call Inner Earth Objects (IEOs)</td>
<td>a&lt;1.0 AU, Q&lt;0.983 AU</td>
</tr>
<tr>
<td>Atens</td>
<td>Earth-crossing NEAs with semi-major axes smaller than Earth's (named after asteroid 2062 Aten)</td>
<td>a&lt;1.0 AU, Q&gt;0.983 AU</td>
</tr>
<tr>
<td>Apollos</td>
<td>Earth-crossing NEAs with semi-major axes larger than Earth's (named after asteroid 1862 Apollo)</td>
<td>a&gt;1.0 AU, q&lt;1.017 AU</td>
</tr>
<tr>
<td>Amors</td>
<td>Earth-approaching NEAs with orbits exterior to Earth's but interior to Mars’ (named after asteroid 1221 Amor)</td>
<td>a&gt;1.0 AU, 1.017&lt;q&lt;1.3 AU</td>
</tr>
<tr>
<td>PHAs</td>
<td>Potentially Hazardous Asteroids: NEAs whose Minimum Orbit Intersection Distance (MOID) with the Earth is 0.05 AU or less and whose absolute magnitude (H) (see Turino’s scale, 1.2) is 22.0 or brighter</td>
<td>MOID&lt;=0.05 AU, H&lt;=22.0</td>
</tr>
</tbody>
</table>

*Table 1 : NEOs group - [3]*)
The different “groups” or “types” shown here collect NEOs according to their orbit but it’s not the only way to sort NEOs. Nowadays classification refers to their composition instead of their orbit as shown in the following section.

2.2 NEOS CLASSIFICATIONS AND CHARACTERISTICS

NEOs classifications are elaborated based on two main characteristics:

- Their orbital trajectory, providing a measure of the likelihood of an impact,
- Their size and composition, providing an indication of the seriousness of the threat.

NEOs impact probabilities are hard to estimate, first because of poor trajectory measurement accuracy due to the tiny size of these objects, and second because of gravity perturbation affecting their evolution over time. So the “true” trajectories of NEOs aren’t really known but estimated and expressed statistically, as probability to be there. Impact probability corresponds to probability that NEOs trajectories (including their unknowns and scatterings) would cross Earth path at the right moment.

The present-day classification was initiated by Clark R. Chapman, David Morrison, and Ben Zellner in 1975 with three categories: C for dark carbonaceous objects, S for stony (silicaceous) objects and U for those that did not fit into either C or S. This classification has since been expanded and clarified. The two most used classifications are [4]:

- The Tholen Classification
- SMAAS Classification

Figure 1- Near Earth Asteroid orbit types [3]
In order to classify the NEOs, the only data available is often observations whose accuracy depends on the distance and objects shape/dimension, and albedo. For example, chemical composition and so density are determined with albedo (brightness under certain constrains).

Information on asteroids composition and NEOs is nowadays coming mostly from observation or sample fallen on earth after burning during atmospheric entry, and not from sample or data recorded on the objects. The consequence is that any knowledge on these objects we have is hypothetic, like density, structural and chemical composition. This information is useful to determine the NEOs mass which is required with relative velocity with Earth to compute the kinematics energy of the NEOs used in different scale as described in the next section.

NEOs attitude / rotating behavior are random (non-cooperating target) as their size and shape. These information could be critical for deflection mission, and is determined from observation if this one allows it. Last but not least this information doesn’t appear on any classification.

2.3 DIFFERENT SCALES

There are many scales to describe hazardous NEOs. The two mains are the Torino Scale and the Palermo Technical Impact Hazard Scale.

The Torino scale uses an integer scale from 0 to 10 and is defined only for potential impacts less than 100 years in the future. A 0 indicates an object has a negligibly small chance of collision with Earth. A 10 indicates that a collision is certain, and the impacting object is large enough to precipitate a global disaster. The Torino hazards impact scale takes into account the object’s estimate energy: size (density, volume) and relative speed to earth, as well as the probability that it will collide with Earth. Figure 2 shows the Torino scale. The value on Torino scales for a NEO isn’t constant and can be downgraded or upgraded with a new observation and so accurate knowledge of NEOs orbit and collision probability. Furthermore an object with multiple potential collisions on a set of dates, a Torino Scale value should be determined for each date. It may be convenient to summarize such an object by the greatest Torino Scale value within the set. [5] [6]
The Palermo Technical Impact Hazard Scale is a logarithmic scale used by astronomers to rate the potential hazard of impact of a NEO. It combines two types of data – probability of impact, and estimated kinematic yield—into a single "hazard" value. A rating of 0 means the hazard is as likely as the background hazard (defined as the average risk posed by objects of the same size or larger over the years until the date of the potential impact). A rating of +2 would indicate the hazard is 100 times more likely than a random background event. Scale values less than −2 reflect events for which there are no likely consequences, while Palermo Scale values between −2 and 0 indicate situations that merit careful monitoring. [7]

These different scales don’t show everything: for example on Torino’s scale, a NEOs with 100% collision probability and an energy estimated just under 1MT is set as 0. If this NEOs falls on a big city like New York, the aftermath will be dreadful. To give an idea, Hiroshima bomb released an energy about 15 kT only, and 1MT is 66 times more.
2.4 NEO’S STATISTIC

Some entries in the following charts and tables depend on a diameter that is roughly inferred from the object’s estimated absolute magnitude (H) and its assumed reflectivity, or albedo. [8]. NEO hunting truly started with the fear of disaster from NEO collision in the 90’. As a result the NEO discoveries really started in the beginning of 90’ and increased with the number and capacity of installation to track it as shown on Figure 3.

![Near-Earth Asteroid Discoveries](image)

*Figure 3 – Number of NEOs discover from 1994, all asteroids – [8]*

Newly discovered NEOs are small, while larger ones have already been cataloged back in the 2000ies as shown on figure 3 & figure 4.
With over 90% of the near-Earth objects larger than one kilometer already discovered (estimation), the NEO Program is now focusing on finding 90% of the NEO population larger than 140 meters. As shown in table 2, most of NEOs discovered are large ones, even if many of less than kilometer sized NEOs have been discovered, but it is still a fraction of all these “little” NEOs. These “little” NEOs are hard to find because detection instruments (telescope) can only spot them if they are close enough to the Earth, thus only during a very limited time. Moreover even when these “little” NEOs are there, their apparent magnitude is so tiny that most of them haven’t been detected yet.

<table>
<thead>
<tr>
<th>Size</th>
<th>Discovered</th>
<th>Estimated</th>
</tr>
</thead>
<tbody>
<tr>
<td>&lt; 30 m</td>
<td>1 500</td>
<td>&gt;8 000 000</td>
</tr>
<tr>
<td>30 - 100 m</td>
<td>2 500</td>
<td>1 000 000</td>
</tr>
<tr>
<td>100 – 300 m</td>
<td>2 700</td>
<td>100 000</td>
</tr>
<tr>
<td>300 – 1000 m</td>
<td>3 100</td>
<td>15 000</td>
</tr>
<tr>
<td>&lt; 1000 m</td>
<td>900</td>
<td>1000</td>
</tr>
</tbody>
</table>

Table 2 – Number of NEOs discovers and the number of NEOs estimated – reference: internal data from CNES

Last but not least, most of the kilometer sized NEOs are known, monitored and can be detected from far away. So nowadays the most hazardous ones are NEOs between 30 – 100 meters, because
their “little” size makes it hard to find except when they are close to Earth and when it is too late to prepare deflection mission or evacuate the city/region concerned. There are much more probabilities this event happens since there are more numerous than kilometers sized ones. An example is the NEO of 17 meters and 12 000 tons (estimation) fallen close to the city Tcheliabinsk in Russia in 2013. Fat Man, the Bomb dropped on Nagasaki in 1945 was around 22 kilotons, only 1/18ème of the NEO impacting near Tcheliabinsk which was estimated to 0.4 megaton. According to a B612 report [9], at least 26 explosions on Earth ranging in energy from 1-600 kilotons happen every year - all caused not by nuclear explosions, but rather by asteroid impacts and all are scored 0 on Torino scale.

2.5 NEO’S PROTECTION

As said previously, we can’t protect us from a threat that isn’t known. So protection passes through two actions: NEOs detection and registration (an NEO’s atlas) and if necessary a deflection mission. Multiple organizations are looking the sky to NEOs and setting an atlas. For example there are:

- B612 foundation
- NEO shield
- NASA
- Japan Spaceguard Association

2.6 SYNTHESIS & CONCLUSION

To sum up all this information, asteroids are sorted according to different classifications: one about their trajectories and one set up with their estimated composition. Especially for NEOs there are scales to identify which one is dangerous or not according to their size/mass and their impact probability. However these categories don’t show every important information such as their shape, their structural composition (rubble or not), their attitude, and their size/mass. Most of kilometer sized NEOs have been discovered nowadays and only smaller ones remain unidentified. These “small” NEOs stay dangerous if they fall on a densely populated area. If these ones remain unidentified it’s because they are too small to be seen from far away with our tools, and so can only be detected when they are close to Earth or when it’s too late.

What makes deflection missions hard to manage is the lack of information or its accuracy when we have some. The purpose of these missions is to reduce the impact probability. The mission and Space Craft (SC) design require to estimate the velocity increment needed, which depends on the NEO mass: but mass and accurate trajectory which are unknown. According to the deflection method chosen NEOs characteristics are of a great importance as well as shape, composition, rubble or not, and its attitude/rotating behavior, information which aren’t accurately known before SC reaches the NEO.
Due to all this unknown information, the deflection method which will be used will have to work in any possible situation. Trajectories and mass incertitude lead to give the NEO a variable $\Delta V$ function of the case and can require to study the NEO before deflection maneuvers.

Last but not least, to avoid a threat, the first step is to know that this threat exists, then all gates are opened. NEOs don’t escape the rule: plan and process to a deflection mission start with identifying the NEO, thus the first action to NEO’s protection remains identification and track.

### 3 NEO’S DEFLECTION, HOW IT DOES WORK

For many people deflecting an asteroid means modifying enough its trajectory so that the NEO doesn’t cross the Earth path anymore. This is possible but would require a huge $\Delta V$ except in some cases. This vision of deflection does not take into account that Earth and NEO are dynamical systems and evolve in a 4D dimension: collision could happen when NEOs crosses Earth path, but it is not enough: it should also do it at the right moment when the Earth is there. To help understanding the principle, let’s start by some explanation about motion of objects in space: two body problem in force free space. Then we will see what is the strategy used to deflect NEOs and some examples.

#### 3.1 SPACEFLIGHT DYNAMICS

The motion of every object in a force free space around a central body is a two body problem and could be described by the three Kepler’s laws which is now often enumerated this way [10]:

- The orbit of an object is an ellipse with the main body (local gravity pit) at one of the two foci.
- A line segment joining an object and the main body sweeps out equal areas during equal intervals of time
- The square of the orbital period of an object is proportional to the cube of the semi-major axis of its orbit.

This law works for a two body problem on a force free space. Actually this describes the main motion of objects, but this is not the true motion because Space isn’t a “force free space” and there are many perturbation sources. For example around the Earth the perturbations are atmospheric drag, J2 perturbation (non-homogenous Earth gravity field), Sun’s gravity, magnetic field, Sun’s radiation pressure, and so on.

Every orbit at an instant is described by its **position** $(x, y, z)$ and **velocity** $(v_x, v_y, v_z)$ in space, and so a set of six variables. This set of six variables makes the orbit hard to display for human mind.
So a set of 6 other variables (see figure 5) and called orbital parameters is commonly used to describe an orbit:

- The semi-major axis: \( a \) (m, km or AU)
- The orbit eccentricity: \( e \)
- The longitude of ascending node: \( \Omega \) (rad, or degree)
- The inclination: \( i \) (rad or degree)
- The Argument of periapsis: \( \omega \) (rad or degree)
- The last one gives the position at a given time, it could be a date of passage to the perigee or the true anomaly, eccentric anomaly or mean anomaly.

Another way to see this set is that the two first parameters \((a\text{ and }e)\) describe the shape and size of the orbit, the three next the orbit location in space \((\Omega, \ i\text{ and }\omega)\), and the last one the position of the object on the orbit at a given time.

This set of parameters may experience some troubles in some situation like circular \((e = 0)\) orbit.

Figure 5 - Orbital parameter, in the case when the plane of reference is also called orbital plane and reference direction is the Vernal equinox
and equatorial orbit ($i = 0$) where some parameters are not defined. When it happens some special set is used.

For further reading on this subject please see reference [11] & [12].

The following equations give useful information for the next section:

- Orbital period $T$ is given by the semi-major axis $a$ and the gravitational parameter $\mu$:
  \[
  T = 2\pi \sqrt{\frac{a^3}{\mu}}
  \]

- The instant velocity at a point of altitude $r$ of an orbit:
  \[
  V = \sqrt{\frac{2\mu}{r} - \frac{\mu}{a}}
  \]

  (The direction of this velocity is given by the cross product between the position vector and the eccentricity vector (normal to orbit plane)).

Then here are some useful notions about orbital mechanics to understand better the maneuver and space motion:

Two objects on the same orbit but at different positions at the same moment will never cross each other: to one orbit corresponds one period, and even if their relative speed is not null, it will vary in such way that both objects move periodically towards and away from each other. Should both objects cross each other, it would happen on different orbits. If one changes its orbit to reach the same as the other object when both are close, this maneuver is called a “rendezvous”. Space Maneuver can be classified in two categories: impulsive maneuver and low trust maneuver.

Impulsive maneuvers are well known and easy to compute and so allow to find the best maneuver combination to reach an orbit. It is a maneuver where the time to apply the $\Delta V$ is short comparing to orbit period and could be considered as instantaneous. It’s interesting to note that in a force free space, when an impulsive maneuver happens at a $(x,y,z)$ co-ordinate then modifying all orbital parameters to a new orbit, the object will pass again to this co-ordinate. In other words when a maneuver happens, the maneuver co-ordinate is included in the new orbit. That implies that if the new orbit has no common co-ordinate with the initial one, at least two impulsive maneuvers shall be done to reach the new orbit.

Low thrust trajectories point out maneuvers done with a low thrust propulsion device such as electrical propulsion. These maneuvers are characterized by a large period to apply the $\Delta V$ comparing to orbit period (sometimes hundreds, thousands of times this period). In my mind the “low thrust” designation is wrong: it should be called low acceleration. Acceleration is the true variable which distinguishes low thrust maneuver and impulsive maneuver, much better than “low thrust”. Nowadays some satellites weight several tons and others like “cubsat” or “nanosat” can be
kilogram-sized, a “low thrust” for one can be a very high thrust for the other category: the true variable is the thrust comparing to the weight.

Low thrust maneuver follows the same rules as impulsive one, except that if we use the thruster only when the maneuver efficiency is the best, the thruster will have to be used thousands of times only to operate one maneuver and this will take a lot of time. So a trade-off has to be done between the efficiency of the transfer and the time to operate it.

To conclude, an orbit is defined at a point by its co-ordinate and its velocity. Should this velocity change, the orbit would change as well (and so every point on this orbit path would, except the one when the change in velocity happens).

3.2 DEFLECTION STRATEGY

Deflecting an asteroid consists in adding a $\Delta V$ to its velocity which passes from $V_1$ to $V_2$ at a given moment or over time. Due to this velocity change the orbit of the NEO also changes. The aim of its orbit change is to reduce the probability that the NEO strikes Earth. To do this there are many strategies.

Let’s now talk about different ways to deflect an asteroid. The most common thought is to fully change the NEOs trajectory to a new one which never crosses Earth path anymore. So changing a NEO orbit from an Aten or Apollo orbit to an Atiras (IEO) or Amor orbit. This solution works but also often requires to change a lot the NEO velocity and so requires a lot of energy/momentum. For example a small NEO of 12000 tons, same as the one which fall near Tcheliabinsk. Add only 10 m/s to this NEO (which should not obviously be inadequate) will require a momentum of 120 MN.s. Assuming the mass of the NEO is much bigger than probe/rocket one. Considering last generation upper stage rocket engine like Vinci’s one with 180kN of thrust, and a propellant consumption of 39.5kg/s, 120 MN.s will require to fire this engine around 666 second and consume 26 tons of propellant. There are few launchers able to send such payload onto LEO, and even less to a NEO in heliocentric orbit.
Now let’s see the problem from another point a view, not just a spatial problem in 3D, but a 4D problem considering the time: the aim is to reduce the probability than NEO strikes the Earth at a given moment. The idea is to increase the NEO position relative to Earth, not to change its path. In other words the concept is to delay or ahead the NEO relative to the Earth, a “time deflection” rather than a “space deflection”.

In fact Earth orbital velocity is 30 km/s so if the NEO’s period could be delayed or advanced by 200 seconds, the distance between Earth and NEO possible position would be upgraded by approximately 6000 km.

As a NEO’s deviation is a combination of crossing date (time) and orbital parameter (distance), computing the deviation just by delaying the NEO, which requires to change its orbital element, is an approximation. As figure 6 shows a little ΔV means a little deviation but also as figure 7 shows, delays and advances the NEO’s Earth path crossing date. It also shows that delaying or advancing the NEO requires to change its orbital parameters and so will change a little its path. Computation shows that the relative distance between Earth and the NEO increases much more due to the “time deflection” rather than “space deflection” in most of cases.
Advantage of this concept is the low \( \Delta V \) required to the deflection comparing to the first method. The second advantage is the time dependence: if the deflection distance is 30 000 km, it requires to modify the period by 1000 seconds one year before the hazardous moment. Two years before only 500 seconds are required; three years before: 333 seconds and so on. The earlier the deflection maneuver happens, the lower is the period modification needed and so the \( \Delta V \) and cost.

3.3 MODEL

Computing the \( \Delta V \) requires to upgrade the minimum distance between Earth and NEO trajectory and needs a lot of computation and to compute all the solutions from a deflection date \( t_f \) till the crossing date \( t_o \) needs too much computations… So another way to compute this \( \Delta V \) is needed.

As said previously the distance deviation \( X_d \) (m) could be approximated by a delay or an advance \( S_d \) (sec), see equation 3.1:

\[
S_d = \frac{X_d}{30\,000}
\]  

(3.1)
Equation 3.2 gives the periods $T$ and equation 3.3 $N_{rev}$ number of revolution between $t_f$ and $t_0$ (with $\Delta t$ number of second between $t_f$ and $t_0$):

$$T = 2\pi \times \sqrt{\frac{a^3}{\mu}} \quad (3.2)$$

$$N_{rev} = \frac{\Delta t}{T} \quad (3.3)$$

By the way the new period wanted is given by equation 3.4:

$$T_2 = T_1 \pm \frac{S_d}{N_{rev}} \quad (3.4) *$$

*The sign depends if the aim is to delay or advance the NEO.

And finally the new semi-major axis is $a_2$:

$$a_2 = \left( \frac{\sqrt{\mu \cdot T_2}}{2\pi} \right)^2 \quad (3.5)$$

As shown section 3.1 the period is only a function of semi-major axis $a$, so modifying the period is equivalent to modifying $a$. The total energy $\varepsilon$ of a body is given by equation 3.6 (in any case) and in the case of an elliptic trajectory it can be expressed by equation 3.7:

$$\varepsilon = \frac{V^2}{2} - \frac{\mu}{r} \quad (3.6)$$

$$\varepsilon_{ellipse} = -\frac{\mu}{2a} \quad (3.7)$$

Using equations 3.6 and 3.7, will give equation 3.8 which links $a$, $r$, $\mu$ and $V$:

$$\frac{1}{a} = \frac{2}{r} \frac{V^2}{\mu} \quad (3.8)$$

The $r$ and $\mu$ are values that are constant at an instant $t$. Equations 3.8 show that “$a$ ” and the norm of the velocity vector are linked for a given radius and $\mu$. Furthermore this equation shows that for a given $\Delta V$, the change in efficiency of the semi-major axis is maximum if this $\Delta V$ is applied in the same direction as the velocity, which changes a maximum the velocity vector norm.

So the new velocity vector to reach the new period is:

$$V_2 = \sqrt{\mu} \sqrt{\frac{2}{r} \frac{1}{a_2}} = \sqrt{\mu} \sqrt{\frac{2}{r} \left( \frac{2\pi}{\sqrt{\mu T_2}} \right)^2} \quad (3.9)$$

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So the $\Delta V$ requires $V_2 - V_1$ with $V_1$ the initial velocity of the NEO before deviation. About the direction of $V_2$, the velocity change is to modify the time before crossing path. As the strategy is to change the semi-major axis, by changing the velocity, the velocity change shall occur in the velocity direction or in the opposite direction.

For example, figure 8 shows $\Delta V$ required to deflect by Earth-moon (384 399 km) distance a NEO from Earth path. If deflection maneuver occurs 20 years before hazardous moment, it requires less than half millimeter by second and ten years before around 1 mm/s. Of course to avoid a collision it isn’t necessary to add such a distance and lowering $\Delta V$ works too. Furthermore this method doesn’t take into account the spatial deflection of trajectory, just the delay.

![Figure 8 - $\Delta V$ required function of deflection moment for a 384 399km deflection](image)

$\Delta V$ require according to the impact date to deflect of one Earth-Moon distance the NEO from Earth path
3.4 DISTANCE DEFLECTION CHOICE

As the Earth radius is around 6370 km, why should we consider a deflection distance of 384 400 km? As said previously the true path of NEOs aren’t known, and this leads to a volume of possible position at a given moment. This volume could be much bigger than the Earth radius. Considering that this distance is set with the objective to reduce the collision probability passing from a probability $P1$ to a $P2$ probability lower or null, a bigger volume leads to a bigger deflection distance than a small one.

Considering the following example:

- The aim is to set to 0 the impact probability.
- 30 years before hazardous moment, the volume is assumed to be a sphere of radius 200 000km. The Earth is fully included in this sphere.
- Only 5 years before hazardous moment, accurate measure of NEO trajectory reduces this volume to a sphere of 5000 km centered on the Earth.
- In both cases it’s assumed that NEO has equal chances to be anywhere in the volume.
- In other words and to simplify, the problem is not considered as dynamic but as static: the collision probability is considered at a given moment. And not for any other one. This means that it doesn’t take into account the direction of deflection which should actually be importance.

The impact probabilities are low in the first case: less than 0.004%. In order to avoid a collision the NEO must be deflected (as well as the volume where it may be located) of a $D1$ distance which depends on the position of the Earth in the volume of potential position: if the position is close to the center, then NEO will have to be deflected by at least 203 000km and if the position is close to the bound of this volume, then deflecting by 4000km will be enough.

To sum up, probabilities are low because of uncertainties and lead to a huge volume of possible locations. The deflection distance required then depends on the location of the Earth in this volume.

In the second case, the potential position of the NEO (volume) is fully included in/on the Earth, and so collision probability is 100%. But to avoid the collision the deflection distance is also smaller: 7000 km lead the NEO to pass close to the Earth but avoid it.

Are 7000 km enough? Should a safety margin be taken? To these questions, opinions diverge and everyone has his own idea. Some people think that NEOs should be deflected just enough to avoid the Earth. Other people think than an Earth-Moon distance is a minimum be sure… There is no scientific answer to this question.

3.5 DISCUSSION
Last section ends on a problem on the deflection distance to consider. The quick answer could be “deflect it as much as we can, to be sure”. But we must also keep in mind that once the NEO has been deflected, its trajectory changes together with the perturbations along its path until it comes close to Earth and these new perturbations could put again the NEO on hazardous trajectory for the Earth. Obviously this scenario has very low probability to happen but shows than the deflection maneuver could not be done randomly and must be computed accurately. This also shows some limits to this deflection model. As a lot of information are missing, this reasoning leads to complicate computation with lots of assumptions or cases… which has no sense. One solution is to take this decision later once having studied more the threat: to deflect the NEO requires to change its velocity/trajectory and this should be done by a probe that must reach the NEO. Nothing prohibits, once the NEO has been reached by the probe, from first studying the NEO and its characteristics before starting the deflection mission. When all useful information are collected then a decision can be taken and the mission can start in good conditions.
4 DEFLECTION ANALYSIS

Various deflection scenarios to deflect NEO have been studied by different organizations but the single example of “celestial body” deflection done until today is the mission Deep Space where an impactor gave a big impulse to a comet in order to eject matter and study the comet inner material: the deflection wasn’t the goal of the mission, just a consequence of the method used.

Lots of studies about asteroid deflection ways have been performed and can be classified into various categories according to deflection kinematic (fast or slow), energies sources and so on. For example figure 9 shows the deflection methods considered by NEOShield organization.

![Deflection methods diagram](image)

*Figure 9 - Deflection methods - ref [13]*

4.1 DEFLECTION SPECIFICATIONS

As deflection mission could be performed in emergency to deal a threat with random characteristics. The deflection method retained for the study shall be universal, i.e. work on all NEOs whatever its structure (rubble or not), attitude (non-cooperative target), chemical properties, and so on. Furthermore it should be able to be launched without social problems, for example put in orbit radioactive material / bomb; if possible. Last but not least the method should be able to
censor accurately the new trajectory and deflection effect to avoid all risk: the aim is to reduce collision probability and not to increase it accidentally. The selection criteria are:

- Non-cooperative target capacity (C1)
- Works on rubble NEOs (C2)
- Socially correct (C3)
- Accurate method (C4)
- Able to measure results (C5)
- Technology maturity (C6)

Some problems could be common to all deflection methods: for example power sources could be nuclear or solar. If it is not linked to the method these kinds of problems are not taken into account in the criteria evaluation.

4.2 DIFFERENT DEFLECTION SOLUTIONS

The following sub-sections present different methods studied to deflect NEOs. In order to compare easily each method, a standard table summarizes each option through the criteria presented in section 4.1 and a quotation (4 quote possible: --, -, + and ++). Moreover a sum-up table section 4.3 gives us an overview.

4.2.1 SPACE TUG

The space tug mitigation methods consist in pushing directly the NEOs to change its velocity. To do this the tug is linked to the NEO and uses its engines. This implies that the tug could hang on to the NEO, whatever its attitude is; and then could push in the right direction (either at the right moment or if not, could first modify the NEO attitude and then push). This could be a big problem if using electrical propulsion for these maneuvers.

<table>
<thead>
<tr>
<th>Criteria</th>
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<tr>
<td>Non-cooperative target capacity</td>
<td>-</td>
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<tr>
<td>Work on every NEOs (rubble etc.)</td>
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<tr>
<td>Accurate method</td>
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<tr>
<td>Able to measure results</td>
<td>++</td>
</tr>
<tr>
<td>Technology maturity</td>
<td>+</td>
</tr>
</tbody>
</table>
4.2.2 KINEMATIC IMPACTOR

The principle of the kinematic impactor mitigation method is to deflect the NEO following an impact from an impactor SC. A momentum transfer happens from the SC to the NEO when the SC hits the NEO at high velocity. Momentum transfer is function of the mass of the SC and its relative velocity to the NEO, causing a velocity change on the NEO according to its proper mass, and so modifies its trajectory. That means the velocity change is also function of the direction and position where impact occurs and this has to be controlled accurately to not convert a part of the momentum in spin rate instead of velocity change or modify the trajectory in a wrong way.

The advantages of this method are multiple: first it’s easy to compute the momentum transfer with this equation: \( M_1 V_1 = M_2 V_2 \) with \( M_1 \) the SC mass, \( V_1 \) the relative speed between SC and NEO, \( M_2 \) the NEO mass and \( V_2 \) the velocity communicated to the NEO. The second advantage is the huge momentum transfer: with 10 km/s in relative speed, 100kg give \( 10^6 \) Ns or could give to a 100 000 tons target a velocity change of 0.01 m/s which is enough to deflect it. The need of relative velocity implies that being on the same orbit and making a rendezvous are not sufficient. This method is one of the most efficient in terms of cost especially when NEOs pass near the Earth: the SC doesn’t require to escape the earth and the relative velocity is easily reached thanks to relative velocity between NEO and earth…

Furthermore this method has been already tested on a comet during Deep Space mission for another purpose.

The main drawbacks are that the method isn’t that accurate even if formula is easy: in fact it is necessary to know accurately the NEO mass that isn’t known without a study with an orbiter. Moreover this method transfers a huge momentum in a very short time and so could split the NEO or go through (for rubble or spongy ones) and not deflect it as expected. Some NEOs could be hollow and simulations show that in this case the impactor could just go through without any momentum transfer… Another source of inaccuracy is that some matter could be ejected during the impact and this isn’t taken into account in the formula. To measure the result accurately if deflection happens in deep space, an orbiter should remain. The last problem consists in setting the impact:

The main difficulties of this method are to be able to impact accurately the NEO and at a high speed. In fact rendezvous is controlled in relative speed or position but controlling both is complicated especially after a long travel, without any second chance. To give an idea the kinematic impact of Deep Space mission happens at 10 km/s. If the target is a kilometer sized NEO with spherical shape, the relative speed is also 10 km/s, and the target is the center of the NEO. That means the error margin is more or less 500m or the NEO is misfired. If we consider a 1D problem to make it easy: at 10km/s, 500m accuracy corresponds to 0.05 second. In other words a necessary condition is to be accurate at less than 0.05 second on the position or the mission fails, and it’s a necessary condition: it’s not sufficient. Keeping such an accuracy after a long travel is hard and technologies are not really perfected nowadays.

A way to improve the accuracy and limit the instant momentum transfer is to send a SC with multiple tiny impactors and send it one by one after a study of the NEO (close orbit).
### 4.2.3 EXPLOSIVES OR NUCLEAR BLAST

Although this method works well in Hollywood’s studio, its performance in true situation has never been tested and the results aren’t easily predictable:

The deflection process depends on the explosive used: pressure wave with conventional explosive, pressure wave and intense X ray emission with nuclear bomb. In the case of nuclear bomb it is hard to determine because such a bomb doesn’t work in the same way whether it’s in atmosphere or in vacuum. Moreover the results are also function of where the blast happens: with the bomb buried, close to the NEO or in a far field. In some case the momentum transfer could be improved by ejection or sublimation of matter from NEO surface. The biggest problem would happen in the case of rubble NEO which would probably be split off.

To be accurate the shape, dimension, mass, (and albedo to know comportment of NEO to ray emission) have to be known to accurately compute the effects of the blast on the NEO, which aren’t known without study. And it’s necessary to have an orbiter which is not destroyed to be able to measure results.

Conventional explosive could be combined to kinematic impactor to improve its efficiency: blast on impact or just after.

One big advantage of this solution is the huge amount of energy carried by nuclear bomb. This bomb technology is well controlled by some nations and lightweight. That could be an efficient safety plan if first solution fails.

<table>
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<tr>
<th>Criteria</th>
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<tbody>
<tr>
<td>Non-cooperative target capacity</td>
<td>++</td>
</tr>
<tr>
<td>Work on every NEOs (rubble etc.)</td>
<td>- -</td>
</tr>
<tr>
<td>Politically correct</td>
<td>++</td>
</tr>
<tr>
<td>Accurate method</td>
<td>-</td>
</tr>
<tr>
<td>Able to measure results</td>
<td>-</td>
</tr>
<tr>
<td>Technology maturity</td>
<td>+</td>
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</tbody>
</table>
4.2.4 MASS DRIVER

This method consists in changing the NEO’s velocity using the own NEO’s matter: Using some probe stuck to the NEO, and eject matter (solid state, liquid or gas, doesn’t matter) with a known velocity and direction to change its momentum and its velocity.

One of the biggest problems with this method is that the probe shall be able to reach the NEO, land on it, stay stuck on it, and has tool to cut/drill matter and after eject it in an accurate direction. All these functions needed in a single probe means a huge payload, plus the power generator for all these systems. Furthermore all these systems are hard to do technically and haven’t been tested yet.

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<tr>
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<tr>
<td>Accurate method</td>
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<tr>
<td>Able to measure results</td>
<td>+ +</td>
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<tr>
<td>Technology maturity</td>
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4.2.5 LASER / FOCUS SOLAR

This method consists in using a power beam to sublimate matter on the NEO’s surface and create little thrust by gas ejection as a rocket engine does. The power needed to make sublimate matter depends a lot on the matter and on the NEO’s characteristics which are not known. The beam can be created by a laser or optical system focusing sun ray. This method doesn’t need any contact with the NEO and doesn’t need matter except to control the attitude but could require focusing a small area during a long time.

However these methods have their drawbacks: both need optical instruments (length and so on) that can be polluted by dust and deposition and affect the yield. In the laser case it requires a lot of power and high power laser is nowadays heavy even the one working by impulsion. For the focus solar method it requires big length which experiences sun orientation constrains to works. Or NEO close field can be pollute by dust, thruster plume or and micro meteorites that can damage optics.

Due to this constrains this technology can’t be considered but could be in the future.

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<td>Able to measure results</td>
<td>+ +</td>
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<tr>
<td>Technology maturity</td>
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</tbody>
</table>
4.2.6 YARKOVSKY EFFECT

The Yarkovsky effect is a force coming from an anisotropic emission of thermal photon which carry momentum: Space objects orbiting the Sun receive moment from photon (mostly from Sun) and their surfaces exposed are heated. Due to this heat, these objects will emit thermal photons too, but not instantaneously, and if objects spin, the emission direction will not be the same as the receive direction. The results is a small force which depend of the object spinning, shape and distance from sun.

Using the Yarkovsky effect to deflect a NEO consists in changing its characteristics to modify its force over a long time. There are several ways to do it: paint the NEO to change its radiation properties, use a device to focus more sun light on the NEO and increase the force or change its direction, or on the contrary use device to hide the sun light to the NEO (make a shadow); and so on.

This method has never been tested, the results isn’t easily predictable, and the implementation uncertain. The force generated is small but works (this force have a significant influence on NEOs trajectories). Last it could be a passive method requiring no fuel once implemented (for example painting the NEO).

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<tr>
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<tr>
<td>Technology maturity</td>
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</table>
4.2.7 GRAVITY TRACTOR

This method consists in putting a SC close to the NEO and letting the gravitational force between the two bodies deflect the NEO. In attempt to avoid the SC crashes on the NEO, some thrusters must keep the distance constant between the SC and the target. Furthermore the plume of the thruster could create on the target a repulsion force and so cancel the gravity attraction force. So the thrusters have to be oriented in such a way that their plumes do not perturb the process. Last but not least the acceleration created does not depend on the target mass but only on the SC mass and the distance between both mass center. However a heavier NEO implies also a bigger NEO and so a longer safe distance between the SC and the NEO and so a lower force. (see figure 10)

As there is no contact it works on any NEO even the one with high spin rate or/and rubble and so breakable. The low deflection speed allows an accurate deflection and monitoring.

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<tr>
<td>Accurate method</td>
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<tr>
<td>Able to measure results</td>
<td>++</td>
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<tr>
<td>Technology maturity</td>
<td>+</td>
</tr>
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</table>
4.2.8 ION BEAM SHEPHERD

Sometimes this method is also called electrostatic pushing. It consists in deflecting the target indirectly by using the plume of the thruster to create a force on this one. There is no contact between the SC and the target but to be efficient the plume goes on the target and so must be close enough.

There are also some drawbacks: first, another engine is required to compensate the thrust of the one used for deflection, and this means twice as much propellant consumption. Second if low thrust propulsion is used for deflection, with the advantage of propellant efficiency, it also means low thrust and longtime deflection and so the effect of gravity could become not negligible compared to the force generated.

However this method allows a great accuracy of the deflection, to study the NEO and an accurate monitoring of the maneuver and of the new trajectory. Moreover the entire components required for it (thruster and so on) are space proven.

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<td>Non-cooperative target capacity</td>
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<td>Politically correct</td>
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<tr>
<td>Accurate method</td>
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<td>Able to measure results</td>
<td>++</td>
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<tr>
<td>Technology maturity</td>
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</table>

4.3 DEFLECTION METHOD CHOICE

Table 3 resumes the different choices

*Table 3- Different deflection options*

<table>
<thead>
<tr>
<th>Criteria</th>
<th>Space tug</th>
<th>kinematic impactor</th>
<th>explosive</th>
<th>Mass driver</th>
<th>Laser Focus solar</th>
<th>Yarkovsky effect</th>
<th>Gravity tractor</th>
<th>Ion beam shepherd</th>
</tr>
</thead>
<tbody>
<tr>
<td>C1</td>
<td>-</td>
<td>++</td>
<td>++</td>
<td>-</td>
<td>++</td>
<td>++</td>
<td>++</td>
<td>++</td>
</tr>
<tr>
<td>C2</td>
<td>+</td>
<td>-</td>
<td>-</td>
<td>+</td>
<td>++</td>
<td>++</td>
<td>+</td>
<td>++</td>
</tr>
<tr>
<td>C3</td>
<td>++</td>
<td>++</td>
<td>-</td>
<td>++</td>
<td>++</td>
<td>++</td>
<td>++</td>
<td>++</td>
</tr>
<tr>
<td>C4</td>
<td>++</td>
<td>-</td>
<td>-</td>
<td>+</td>
<td>++</td>
<td>+</td>
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<td>++</td>
</tr>
<tr>
<td>C5</td>
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<td>-</td>
<td>-</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>+</td>
<td>++</td>
</tr>
<tr>
<td>C6</td>
<td>+</td>
<td>++</td>
<td>+</td>
<td>-</td>
<td>-</td>
<td>-</td>
<td>+</td>
<td>+</td>
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</table>
Considering constrain for the choice, only two methods remain: the gravitational link and ion beam shepherd. A deeper analysis on their advantage/drawback/unknown and efficiency is required.

In order to compare both methods, a type NEO is parameter with an ellipsoidal shape where the bigger dimension is \( R_1 \), and the two other dimensions \( R_2 \) and \( R_3 \) are assumed to be equal. These three dimensions correspond to the three semi principal axes of the ellipsoid. The case where NEO is a Sphere corresponds to \( R_1 = R_2 = R_3 \). Efficiency of the deflection method changes according to the distance where deflection maneuver happens. This distance is set to be 1.5 this bigger distance to keep safe the SC because the NEOs attitude is random and the SC should not collide the NEO.

Comparison will be done using Apophis example: Apophis is one of the NEOs which has been listed on 4 on Torino’s scale before being decreased to 1 and then 0, its mass is estimated to \( 4.2 \times 10^{10} \) kg. Latest measure estimated Apophis bigger distance to 325 meters and its density to 2.6 tons/m\(^3\). Assuming that the two other semi principal axes are equal, a computation of the NEO volume, considering its density and bigger distance set these principal axes to 109 meters.

4.3.1 GRAVITATIONAL LINK

Figure 11 shows a simpler model of the method: a NEO of mass \( M \) and a SC of mass \( m \) separated by a distance \( d \), the thrusters are oriented in such a way that plumes don’t affect the NEO.

![Figure 11 - Model for gravity tractor method](image.png)
Attraction gravitational force \( F_g \) between two bodies is given by equation 4.1:

\[
F_g = \frac{G \cdot M \cdot m}{d^2}
\]  

(4.1)

Considering a constant mass for the SC, the two variables in this equation are the NEO mass \( M \) and the distance between the two centers of mass \( d \). As said previously this distance is set to 1.5 time the larger size \( R_l \) and \( V \) is the volume which is \((4/3) \cdot \pi \cdot R_1 \cdot R_2 \cdot R_3 \) for ellipsoid one (and \((4/3) \cdot \pi \cdot R_s^3 \) for spherical one, this case is just a special case of ellipsoid one). So the force equation (4.1) could be rewritten as equation (4.2):

\[
F_g = \frac{G \cdot 4 \cdot \pi \cdot R_1 \cdot R_2 \cdot R_3 \cdot \rho \cdot m}{3 \cdot 2.25 \cdot R_1^2} \quad \text{and for Apophis:} \quad \frac{G \cdot 4 \cdot \pi \cdot 0.112 \cdot R_1 \cdot \rho \cdot m}{3 \cdot 2.25}
\]  

(4.2)

(Considering that \( R_2 = R_3 = 0.3353 \cdot R_1 \) for the second equation (Apophis)). If \( R_2 \) and \( R_3 \) are written according to \( R_1 \) as in the second equation, it appears than the force variation follows the variable \( R_1 \) and a coefficient defined by \( R_2 \) and \( R_3 \). Last this method is more efficient on NEOs with higher density. Figure 12 shows the influence of the density on the attraction force.

*Figure 12 - \( F_g \) according to the density and distance between a 2000 kg SC and NEO gravity center for \( R_2 = R_3 = 0.5 R_1 \).*
Another way to see the unique advantage of this method is to express the results not as a force but as an acceleration:

\[ a \cdot M = \sum F \quad (4.3) \]

The thrusters of the SC cancel the force created by NEO on the SC, in order not to project the beam on the NEO and so create a perturbation force. So only the NEO moves under the action of the gravitational force. So the acceleration due to gravity tractor is expressed from equation 4.1 and 4.3 and is given by:

\[ a = \frac{G \cdot m}{d^2} \quad (4.4) \]

This equation confirms Galileo’s observation that bodies of different masses fall to Earth with the same acceleration: Here the bodies are the NEOs and Earth is the SC. So the acceleration only depends on the SC mass and distance but not on the NEOs mass. Nevertheless a heavier NEO also means a bigger NEO and so a bigger safety distance. As shown on figure 13, acceleration decreases when distance increases. Nevertheless the acceleration doesn’t depend on NEO mass.
Lastly, this method requires a mass that should be big enough. It could be a dead mass (lead or other metal), useful mass (additional propellant), or another solution if this deflection method misses (bomb). Although this method doesn’t consume propellant to create thrust it always requires to cancel the force generated by gravitation interaction. This should be done without blowing the NEO with the thruster beam in order not to create a repulsion force canceling the gravitational force. So the orientation and efficiency of the thruster (and so propellant efficiency) are defined by the thruster characteristics (divergence of the beam and far field beam dispersion) and the distance between the thrusters and the NEO surface.

Example with Apophis:

The distance maneuver is set up using the Apophis bigger dimension and a certain safety coefficient. Assumption about Apophis, ΔV required, deflection parameters and SC mass as the results is summed up in table 4.

Figure 13 - Acceleration generated by gravity link method for a NEO configuration where R2=R3=0.5R1 according to the distance between gravity center and a 2 tons spacecraft
Table 4 - Apophis deflection using gravity link method

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<table>
<thead>
<tr>
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</thead>
<tbody>
<tr>
<td>Apophis mass (kg)</td>
<td>4.2.10^{10}</td>
</tr>
<tr>
<td>Apophis bigger dimension (m)</td>
<td>325</td>
</tr>
<tr>
<td>SC mass (kg)</td>
<td>2000</td>
</tr>
<tr>
<td>ΔV required over 18.5 years (m/s)</td>
<td>2.10^{-6}</td>
</tr>
<tr>
<td>Safety coefficient</td>
<td>1.5</td>
</tr>
<tr>
<td>Maneuver distance (m)</td>
<td>487.5</td>
</tr>
<tr>
<td>Attractive force</td>
<td>10^{-6}</td>
</tr>
<tr>
<td>Acceleration</td>
<td>5.6.10^{-13}</td>
</tr>
<tr>
<td>Maneuver time (days)</td>
<td>41.2</td>
</tr>
</tbody>
</table>

Increasing just a little bit the safety coefficient leads to double the time maneuver. This shows how this method highly depends on the distance maneuver. But this reduces this distance and exposes the SC to a risk: an emergency maneuver could be needed and blow the NEO with thruster and so disturb the maneuver.

4.3.2 ION BEAM SHEPHERD

Figure 14 shows a simple representation of Ion Beam Shepherd method. Two thrusters work together creating two pretty similar thrusts $F_B$ & $F_M$ in opposite directions. Resulting force acting on the SC is $F_s$, used to keep the SC position in the right deflection configuration. One of the thruster beams is oriented to an object to deflect, creating pressure repartition on object surfaces. This pressure equivalent to thrust $F_D$ is equal to a fraction to $F_B$ and a torque depending on the mass center position to the force acting point. The difference between $F_D$ and $F_B$ corresponds to particles missing the target and can be expressed as efficiency term $\eta_B$ as shown figure14. There are several ways to consider the force origin: pressure difference over the NEO surface, or considering the NEO as an object in a fluid motion and so it’s a drag force or a direct momentum transfer from the impinging particle.

![Figure 14 – Model for Ion Beam Shepherd deflection method described previously and applied to space garbage [14]](image-url)
The fraction of $F_B$ transfer to the object depends on the distance between the object, its shape and beam divergence, and interaction between ion particles and the object (momentum transfer or converted to heat). Figure 15 shows more details on interaction between beam and target. The thrust can be modeled through multiple ways: the NEO is an object in a particle/low density flow and there is a drag, or a pressure on the face illuminates by thruster and the vacuum on the other face, or just a momentum transfer. These different ways to see the energy transfer lead to different means of computing the thrust created. The technique I focused on for what follows is the momentum transfer.

Figure 15 - Ion beam shepherd method applied to NEO.

In order to have a good estimation of momentum transfer between the beam and the NEO following assumption will be assumed:

- The momentum of impinging particle is fully transferred to the NEO as a momentum and not converted to heat or so.
- The SC is at a distance from the NEO where 60% of the energetic part of the beam§ (45° half angle for HET, [15] [16] [17]) is impinging the NEO whatever its attitude/orientation over time (mean values). This coefficient is called $\eta_i$.
- The NEO isn’t electrically charged and so doesn’t affect the beam/charged particle.
• The NEO surface is hard and doesn’t damp: the return back to velocity is almost equal in norm to impinging velocity.

• High energy particles have a much bigger axial velocity than radial ones, and their speed is considered as fully axial and at an infinite distance from thrusters, all particles go in the same direction. Another way to see it, is that the plume have a small divergence angle far from thruster instead of a 45°.

Then let’s have a look on what happens when a particle hits the NEO surface. Figure 16 shows a particle path when hitting the NEO. Surface is assumed to be hard and the collision fully elastic. So impinging particle rebounds on the surface without losing energy. The momentum transfer operates in two times: first incoming particle transfers all its momentum to the NEO. Second particle rebounds and is expulsed to space. As the particle was emitted by the NEO in another direction than the incoming one.

Final momentum transferred by one particle is $2.m_p.V_p.cos (\alpha)$, with $m_p$ and $V_p$ respectively particle mass and velocity and $\alpha$ impinging angle relative to impinging particle/surface direction. To find the momentum transfer all over the NEO, these results have to be integrated all over the NEO’s surface taking into account the direction of the surface’s normal. As NEO’s shape isn’t known, we assume that the NEO’s projected surface over a full revolution/spin is equivalent to a spherical surface of radius $R_m$ (mean value of $R1$, $R2$ and $R3$) to compute the loss due to the normal direction. As said previously, it is considered that all particles have the same direction: thruster axial direction. Next step is to integrate the particles flow equivalent to $2.m_p.V_p.cos (\alpha)$ all over the sphere surface from 0 to $\pi$/2 with $dS=2.r.sin(\alpha).d\alpha$ ($Mmt$ is the momentum quantity per projected surface unit hitting the NEO). The momentum quantity transferred to the NEO is (without considering the particles parts that miss the NEO):

$$Mt = \int_{0}^{\pi/2} 2.\pi.2.Mmt.\cos(\theta).\sin(\theta).r^2.d\theta = 2.\pi.r^2.Mmt$$

(4.5)

The projected surface of a sphere on a plane is $\pi.r^2$, and $\pi.r^2.Mmt$ is the total momentum transferred from thrusters to the NEO without considering the part of the beam missing the target and other force. The part of ions missing is added (loss term) and the gravitational force equation 4.5 becomes:
\[ F_p = 2. \frac{\partial M F t}{\partial t} \cdot \eta i - G \cdot \frac{m}{d^2} = 2. F_p \cdot \eta i - G \cdot \frac{m. M}{d^2} \] (4.6)

With \( F_t \) the thruster force, \( F_p \) the force acting on the NEO and \( m \) and \( M \) respectively the SC and NEO mass. The loss term \( \eta_i \) depends on NEO’s shape and rotation and beam divergence. Divergence half angle for electrical propulsion is around 45° for HET and 5-10° for ion engine. The most popular and developed technology is nowadays the HET. Furthermore it has the best perspective for high power engine. Due to this only the HET will be for the rest of the study.

The beam spreads in \( 1/d^2 \), where \( d \) isn’t well defined: it varies with the shape and the NEO attitude: although NEO projected surface changes over time, it’s possible to define a mean surface as it is a long time maneuver. The NEO is modeled as an ellipsoid, as shows figure17. This ellipsoid is parametered by its three semi principal axes of length \( a, b, c \); and it corresponds to the semi major and minor axis of three ellipses in the ellipsoid. The mean projected surface \( S_p \) is defined as the mean values of the surface of these three ellipses: \( S_p = (S1 + S2 + S3)/3 \), so the three surfaces are considered to be equally exposed. This representation is below reality: In fact considering a homogenous density NEO, it would spin around its major inertial axis which is the one with the lower length and so the two most exposed surfaces would be the two bigger.

Considering our ellipsoid NEO defined in section 4.3.1:

\[ S_p = \pi. \frac{r1^2 + r1.r2 + r1.r3}{3} = \pi. \frac{r1^2 + 2.r1.r2}{3} \] (considering \( r2 = r3 \)) (4.7)

For the surface beam it changes according to the distance between NEO and SC and the half divergence angle: considering a circular beam area, that momentum is homogenously spread over the surface and the plume doesn’t expand over the cone defined by divergence angle. The beam surface \( S_b \) is so defined by \( S_b = \pi. r^2 = \pi. d^2. tan^2(\alpha) \), with \( d \) the distance between NEO and SC and \( \alpha \), the half divergence angle.

Thus the loss term \( \eta_i \) is defined by \( \eta_i = S_p/S_b \). Considering that \( d \) is 1.5 the biggest distance, \( c \) on the example figure 13, this equation becomes:
\[ \eta_i = \frac{\pi (r^2 + 2. r_1 r_2)}{3. \pi. 2.25. r_1^2. \tan^2(\alpha)} = \frac{(r^2 + 2. r_1 r_2)}{6.75. r_1^2. \tan^2(\alpha)} \quad (4.8) \]

This term should be bounded by 1 if it is bigger: it’s possible for some value of \( \alpha \), but has no physic meaning in this case. For HET thrusters, we have the equation 4.9:

\[ \eta_i = \frac{(r^2 + 2. r_1 r_2)}{6.75. r_1^2} \quad (4.9) \]

We should keep in mind that this loss term \( \eta_i \) is an approximation and aims at representing the part of the beam that loses the target: As NEO’s shape is random, it’s impossible to set up a theory or accurate computation, and so to deal with such an approximation to compute the thrust generated by this method.

**Figure 18** - Momentum transfer Efficiency according to distance maneuver with R1=325m NEO and different values for other NEO dimension.
So the final momentum transfer rate $F_p$ is (with distance maneuver equal to 1.5 times the NEO bigger dimension):

$$F_p = 2. F_t. \frac{(r_2^2 + 2. r_1.r_2)}{6.75. r_1^2. \tan^2(\alpha)} - G \frac{m.M}{2.25. r_1^2} \quad (4.10)$$

This equation shows that the momentum transfer is linked to the thruster momentum. Another thruster is required to cancel the SC thrust, which doesn’t give its momentum to the NEO. But the momentum transfer has a factor two and so it is possible to just consider that all the momentum ejected by SC is transferred to the NEO. About the efficiency factor, the lower the difference between the smallest and biggest dimension/ semi principal axis of length, the better the efficiency, and on the contrary, the bigger this difference is, the lower is the efficiency. However the thruster half divergence angle has also a great importance.

To finish, the momentum transfer is small as the acceleration generated and as the gravitational force. Under some condition, gravitational force may affect a lot the results.

Figure 19 – Forces acting on Apophis during Ion beam shepherd deflection maneuver. Green is gravity attraction, Blue the Beam pushing force and Red the resultant force (pushing). Considering a 2 tons SC with 0.24N thrust
Example with Apophis:

Still considering the 2000kg SC with 0.24 N of thrust (0.24 N for push and another 0.24N thruster to cancel the first one) using HET with 45° of divergence half angle, Apophis characteristics are $a=325m$ and $b=c=109m$, for $4.2.10^{10}kg$. Considering that $2.10^{6} m/s$ is needed 18.5 years before hazardous moment:

The efficiency transfer is $\eta_i = 0.116$. The momentum transfer rate is $M_t=0.055$ N.s/s or kg.m/s² without gravity force and $M_t=0.032$ with it. Total momentum needed is 84000 N.s. So 726 hours are required or 30.2 days considering the 1.5 bigger radius for safety distance. Figure 19 shows the momentum transfer rate according to the deflection maneuver distance. In this case: 487meters.

4.3.3 DISUSSION & CONCLUSION

Although both methods work on different ways, there are lot of similarities and a SC made for one method could be used for the other one if thrusters are mounted on gimbal devices. Characteristics are also similar: for both methods efficiency decreases with the square of the distance, and the payload could be summed up by the momentum available. In fact in both cases momentum is required to transfer it to the NEO or keep the SC at the right distance.

Even so there are differences: One method uses attraction force and the other one repulsive force. The gravity link is more efficient with high density NEO, and the thrust generated grows up with the NEO size (not the acceleration). Ion beam shepherd method efficiency is decreased by gravitational force used in gravity link method and so this perturbation follows the same rule. However with low divergence half angle it could be very efficient and could allow a bigger safety/maneuvers distance without any loss in efficiency and so could reduce the gravitational perturbation.

Lastly for both methods some assumptions about the far field plume are made and if wrong could change a lot the results.

The method retained for the continuation of the study is the ion beam shepherd because of better results on Apophis example and because an error on the far field plume will not affect that much the results as gravity link method: the main energetic part of the beam is the one with high velocity and so a low divergence angle, it’s also this part which carries most of the momentum. For gravity link the part with low velocity (non-ionized Xenon particle) could create a particle cloud hitting the NEO and generating a perturbation thrust. This assumption on plume shape depends a lot on the thruster choice and if another thruster than HET could be used, with accurate information on far field plume, the choice could change too.
5 SPACECRAFTS

One of the aims of this study is to furnish bases to a global scenario for NEO protection. Reflection and action for NEO protection started 20 years ago and nowadays there is nothing: not even a single rocket layaway in case of problem. As seen previously, time could be a primordial factor for an efficient response against NEO threat, and each step of the response, SC designing, building and the launch campaign take months/years. In order to reduce this response time, it is assumed that a rocket intended for another launch campaign could be requisitioned. The campaign launch time should be compressed on the time. The last problem is the design and building of the SC. To reduce the time required, the SCs shown in this section is assumed to be built with devices available in the coming years and so does not require any special devices.

Three SC configurations will be tested for the mission: The first one comes from a project of versatile electrical tug that was the object of a concept study in CNES. The two others are designed especially for the mission with available devices and different characteristic choices. The SC design is described in the document [18]. It isn’t a full design of the SCs but a pre-design, estimation to see what SCs characteristics could be hoped for this mission and what performance they could give. During the time the method could work i.e. transfer momentum, this one is defined by the time the thruster could work (life time and propellant available). As the life time of thruster is considered as big enough, the only limiting factor is the propellant, and so the payload could be expressed as propellant that can be carried to the NEO.

5.1 SC CONFIGURATION 1

The first configuration comes from a versatile electric tug project studied as a draft by CNES. This tug is built around a central “block” where optional parts can be added depending on the mission. The Spacecraft is equipped with 4 thrusters’ PPS5000 using Xenon as propellant. The data shown corresponds to a version adapted for deep space mission and with some parts added especially for this mission: a High Gain Antenna (HGA) pack, additional propellant, and two thrusters for deflection mission (so deflection is made with a 2-2 thrusters’ configuration).

<table>
<thead>
<tr>
<th>SC configuration1</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Total mass (kg)</td>
<td>9000</td>
</tr>
<tr>
<td>Xenon mass (kg)</td>
<td>4900</td>
</tr>
<tr>
<td>ISP range (sec)</td>
<td>2180</td>
</tr>
<tr>
<td>Thrust range (N)</td>
<td>0.25</td>
</tr>
<tr>
<td>Total thrust (N)</td>
<td>1</td>
</tr>
<tr>
<td>Initial acceleration (m/s)</td>
<td>0.00011</td>
</tr>
<tr>
<td>Total impulse (MN.s)</td>
<td>104.8</td>
</tr>
</tbody>
</table>
For simulation only the thruster working point at 240 mN newton and 2180 sec ISP is considered. The solar panel is able to produce enough power for engine and other systems until 1.2 AU. Once this distance is reached, the thrusters have to work on a degrading mode because there is not enough power.

Total thrust is the thrust SC could generate with all the maximum thrusters able to work together (it’s limited by power supply). So for deflection operation only half of this thrust is available.

### 5.2 SC CONFIGURATIONS 2&3

These configurations are pre-designed estimations of two SC using data from manufacturer data and devices, based on existing SC. Two different versions correspond to two different concepts. As these concepts are close to each other, only one pre-designed study is performed and when there is a difference it is said. These two SC are lighter than configuration 1: If the deflection mission is performed soon enough the ΔV required is tiny and so a smaller SC can do the mission. Configurations 2 and 3 are set up to match this philosophy: sending a small SC at a low cost quickly to avoid problems rather than launching a bigger one later.

One version matches a SC with fully electrical propulsion (that implies lot of thrusters) and an auxiliary rescue system (only in case of problem, attitude control using electrical propulsion). The other one matches a SC also with electrical main propulsion system with only 3 big thrusters (2 for cruising, and 1 used with one of cruising thruster for deflection) but with a secondary propulsion system, for unsaturated attitude control system when main propulsion can’t.

Following table gives SCs characteristic retained for simulation:

<table>
<thead>
<tr>
<th>SC</th>
<th>Configuration 2</th>
<th>Configuration 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total mass (kg)</td>
<td>3600</td>
<td>4750</td>
</tr>
<tr>
<td>Propellant mass (kg) (Xenon)</td>
<td>2138.4</td>
<td>2610</td>
</tr>
<tr>
<td>Thruster model</td>
<td>TMA (PPS …)</td>
<td>PPS5000 with gimbal</td>
</tr>
<tr>
<td>ISP</td>
<td>1500</td>
<td>2180</td>
</tr>
<tr>
<td>Engine Thrust (N)</td>
<td>0.083</td>
<td>0.24</td>
</tr>
<tr>
<td>Total thruster number</td>
<td>12</td>
<td>3</td>
</tr>
<tr>
<td>Thruster working together</td>
<td>4</td>
<td>2</td>
</tr>
<tr>
<td>Total Thrust (N)</td>
<td>0.33</td>
<td>0.48</td>
</tr>
<tr>
<td>Initial acceleration (m/s)</td>
<td>0.0000885</td>
<td>0.00010105</td>
</tr>
<tr>
<td>Total impulse (MN.s)</td>
<td>31.2</td>
<td>55.81</td>
</tr>
</tbody>
</table>
As for SC configuration 1, thrust available for deflection is half of total thrust.

Figure 20 & 21 show respectively shape and thrusters configuration of SC 2 & 3

5.2.1 SC CONFIGURATION 2, DETAILS

Configuration 2 is designed to test a full electrical propulsion configuration. This configuration is thought to do every maneuver with main propulsion. It’s possible thanks to 12 Hall Effect Thrusters (HET) all mounted on gimbal support (TMA, [15]). This allows changing thruster orientation to have in each case the best efficiency for wished maneuvers. Power generator is designed to produce enough power for propulsion and all subsystems until 1.5AU. Once this limit is reached the thruster has to work on degraded mode not to damage other systems and keep working life systems (thermal control and so on). The design includes 4 PPU (Power Process Unit, [19]) which allow 4 thrusters to work together. SC design also allows to all thrusters a configuration to make the deflection maneuvers: it’s a highly redundant configuration. In case of problem a secondary propulsion system has been added. It’s a cold gas system using also the
Xenon from main propulsion system: thanks to this, there is no need for additional heavy tank and “useless” mass.

Full electrical SC, including attitude control (unsaturated device and so on) has never been tested yet and making a rendezvous with such a system is still theoretical today.

5.2.2 SC CONFIGURATION 3, DETAILS

Configuration 3 has lot of similitudes with configuration 2, same shape, low thrust propulsion system, and characteristics (power supply design and so on). However there are many differences: Contrary to configuration 2, configuration 3 uses only 3 Hall Effect thruster PPS 5000 developed by Snecma [20], (see figure 14) able to work 2 by 2. This means than a secondary propulsion system is needed for attitude control and if necessary to help main propulsion system give quick answer. The secondary propulsion system doesn’t work with Xenon used by primary one, but H₂O₂ and kerosene as monopropellant. That leads to have two different storage and propellant supply systems.

If configuration 2 is more a concept because lots of choices have never been tested (fully electrical propulsion and so on), configuration1 looks like SCs that have been tested.
6 MISSION ANALYSIS

In my mind a space mission analysis could be split in two phases: the ground phase which includes all the events before the launch. Then a flight phase with all the events from launch until the end of mission. The first phase is often not retained in mission analysis but in a NEOs deflection mission the time remaining before hazardous moment is a critical variable and the time spent on this phase should be included in the mission analysis.

6.1 GROUND PHASE

As said previously this phase includes all the important events for the mission before the lift off. This is set as a phase of the mission due to the time needed to achieve all the steps. This starts with detecting the NEO until rating it as dangerous. The second step includes the discussion until the deflection mission decision has been set. Third step is the SC design and mission analysis. Then the building of the SC and rocket. Last but not least the launch campaign.

All these steps could require a long time, and such a mission is more efficient when started early. Saving time is almost impossible on the second phase which is ruled by space motion laws, and saving time requires a huge amount of fuel and so a huge cost. Therefore if time could be saved it’s on the first phase which is ruled by human factors. There are many solutions to save time on this phase, according to the step: for the first step, adapted detection tools and detection campaign are a good start. On the second step, just starting to think about this threat is a beginning, otherwise there is actually no miracle solution. For the third step, having already studied mission scenario will reduce the mission analysis time and having solution requiring available devices will even more reduce the design part as well as the building part if these components are available. For the launch campaign there is no solution.

Making a time quotation for all these steps is hard because all these steps depend on human behavior which is hardly predictable. Nevertheless whatever this time is, it could be decreased if well prepared.
6.2 FLIGHT PHASE

The flight phase for a full asteroid deflection mission could be split in 4 different steps summed up in the following table:

**Lift off to injection:**
*It consists in launching the spacecraft from Earth to injection orbit and configuring it for the next step (solar array deployment, unspinning it from residual spin from separation maneuver, sun orientation for power generation and reloading batteries, initiating communication link with Earth and so on).*

**Earth escape transfer:**
*It consists in using propulsion (electrical propulsion for this mission) to escape earth influence sphere and putting the spacecraft in a near earth heliocentric orbit.*

In some cases, this phase could be avoided by using an earth escape injection orbit. In this case the SC is directly released on an escape trajectory.
Heliocentric transfer:
During its maneuver the spacecraft will use its electrical propulsion to transfer it to the NEO. It’s possible that this maneuver requires to wait for the right moment for the different maneuvers (inclination change, orbit adjustment, final approach) and so to turn on/off the engine many times.

Deflection:
It is during this phase that the deflection maneuver happens. It would be a two steps phase: first studying the NEO to have accurate information in order to take the right decision for deflection. Second step consists in changing a little the NEO velocity to make it miss the earth with the parameters choice thanks to step one. Figure 22 shows an artiste view of deflection maneuver.

Figure 22 - artiste view of gravity tractor deflection maneuver [13]

6.2.1 LIFT OF TO INJECTION

According to the launcher choice (A5 ME, A6, Soyuz or Vega) the mass available for a given injection orbit is different. As the main propulsion system has to be electrical, transfer from injection to earth escape is given in low thrust. Following table 5 gives the mass capacity injection for different European rockets.

<table>
<thead>
<tr>
<th>Launcher</th>
<th>LEO</th>
<th>800 km - Sun synchronous</th>
<th>GTO</th>
<th>Escape</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ariane 5 ME</td>
<td>21 tons 300km</td>
<td>16 tons</td>
<td>12 tons</td>
<td>7.5 tons (to Mars)</td>
</tr>
<tr>
<td>Ariane 6</td>
<td>14 tons 300km</td>
<td>10 tons</td>
<td>6.5 tons</td>
<td>4 tons (to Mars)</td>
</tr>
<tr>
<td>Soyuz</td>
<td>4.85 tons 920km</td>
<td>4.4 tons</td>
<td>3.25 tons</td>
<td>2.15 tons</td>
</tr>
<tr>
<td>Vega</td>
<td>2.15 tons 500km</td>
<td>1.5 tons</td>
<td>--</td>
<td>--</td>
</tr>
</tbody>
</table>

Table 5 - Launcher vehicle performance
Data for LEO is for circular orbit. Reference [21] [22] [23] [24].

The choice of the injection orbit is described in the next part and depends a lot on the thrust capacity (N/kg). The departure date is computed once all steps of transfer (Earth escape and heliocentric transfer) are computed and so travel time, trajectories and so on are known. Two launching strategies are considered:

- Launching to an escape orbit directly. This solution avoids the earth escape problem in low thrust but leads to using a bigger rocket.
- Or launching to a higher injection orbit as possible to avoid atmospheric drag if possible. Due to simulation limit, only circular orbits are considered.

SC configuration 1 should be launched using Ariane5 rocket and do not enable to escape orbit. A sun synchronous orbit at 800km in a plane without shadow area (at launch moment) is retained and will increase enough the orbit before shadow area occurs. SC configurations 2 and 3 could be launched with Soyuz or Ariane6 on circular orbit at 800 km altitude.

Configurations 2 & 3 could also be launched on an escape orbit respectively with Ariane6 and Ariane5; in this case the Earth escape phase is avoided. Although at beginning, other injection orbits like GTO were finally not considered in this study due to a simulator stability issue because of a bug.
6.3 EARTH ESCAPE TRANSFER

This step is highly dependent on the injection orbit. In the longer case (LEO injection) it can be split in two cases: one transfer with atmospheric drag, and one transfer where other perturbations like J2 are bigger than atmospheric drag, in accordance with altitude.

As shows figure 23 the effect of atmospheric drag decreases quickly as altitude grows. It shows the acceleration due to thrusters and atmospheric drag deceleration. Black curves are configuration 1, the projected surface is set to 29m² and drag coefficient 1,1. Green curves are configuration 3, the projected surface is set to 6m² and drag coefficient 0,8. Blue and red curves are configuration 2 in two different modes: (red) one using only 2 thrusters and so a lower acceleration but also a lower projected surface and drag coefficient, respectively 6m² and 0,8. Second mode (blue) uses 4 thrusters but has a bigger projected surface and drag coefficient, respectively 25m² and 1,1. Velocity corresponding to a circular orbit, and drag from solar panel aren’t considered. The thrust cancels drag at an altitude between 150 and 200 km for all configurations. This altitude is much lower than the injection altitude considered so atmospheric drag is neglected as other perturbations are.

Figure 23 - Drag & SC acceleration according to altitude. Black is conf1, blue and red are conf2 in two different flight configurations (see below) and green conf3
The $\Delta V$, propellant consumption and time spent during this phase are computed from a low trust simulator. The simulation is done using a circular initial orbit and the finishing one is also a circular orbit with as semi-major axis the Earth activity sphere radius or the moment when the escape velocity is reached. This radius is computed from equation 6.1:

$$\frac{r}{R} = \left(\frac{m}{M}\right)^{2/5}$$  \hspace{1cm} (6.1)

With $r$ the sphere activity radius, $R$ the distance from sun (1UA), $m \& M$ respectively the Earth and Sun masses. Numerical application gives $r = 924\,600\,000$ meters.

The escape velocity is given by $v_{esc} = \sqrt{2\mu/r_e}$ with $r_e$ the distance from Earth center of gravity.

Once the Earth activity sphere or escape velocity are reached, the SC is considered as being in heliocentric orbit, on the same orbit as the Earth.

6.4 HELIOCENTRIC TRANSFER

Generally heliocentric transfer between two bodies is computed as a Lambert problem and the result expressed on a Lambert diagram shows the $\Delta V$ required according to departure time. This method could be applied or adapted for low thrust trajectories if the thrust enables an acceleration big enough to make the two maneuvers in a short time compared to the period. Period on heliocentric orbit at 1UA takes about 1 year and so a travel on 5° arc takes 5 days. In this case speaking of low thrust alone doesn’t mean anything without the SC mass. Nevertheless the SC should enable to do both maneuvers and low thrust has often a very low acceleration, and is almost never able to do both maneuvers. Last but not least if power generator is solar array and not nuclear, the thrust available could change according to design and distance from Sun.

All these constrains make the Lambert problem inefficient to compute low thrust trajectories for heliocentric transfer. Because of this a simulator for low thrust trajectories was to be used.

The Heliocentric transfer also includes the approach and rendezvous maneuvers. This is computed from the best possible transfer to the target orbit (without rendezvous) according to travel time and propellant consumption. Once the trajectory is found, we consider than we can achieve the right position (rendezvous) by delaying the arrival days i.e. keeping the SC on a orbit close to final or by waiting during the travel to change arrival time. So consumption is considered to be the same and arrival time delayed by maximum one period of time from the target orbit. The rendezvous phase is also done with the second propulsion system for configuration 1 and 3.

The simulation starting point is Earth orbit and position and finishes when SC reaches target orbit.
7 SIMULATION TOOLS

All SC tested for the mission will use low thrust propulsion. Compared to impulsive maneuvers, low thrust maneuvers can’t be computed easily and require large resources. In order to have an estimation of the time and fuel consumption for the mission, a low thrust simulator has been created. Another tool available is a set of equations for low thrust trajectories for circular or near circular orbits.

7.1 LOW THRUST EQUATIONS FOR NEAR CIRCULAR ORBITS

It is assumed that for low thrust the acceleration is so low that the increase in velocity per period is also low compared to absolute velocity. Moreover the rate at which the energy of the orbit increases must be maximized, which means than the dot product $A.v$ must be maximized (with $A$ the acceleration vector and $v$ the velocity vector) and so must stay in the same direction. Considering a circular orbit, it is assumed that the orbit stays circular over the path and the altitude slowly increases. Actually it’s a low elliptical orbit, and the final trajectory will look like a spiral. [11] (p97 - 99)

For a circular orbit the velocity is given by equation 7.1:

$$v = \sqrt{\frac{\mu}{a}}$$

(7.1)

Considering start and final time $t_0$ and $t_1$, the total time of the maneuvers $t_m$ is given by:

$$t_m = t_1 - t_0 = \frac{\sqrt{\mu}}{A} \left( \frac{1}{\sqrt{a_0}} - \frac{1}{\sqrt{a_1}} \right)$$

(7.2)

Escape time is given for $a_1 \rightarrow +\infty$.

The total velocity increment required is given by $\Delta V_{tot} = A. t_m$.

Discussion:

The models have lot of limits: Actually the assumption that the orbit remains circular will break down as the SC approaches to escape velocity. Moreover the model assumes that acceleration is constant and so the mass doesn’t change over the time/maneuver. As we will see, these two limits are reached when the maneuver is to escape the Earth. However it’s enough to have an order of magnitude to check the simulator results.
7.2 SIMULATOR CHARACTERISTICS

This simulator has been programmed on Scilab [25] for the simulation and doesn’t use Pontryagin principle. Thus the solution isn’t the optimal solution and could be improved. The advantage of this simulator is the computation time lower than the one using Pontryagin principle, and much easier to use. Figure 24 shows a run example (modification of semi-major axe and RAAN with a non-null inclination).

![Figure 24 – Run example done with the simulator. In green the initial orbit, in red the final one and in black the trajectory computed.](image)

The main characteristics and assumptions are summed up there:

- It computes the trajectories of an object/SC considering a two body problem.
- It works only for elliptical trajectories. That is why escape trajectories are considered to be reached once the sphere of activity of the planet is left.
- It uses first order integration.
- No perturbation is assumed.
- It only works for low acceleration.

Three versions of the simulator have been developed:

The first one was to test the concept: it had a control variable (the efficiency of the transfer). The problem is that it was slow and became complicated. It used equation from [12] for orbit computation and worked only for non-circular and non-equatorial orbits. Lastly it was successfully tested for simulation around the Earth and the Sun.

The second version was achieved to tidy, store and then facilitate upgrades. It used Celestlab functions [26] for orbit computation, which allows circular and equatorial orbits. The problem was the very long computation time. It was only tested for simulation around Earth before being stopped.
The third and last version is the code of second one (with some improvements) and the equation used in the first version instead of Celestlab functions for computation. So it can’t work on circular and equatorial orbits, but is much faster! There are many problems with this version but the two main are a bug in a “while” loop and the lack of control variable. The absence of control variable means that simulation is made for the shorter time, and not for fuel efficiency with a maneuver time as other simulators do. So it worked for Earth trajectory, but experienced some troubles for heliocentric trajectory (too long period) since when the engine works in a continuous mode it leads to depleting fuel reserve and so on.

In the three versions the “step” is set as a variable part computed according to the position on the orbit and to the orbit, bounded by a constant part: the maximum value for the step which can’t be overcome. This step could be the angle or time between two iterations. The lower it is, the more accurate is the run. It could be interesting to make only one accurate run for the three configurations and extend the results thanks to formulas as seen later.

7.3 SIMULATION STRATEGIES

Some simulations with very low acceleration (like our SCs) coupled with short period orbit take a very long time. In case of low thrust transfer, for different thrusts, the consumption stays the same and the time of the transfer changes linearly with the thrust. A quick simulation has been done to check how the simulator operated.

Table 6 shows SCs acceleration.

<table>
<thead>
<tr>
<th>SC</th>
<th>Configuration1</th>
<th>Configuration2</th>
<th>Configuration3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass (kg)</td>
<td>9000</td>
<td>3600</td>
<td>4750</td>
</tr>
<tr>
<td>Thrust (N)</td>
<td>1</td>
<td>0.332</td>
<td>0.48</td>
</tr>
<tr>
<td>Acceleration (m/s²)</td>
<td>0.00011</td>
<td>0.000088</td>
<td>0.000101</td>
</tr>
</tbody>
</table>

Table 6 - SCs acceleration

A comparison of 9 “low thrust” transfers has been done. The parameters are:

- Initial orbit [1.10^7, 0.1, 6, 20, 60, 30];
- Final orbit [1.310^7, 0.1, 6, 20, 60,30];
- Spacecraft initial mass: 3500kg
- Thruster ISP: 1500 sec

An orbital parameters legend is [semi-major axis (m), eccentricity, inclination (°), RAAN (°), argument of periapsis (°), mean anomaly (°)]. The different forces tested are 1, 2, 5, 10, 15, 20, 25, 30, 40 Newton.
Figure 25 - semi-major axis and eccentricity variation for 1N (bottom) and 30N (top) run

Figure 25 show the important orbital parameters variation over time for two simulations. For both tests orbital parameters variation follows the same path. As far as velocity increases, every run follows the same command law shown on figure 26.
Table 7 gives the results for different runs. It shows that the time needed to make the transfer is linearly linked to the acceleration (so the thrust for the same mass). And for all runs the consumption is almost constant. It’s easy to explain because the consumption is linked to the ISP (constant for all runs), the force and the time which is linked linearly to the force.

<table>
<thead>
<tr>
<th>Thrust (N)</th>
<th>1</th>
<th>2</th>
<th>5</th>
<th>10</th>
<th>15</th>
<th>20</th>
<th>25</th>
<th>30</th>
<th>40</th>
</tr>
</thead>
<tbody>
<tr>
<td>Acceleration (m/s²)</td>
<td>0.00028</td>
<td>0.00057</td>
<td>0.0014</td>
<td>0.0028</td>
<td>0.0043</td>
<td>0.0057</td>
<td>0.0071</td>
<td>0.0086</td>
<td>0.0114</td>
</tr>
<tr>
<td>Time (s)</td>
<td>3.5E+6</td>
<td>1.75E+6</td>
<td>7E+5</td>
<td>3.4E+5</td>
<td>2.4E+5</td>
<td>1.73E+5</td>
<td>1.46E+5</td>
<td>1.21E+5</td>
<td>87000</td>
</tr>
<tr>
<td>Consumption (kg)</td>
<td>240</td>
<td>225</td>
<td>237</td>
<td>229</td>
<td>235</td>
<td>234</td>
<td>250</td>
<td>241</td>
<td>229</td>
</tr>
</tbody>
</table>

Table 7 - Results of simulation at different Thrusts
Figure 27 & 28 respectively show the time and consumption according to the thrust.

This figure shows that the results for a given thrust could be estimated using a run done with a bigger thrust: the consumption is the same, and the maneuver time easily computed from the results and actual thrust. This enables to save a lot of computation time: for example 1N run just to upgrade altitude to 3000 km takes around 6 hours, at 20N it’s only 18 minutes.

\[ \Delta V = V_e \ln \left( \frac{m_0}{m} \right) \]  

(7.3)

And \( V_e = g \cdot ISP \), with \( g \) the gravity acceleration on Earth.

With \( V_e \) the exhaust velocity, \( m_0 \) the initial mass and \( m \) the final mass.
To sum up two tricks are applied there: one to reduce the computation time of different runs by using a different thrust. And a second to apply this result to other configuration which has not the same mass, ISP and thrust and saves even more time. This saved time is spent to allow to make runs with a lower step: they are more accurate.
8 MISSION EXAMPLE: APOPHIS DEFLECTION

This simulation aims at having a look at the possibilities offered by different SC studies in terms of deflection against a NEO threat. This will also show the differences between the three configurations and how different factors affect the results (ISP, thrust and so on). The example is realized with the NEO Apophis.

Apophis was one of the NEO which had the highest score ever recorded on Torino scale: 4 [27]. After more accurate measurement of its path its mark was decreased to 0. Apophis is the example of potentially hazardous NEO by its size, and so widely used as an example for deflection mission.

Apophis orbital elements used for simulation are [28]:

- $a$: 0.92227889 AU
- $e$: 0.1910795169
- $i$: 3.33128998 degree
- RAAN: 204.45715738 degree
- $\omega$: 126.39365285 degree
- $M$: 215.5399846 degree
- Measured the: 2454571.318619088747 JED (Julian Ephemeris Day).

As said in analysis mission section, the travel is split in different parts for simulation, an Earth orbiting part (inside the activity sphere of Earth) and a heliocentric one. Once the activity sphere is reached, the SCs are considered in heliocentric orbit. At this moment a new simulation starts with as an initial orbit the Earth one and as a final orbit Apophis.

Earth orbital elements used by simulator are:

- $a$: 1.00000011 AU
- $e$: 0.01671022
- $i$: 0.00005 degree
- RAAN: 348.73936 degree
- $\omega$: 102.94719 degree $|_{\psi}$
- $M$: 100.46435 degree $|$
- Time reference: J2000
8.1 MISSION SIMULATION

8.1.1 EARTH ESCAPE

According to mission analysis plan, the mission starts from injection orbit. And finishes once Apophis is deflected.

Every configuration could be launched on a circular orbit of 800 km of altitude, or higher. Thus the injection orbit for simulation is a circular one at 800 km of altitude.

Due to instability of the simulator version 3, the simulation has been done in four steps, and the thrust values used are different of the SC configuration one: it’s a trick which allow a faster computation:

- 800km \(\rightarrow\) 16 000km, with 20N thrust (simulation 1)
- 16 000km \(\rightarrow\) 32 351km, with 30N thrust (simulation 2)
- 32 351km \(\rightarrow\) 160 000km, with 2N thrust (simulation 3)
- 160 000 \(\rightarrow\) Escape, with 0.2N thrust (simulation 4)

The Spacecraft characteristic choices for the simulation correspond to configuration 3 (4750kg initial mass, 2180 ISP). Information on different runs are summed up in the following table:

<table>
<thead>
<tr>
<th></th>
<th>Simulation1</th>
<th>Simulation2</th>
<th>Simulation3</th>
<th>Simulation4</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial mass</td>
<td>4750</td>
<td>4224.7</td>
<td>3943.9</td>
<td>3605.5</td>
<td>4750</td>
</tr>
<tr>
<td>Final mass</td>
<td>4224.7</td>
<td>3943.9</td>
<td>3605.5</td>
<td>3445.3</td>
<td>3445.3</td>
</tr>
<tr>
<td>Propellant initial mass</td>
<td>2610</td>
<td>2084.7</td>
<td>1803.9</td>
<td>1465.5</td>
<td>2610</td>
</tr>
<tr>
<td>Propellant final mass</td>
<td>2084.7</td>
<td>1803.9</td>
<td>1465.5</td>
<td>1305.3</td>
<td>1305.3</td>
</tr>
<tr>
<td>Propellant consumed</td>
<td>525.3</td>
<td>280.3</td>
<td>338.4</td>
<td>160.2</td>
<td>1304.2</td>
</tr>
<tr>
<td>Time (days)</td>
<td>277.6</td>
<td>162.8</td>
<td>201.6</td>
<td>101.5</td>
<td>743.5</td>
</tr>
<tr>
<td>Velocity increase (m/s)</td>
<td>2506.4</td>
<td>1470.9</td>
<td>1918.1</td>
<td>971.8</td>
<td>6867.2</td>
</tr>
</tbody>
</table>

It’s interesting to note that the escape happens when velocity reaches the escape speed and not the Earth sphere of activity. As said previously, the high difference of thrust values chosen for simulation, from 30N to 0.2N is to save time computation: the higher value is to allow a more
accurate simulation with affordable computation time. Even with 30N, the trajectory path looks like low thrust one and enables to compute the results with the real thrust. Once altitude is higher than 300 000km the SC trajectory is no more like low thrust but looks like impulsive maneuvers (period becomes too high) even if the thrust is set to the real value. This leads to set a thrust value lower than the real one (0.2N for this run), but even then the trajectory doesn’t look like low thrust trajectories and reaches escape speed. Another interesting thought is that fuel consumption is the highest at the beginning and becomes lower and lower as the altitude grows. Increasing altitude by 1km costs much more fuel when altitude is low than when altitude is high. This results is only true for circular orbits, other simulations have to be done for other injections and escape strategies.

The other method described section 7.1 to have an estimation of velocity increment and of time required for low thrust transfer for circular orbits gives the following results to reach 926 000km:

- 7025 m/s
- 778.7 days

And these results to reach and infinite distance from Earth (a1 \(\rightarrow +\infty\)):

- 7456 m/s
- 854 days

**Discussion:**

The differences in the results between two methods are due to the fact that the simulator takes into account the mass variation over time contrary to the equation which works well if the SC mass changes little during the maneuvers. Another source of error could be the orbit difference: as simulator can’t be used with circular trajectory, initial and final orbits for each run were low elliptical trajectories (e=0.05).

**Table 9 – Results for all configurations computed from the simulation’s results given table 6.**

<table>
<thead>
<tr>
<th></th>
<th>SC1</th>
<th>SC2</th>
<th>SC3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial mass (kg)</td>
<td>9000</td>
<td>3600</td>
<td>4750</td>
</tr>
<tr>
<td>Final mass (kg)</td>
<td>6568</td>
<td>2257</td>
<td>3445</td>
</tr>
<tr>
<td>Xenon consumed (kg)</td>
<td>2487</td>
<td>1342.4</td>
<td>1305</td>
</tr>
<tr>
<td>Final Xenon mass (kg)</td>
<td>2413</td>
<td>796</td>
<td>1305</td>
</tr>
<tr>
<td>ISP (sec)</td>
<td>2180</td>
<td>1500</td>
<td>2180</td>
</tr>
<tr>
<td>Thrust (N)</td>
<td>1</td>
<td>0.332</td>
<td>0.48</td>
</tr>
<tr>
<td>Impulse consumed (MN.s)</td>
<td>53.2</td>
<td>19.6</td>
<td>27.9</td>
</tr>
<tr>
<td>Final impulse (MN.s)</td>
<td>51.6</td>
<td>11.6</td>
<td>27.9</td>
</tr>
<tr>
<td>Time (days)</td>
<td>676</td>
<td>815</td>
<td>744</td>
</tr>
</tbody>
</table>
It’s possible to have better results using different injection strategies, for example GTO orbit instead of circular. Or to use an elliptical orbit to reach the escape velocity instead of trying to reach the sphere of activity. But this remains an assumption until run could be made to test.

8.1.2 HELIOCENTRIC TRANSFER

Because simulator wasn’t operational for heliocentric transfer this part has no results. Nevertheless due to high period it would be possible to compute these maneuvers like impulsive maneuvers and to apply a little coefficient to results to have an idea. Then, it is assumed that these maneuvers haven’t been done and will be considered in results.

8.1.3 DEFLECTION MANEUVER

It is a two-step maneuvers: a part where NEO is studied. It consumes very little propellant and this is insignificant and time needed can’t be predicted. The second step is the deflection maneuvers where time depends on the ΔV / momentum required: As said previously the payload could be resumed to a momentum which will be transferred to the NEO, and time is defined by the efficiency of the transfer (part transfer and the part loss) and by thruster characteristics (Thrust).

Table 10 shows some results and gives an idea of SC deflection characteristics. First row shows the final impulse available after escape (without loss). Second row shows the impulse after heliocentric phases, considering that ¼ of the impulse (total propellant) was spent to do this (arbitrary) and still without any loss. And the third row is the same as the second one with an efficiency of 30% (momentum loss because beam misses the Apophis). Last row is the distance deflection of this ΔV over 10 years.

**Table 10 – Deflection results**

<table>
<thead>
<tr>
<th></th>
<th>SC1</th>
<th>SC2</th>
<th>SC3</th>
</tr>
</thead>
<tbody>
<tr>
<td>ΔV with momentum</td>
<td>1.23</td>
<td>0.28</td>
<td>0.67</td>
</tr>
<tr>
<td>after escape (mm/s)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>ΔV with momentum</td>
<td>0.6</td>
<td>0.09</td>
<td>0.33</td>
</tr>
<tr>
<td>once NEO reached</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>(mm/s)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>ΔV with efficiency</td>
<td>0.18</td>
<td>0.027</td>
<td>0.1</td>
</tr>
<tr>
<td>(30%) (mm/s)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Deflection distance</td>
<td>145 000</td>
<td>21 800</td>
<td>80 500</td>
</tr>
<tr>
<td>over 10 years (km)</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Assumption:

- Apophis mass was set to $4.2 \times 10^{10}$ kg.
- The transfer efficiency was set to 30% due to Apophis shape which is far away from a sphere (bone-like).

9 DISCUSSION & CONCLUSION

First the method used for this example was the ion beam shepherd. But the difference between this method and the gravity link is just the origin of the force used, and also the position of thrusters. For example in configuration 2, the number of thrusters and their position allows to be used in gravity link maneuvers. A versatile SC which could perform both maneuvers according to which one is the most efficient once the NEO has been studied: for high density NEO, gravity link and for low density one, ion beam shepherd.

About the results of the simulation, it’s interesting to note that only a fraction of the momentum loaded is transferred to the NEO. Nevertheless Apophis is still deflected between 21 000 and 145 000 km according to different SC, with maneuver happening 10-12 years before. In fact with low thrust engine, transferring such a momentum could take years. Such mission is built up with estimation as data. Right values are only known after an accurate study of the NEO by the SC. So the SC is designed without accurate values about the momentum really required. The last thought is the $\Delta V$ transfer to the NEO, around $10^{-4}$ m/s which is enough according to studies that show that $10^{-6}$ m/s are required.

The last interesting assumption is that the deflection maneuver finishes ten years before hazardous moment. Considering two years to do the maneuver and to study the NEO, one year and a half of heliocentric trip to reach the NEO, a little bit more than two years to escape the Earth; it means that the rocket carrying the SC was launched at least fifteen years and a half before hazardous moment. And it takes many years to detect the threat, to take a decision, to set up the mission, and to design the SC and so on. To be efficient, the maneuver should happen early, but if we consider all the mission steps, it should start very soon.

To improve response time against such a threat there are solutions: be ready, have mission scenario, have available components, and have an international consensus help. But the most important thing should be to have a SC already on orbit, sleeping until a threat emerges. Such a SC should have to be sleeping on a “neutral” orbit where time wouldn’t damage it, so not on Van Allen belt, on Earth or Moon orbit protected by the Earth magnetic field, in penumbra or in “barbecue” mode to not be damaged by thermic cycle.
Three thoughts to keep in mind about asteroid protection. First we can’t protect us from a threat which is unknown and so the first protection actions are adequate detection means. Second, a lot of time is required for an efficient deflection mission: this mission should not fail (increase impact probability, or not enough deflection), that why have a B plan is necessary. Lots of nations have light but powerful enough nuclear bombs to be carried with the mission. It’s a hard option, but when there is nothing left to lose… Third thought is the reality of this threat since thousands of small objects fall on Earth every year, among which about 26 release an energy between 1 to 600 kilotons; and approximately every 18 years a major NEO (several Megatons) falls as well on Earth: Tunguska event in 1908, Curucá river event in 1930, Sikhote-Alin event in 1947 and so on [29].

10 THANXS

I wish to thank all the CNES team in SFR service for their welcoming and especially Duparcq Celine, Frish Pierre, Ruault Jean-Marc, Pinhede Herve and Mr Bonnal for their technical advices. Many thanks to Bultel Pascal for the all the time he spent with me and all the knowledge he shared with me.

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