Paper AAS 05-147

Strategies for Near Earth Object Impact Hazard Mitigation

Dario Izzo, Cristina de Negueruela, Franco Ongaro, Roger Walker

ESA - Advanced Concepts Team

e-mail: dario.izzo@esa.int

15th AAS/AIAA Space Flight Mechanics Conference

Copper Mountain, Colorado January 23-27, 2005

AAS Publications Office, P.O. Box 28130, San Diego, CA 92198

AAS 05-147

STRATEGIES FOR NEAR EARTH OBJECT IMPACT HAZARD MITIGATION

Dario Izzo, Cristina de Negueruela, Franco Ongaro, Roger Walker

Advanced Concepts Team

e-mail: dario.izzo@esa.int

The more we search for objects orbiting around the Sun the more we realize that the space out there is not empty at all. The incredible number of asteroids that have been discovered in the last decade is a good hint that there are many more yet undiscovered. Among these, the Near Earth Objects have been responsible for a number of disastrous events in the past history of our planet. A discussion has therefore recently started on how to mitigate the hazard, if at all possible, posed by these natural events to humanity. In this paper, a brief review of these strategies is made and their effectiveness is studied by the use of a novel theoretical result recently developed within the Advanced Concepts Team of the European Space Agency. In particular, a trade-off is made between a kinetic-energy-impactor deflection and a long-duration-thrust deflection.

NOTATION:

$d_{\min}$:	Miss distance due to the deflection effort.
$\vec{A}(t)$:	Time history of the acceleration vector applied to the asteroid.
R_{Earth}:	Earth-Sun distance at encounter.
V_{Earth}:	Earth velocity at encounter.
a:	Asteroid orbit semi-major axis.
t_s:	Time before impact at which the action is applied.
γ:	Encounter geometry parameter.
t_p:	Action duration, or ‘push time’.
η:	Impact efficiency.
$\vec{U}$:	Relative velocity between the spacecraft and the asteroid at impact.
M:	Asteroid mass.
m:	Spacecraft mass.
β:	Angle between the asteroid orbit and the Earth orbit at encounter.
$\vec{T}(t)$:	Time history of the spacecraft thrust.

$\vec{v}_{s\backslash c}$: Spacecraft velocity in the heliocentric frame.

$\vec{v}$: Asteroid velocity in the heliocentric frame.

INTRODUCTION

Near Earth Objects (NEOs) are asteroids and comets whose orbits are in close proximity with the Earth's orbit. The size of the population of these objects is not known as their small size, ranging from a diameter of 50 m to several hundreds of meters, makes their detection very hard. At the beginning of the eighties, the number of known NEOs was less than a few tens; today, thanks to the combined efforts of a number of search programmes involving ground based optical telescopes, this number has increased to over eight thousand. NASA stated that the current goal of its search programme is to detect 90% of the entire population of large NEOs (>1 km in size) within the next decade. The sizes of these objects are usually estimated from their absolute magnitude *H* assuming the albedo of the object to be in a particular range. In Figure 1, the orbital parameters distribution of a subset of NEOs called Potentially Hazardous Objects (PHOs) is shown. These asteroids have a Minimal Orbit Intersection Distance (MOID) smaller than 0.05 AU, and an absolute magnitude larger than 22. Catastrophic impacts can therefore be expected from objects belonging to this family. But how frequent is the impact of a NEO with our planet? The answer is closely related to the NEO population model that we accept as reliable. It is currently thought that the number of asteroids with a diameter of more than 1 km is 1000 ± 200. We have though to admit that the size of the population for smaller NEOs is yet quite uncertain. This introduces severe uncertainties on the prediction frequencies of catastrophic impact events such as the Tunguska one (see Chyba et al.[1]). This leads us to start at least considering which possible mitigation strategies we could implement to reduce the risk of an impact. These would, of course, all be related to the idea of somehow deflecting the asteroids trajectories away from their meeting with the Earth.

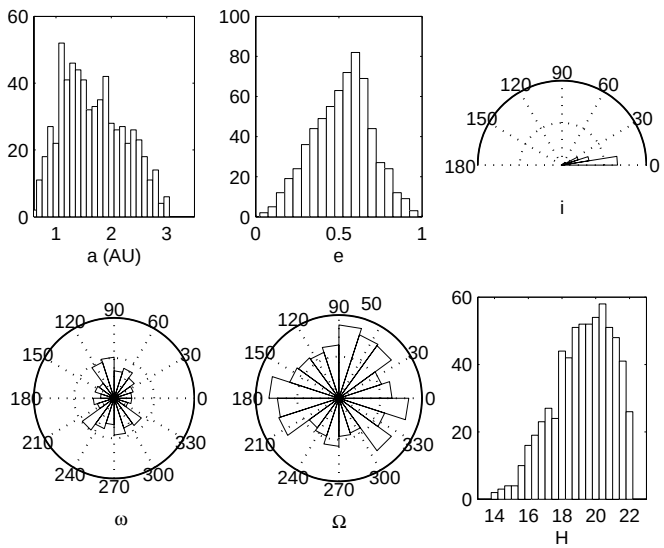

Figure 1: Parameter distribution of all known PHOs at epoch 05/11/04. The absolute magnitude is indicated with the letter H.

The deflection of an incoming Near Earth Object has been studied in several past scientific papers. Ahrens and Harris[3] considered the effect of a small impulsive ΔV on the asteroid orbit and on the final Earth encounter, assuming a circular orbit for the asteroid; Carusi et al.[2] removed this assumption considering an impulsive momentum transfer. A first expression concerning low thrust deflection strategies was given by Scheerers and Schweickart[4] in the case of a thrust constantly aligned with the velocity of an asteroid in a circular orbit. More recently Izzo[5] developed a general expression that accounts both for impulsive and low thrust ΔV and generic asteroid orbits. Walker et al.[6] used this result to assess deflection strategies based on the use of advanced solar electric propulsion and nuclear electric propulsion. This last work was performed as part of the ESA Advanced Concepts Team internal research on the Near Earth Object impact hazard mitigation problem.

ONE EXPRESSION FOR MANY DEFLECTION STRATEGIES

A small kinetic energy impactor precursor mission (named Don Quijote) has recently been selected by the European Space Agency as the highest priority for near-term implementation of NEO missions. The objective of this mission is to demonstrate the ability to perturb the state of an asteroid orbit via an impulsive momentum transfer, by measuring the asteroid state before and after impact. This will enable accurate models to be applied to properly assess the effectiveness of this method at full scale. The mission involves two spacecraft – one orbiter (Sancho) and one impactor (Hidalgo). In one of the reference scenarios being considered, both spacecraft are launched onto an Earth swing-by trajectory. The spacecraft take different swing-by geometries. The orbiter achieves a rendezvous/orbit insertion with a target asteroid, and the impactor performs an additional Venus swing-by to impact the asteroid at over 10km/s some six months after arrival of the orbiter. This mission stems from the recommendations of the NEO Mission Advisory Panel (NEOMAP) to ESA (Harris et al.[8]) and from earlier ones such as Carusi et. al.[2] (2002): ”*…one of the issues that the international community should handle is planning a deflection experiment of a real, innocuous NEO, to be chosen among those that have no possibility of representing a real threat.*”. Don Quijote, though, is primarily a technology demonstrator involving a deflection test. It is anyway important as it brings the kinetic impactor deflection strategy to the foreground as a viable mitigation measure.
Other strategies proposed in the past range from the exploitation of nuclear explosion radiation (Ahrens and Harris[3]) to the use of advanced propulsion devices (Scheeres and Schweickart[4], Walker et al.[6]). Other more speculative concepts have also been proposed and, overall, the mitigation strategies may essentially be divided into two basic categories:

- **high energy impulsive methods**: kinetic energy interceptor striking the object at high relative velocity or by a stand-off nuclear blast explosion

- **long-duration low-thrust methods**: surface ablation of the object using a laser or solar concentrator; mass drivers and propulsive devices in contact with the asteroid surface; exploitation of solar flux induced perturbations.

From a technology development standpoint, the two methods that could be most attainable in the nearer term are, in the opinion of the authors, kinetic energy interceptors and surface-attached propulsive devices.

In the work by Izzo[5], an analytical expression is introduced and is shown to accurately assess the long-duration low thrust methods efficiency. We here show that the same expression may be

used also for the high energy impulsive methods. The expression (named the asteroid deflection formula) has the following form:

$$d_{\min} = \frac{3a\gamma V_{Earth}}{\mu} \int_0^{t_p} (t_s - t)\vec{v} \cdot \vec{A} dt \tag{1}$$

where $d_{\min}$ is the minimal distance between the asteroid and the Earth whenever a deflection strategy $\vec{A}(t)$ is applied to the asteroid; V_{Earth} is the Earth's velocity at encounter, a is the asteroid's orbit semi-major axis, t_s is the time before impact the strategy is started, t is the time counted from t_s, γ is a non dimensional parameter that depends on the encounter geometry[5], μ is the gravitational parameter of the Sun and $\vec{v}$ is the asteroid velocity along its unperturbed orbit. In the case of an impulsive strategy we may think in terms of an instantaneous ΔV by writing the deflection strategy as $\vec{A}(t) = \Delta\vec{V}\delta(0)$, where δ is the Dirac's delta function. We immediately get, substituting into eq.(1):

$$d_{\min} = \frac{3a\gamma V_{Earth}}{\mu} t_s \vec{v} \cdot \Delta\vec{V} \tag{2}$$

which tells us, in accordance with Ahrens and Harris[3] but in a more general case, that the most effective way to deflect an asteroid is to impart the ΔV in the along-track direction. In this optimal case we may get the ΔV magnitude required to obtain a given minimal distance (this last quantity has also to take into account the Earth's lensing effect, see Scheeres and Schweickart[4] and Valsecchi et al.[7]). After some basic manipulation we get:

$$\Delta V = \frac{d_{\min}\sqrt{\mu r}}{3t_s \gamma V_{Earth}} \frac{1}{\sqrt{a(2a - R_{Earth})}} \tag{3}$$

This expression may be compared to the one developed by Carusi et al.[2] by noting that in their work they assume $\gamma = \frac{\left\|\vec{v} - \vec{V}_{Earth}\right\|}{V_{Earth}} \sin\beta = \frac{U}{V_{Earth}} \sin\beta$ and use non dimensional units. Substituting these last equalities into eq.(3) we get:

$$\Delta V = \frac{d_{\min}\sqrt{r}}{(3t_s U \sin\beta)\sqrt{a(2a - R_{Earth})}}$$

whereas the expression obtained by Carusi et al.[2] has the form:

$$\Delta V = \frac{d_{\min}\sqrt{r}}{(3t_s U \sin\beta + 2d_{\min})\sqrt{a(2a - R_{Earth})}}$$

The numerical difference* between the two expressions is negligible, and the error analysis performed by Carusi et al.[2] applies therefore also to eq.(3). This convinces us, together with the results derived by Izzo[5], that eq.(1) is valid also for high energy impulsive methods and may be used to trade-off different mitigation strategies applied to any asteroid.

THE KINETIC IMPACTOR

The kinetic energy impactor deflection method relies upon an impactor vehicle launched onto an interplanetary intercept trajectory. To make the asteroid deflection formula specific to this case, we write the conservation of momentum assuming a perfectly inelastic impact:

$$m\vec{v}_{s\backslash c} + M\vec{v} = (m+M)(\vec{v} + \Delta\vec{V})$$

where M is the asteroid mass, $\vec{v}_{s\backslash c}$ the spacecraft velocity and $\vec{v}$ the asteroid velocity before the impact. From the above expression we may evaluate the $\Delta\vec{V}$ imparted to the asteroid as $\Delta\vec{V} = \frac{m}{m+M}\vec{U}$ where m is the impactor mass and $\vec{U}$ the relative velocity vector. To take into account the increased momentum exchange due to ejecta materials from the impact and due to some inevitable translational-rotational energy exchange, we introduce the impact efficiency η and we substitute the expression found for the $\Delta\vec{V}$ into eq.(2) obtaining:

$$d_{\min} = \eta\gamma \frac{3aV_{Earth}}{\mu} \frac{mt_s}{m+M} \vec{v} \cdot \vec{U} \tag{4}$$

This expression tells us that all we can do to optimise the design of this kind of mitigation mission is to make $mt_s\vec{v} \cdot \vec{U}$ as high as possible (mission design) as well as η (impact design). This last parameter is highly dependent upon the surface/internal structure of the object concerned, particularly its material density/strength, porosity and aggregation, and on the precise knowledge of the asteroid centre of mass. Hence, missions to gather this physical data and to learn about the dynamic response of an asteroid to external forces are very important for refining such a mission scenario. To perform a preliminary assessment of this kind of concept we will assume pessimistically η =1. We note that the same result could have been achieved by starting directly from the asteroid deflection formula written in the form:

$$d_{\min} = \frac{3\gamma aV_{Earth}}{\mu} \int_0^{t_a} \frac{(t_s - t)}{M} \vec{v}(t) \cdot d\vec{p}$$

where $d\vec{p}$ is the asteroid momentum change given by $d\vec{p} = \eta \frac{Mm}{M+m} \vec{U}\delta(t)dt$. A simple evaluation of the integrand returns eq.(4). This offers us the physical interpretation of what we called impact efficiency, i.e. the fraction of the asteroid momentum that get actually transferred

* The difference arises from an inconsistent linearization of the energy equation performed by Carusi et al.[2]

to the asteroid. Note that this number can be significantly larger than one as a consequence of the materials ejected.

SURFACE ATTACHED PROPULSIVE DEVICES CONCEPT

This concept is based on the idea of designing a spacecraft equipped with a high specific impulse propulsion device and able to rendezvous with the asteroid, land on its surface and start pushing it so as to perturb its orbit and avoid the impact. To analyse this mitigation strategy the correct form of the asteroid deflection formula is:

$$d_{\min} = \frac{3\gamma a V_{Earth}}{\mu} \int_0^{t_p} (t_s - t)\vec{v} \cdot \frac{\vec{T}(t)}{M} dt \tag{5}$$

where $\vec{T}$(t) is the spacecraft thrust. The major problem of this strategy is the attitude motion (often a rotation) of the asteroid that has to be taken into account, as it influences the thrust direction. Some *ad hoc* form of attitude motion control has been proposed and studied by Scheeres and Schweickart[4], to allow the thrust direction to be constantly aligned with the asteroid velocity, thus maximising the effect on the miss-distance as shown in the equation above. Their concept assumes a rotating asteroid with a low eccentricity and requires some extra time to perform a preliminary spin axis reorientation manoeuvre. Whatever the thrust direction history is with respect to the asteroid velocity, eq.(5) might be used to build graphs describing the miss-distance variations with respect to the push time t_p, constrained by the available fuel remaining after rendezvous and landing, and the start time t_s. Walker et al.[6] used this methodology to perform a preliminary design of an advanced spacecraft able to land on an asteroid and push it up to 10000km away from a hypothetical impact, with an overall mission duration of roughly 12-18 years (different advanced propulsion methodologies and power systems were considered).

OPTIMISATION AND TRADE-OFF

In order to get an idea of which mitigation strategy is the most effective between the kinetic impactor and the rendezvous and push, we here take an electric propulsion spacecraft design similar to that introduced by Walker at al.[6], and we perform a preliminary design of two different missions to deflect the same hypothetical hazardous object. As a working scenario we select the orbit of asteroid 2003 GG21 and we assume its mass to be 10^{10} kg, corresponding to a 200 m diameter asteroid with a 2.4 g/cm^3 density. We assume the wet mass of the spacecraft to be 18000 kg, we assume advanced Nuclear Electric Propulsion capable to deliver 2N of thrust with a specific impulse of 6700 s. We also assume a zero departure C3 reached after a spiral out phase common to both mission profiles and requiring roughly 2000kg of fuel mass[6]. From that point we perform an optimisation of the heliocentric trajectory for the two cases and an assessment of the overall deflection capabilities.

Kinetic Impactor

For the kinetic impactor mission, as clearly shown in eq.(4), we want to maximise the mass of the spacecraft at the impact, to impact the asteroid as soon as possible, and to maximise the dot

product $\vec{v} \cdot \vec{U} = v_{ast}^2 - \vec{v}_{ast} \cdot \vec{v}_{s\backslash c}$. For this preliminary study, it was decided to use this last expression as an objective function of the optimisation. Further improvements (larger deflection distances achieved) may be possible by considering the full expression $mt_s \vec{v} \cdot \vec{U}$. A 2016 launch window was considered.
The simple equations of motion, written for the three-dimensional case:

$$\dot{\vec{v}} = -\frac{\mu}{r^3}\vec{r} + \frac{\vec{u}}{m}$$
$$\dot{\vec{r}} = \vec{v}$$
$$\dot{m} = -\frac{u}{I_{sp} g_0}$$

have been transformed into a Non Linear Programming (NLP) problem using a direct transcription method.

Kinetic Impactor Scenario		
Departure Epoch	6202	MJD
Interception Epoch	7993	MJD
Avoided Impact Epoch	9210	MJD
Heliocentric phase Duration	4.9	years
Final Mass	11852	kg
Final $\vec{v} \cdot \vec{U}$	1630	$km^2\backslash s^2$
Final $\vec{v}$ (heliocentric)	[5, 42, 1.5]	km\s
$\vec{U}$ (heliocentric)	[47, 30.7, -2.1]	km\s
Obtained miss-distance	43851	km
Minimal Earth-Sun distance	.22	AU

Table 1: Results of the optimised trajectory: kinetic impactor case. The impact efficiency has been set to η =1.

Initial and final states were constrained appropriately to ensure a zero C3 Earth departure and an asteroid impact, whereas the initial time was also optimised. The resulting trajectory, shown in Figure 2, resulted in a mission 4.9 years long hitting the asteroid near the perihelion and with an almost perpendicular direction. Some other information are given in Table 1, we note that the spacecraft is required to perform a close solar swing-by. A constraint may then be considered on the minimum spacecraft-Sun distance reached, in which case the impact takes place further up in the asteroid orbit. It was found, though, that the obtained miss-distance does not become less than 25000km even when a .5 AU minimum distance is required.
Because of the objective function shape $J = v_{ast}^2 - \vec{v}_{ast} \cdot \vec{v}_{s\backslash c}$ the optimiser tries to make the impact as close as possible to the perigee, and with a small component (negative if possible) of the asteroid velocity along the spacecraft velocity.
If we consider that the forecasted impact was at MJD=9210 (this is in reality just a close approach of the asteroid that reaches in that date a distance equal to the MOID) we would obtain

that the mitigation strategy above simulated would have moved the asteroid 43851 km (the miss-distance) away from the impact point!

Long duration thrust

Let us now consider the other scenario. From eq.(1) is clear that in the case of a strategy based upon a long duration high impulse low-thrust we have to maximise the integral $\int_0^{t_p} (t_s - t)\vec{v} \cdot \frac{\vec{T}(t)}{M} dt$. This is mainly related to the push time available before all the fuel on board is consumed, and therefore to the final mass at rendezvous. For this reason the heliocentric transfer phase has been optimised with respect to the mass, constraining the final relative velocity to be zero as to obtain a capture of the spacecraft. The resulting trajectory, shown in figure 2, is 5.7 years long and allows for a final mass of 12985 kg. More data are given in Table 2. In an overoptimistic scenario we would start to push the asteroid, from as soon as it has achieved rendezvous, and we would be able to push it in a direction always parallel to the asteroid velocity. In this case the asteroid deflection formula returns a value of 3697 km.

Long Duration Thrust Scenario		
Departure Epoch	5937	MJD
Rendezvous Epoch	8035	MJD
Avoided Impact Epoch	9210	MJD
Heliocentric Phase Duration	5.7	years
Final Mass	12985	kg
Obtained miss-distance	3297	km

Table 2: Results of the optimised trajectory: long duration push case.

Comparison

It is immediately seen that the same spacecraft is able to achieve a much larger deflection of the asteroid when it uses its high specific impulse engines to accelerate toward a maximum momentum exchange impact, rather than rendezvousing with the asteroid and pushing. Here, the trade-off is really between using each kg of mass to expel it at high speed via an advanced propulsion system or to impact the asteroid at high speed. In both cases, the resulting perturbation is due to the reaction principle stated by Newton's third law, and one might think of the impact case as being an advanced propulsion system able to expel at once the entire final spacecraft mass with an exhaust velocity of $-\vec{U}$. This observation allows us to evaluate the equivalent effective specific impulse of the push in the kinetic impactor scenario, here defined as $I_{sp}^* = \frac{\vec{U} \cdot \hat{v}}{g_0}$, and that in the simulation performed was evaluated to be $I_{sp}^* = 3800$ sec.

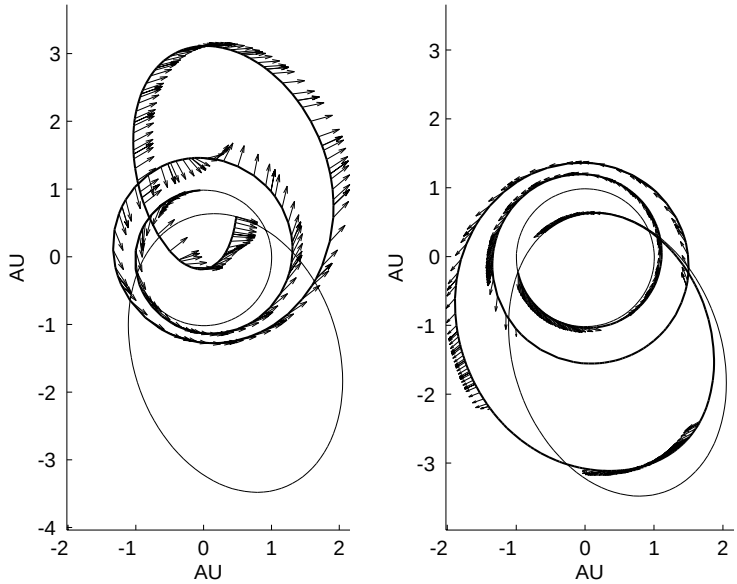

Figure 2: Optimised trajectories for the kinetic impact scenario and the long duration thrust scenario. The thrust direction is represented by arrows.

This value is only about one half of the specific impulse assumed for the actual advanced propulsion system[6] that would otherwise perform a slow push, but the chance to impart the ΔV all at once near the perihelion makes, in the case considered, the kinetic impactor strategy more efficient from the dynamical point of view. In the asteroid deflection formula this is accounted for by the integrand $(t_s - t)\vec{v} \cdot \frac{\vec{T}(t)}{M}$ that tells us that any mass expelled after a time t from the deflection start t_s contributes increasingly less to the miss-distance. Note also that in the kinetic impactor scenario, all the spacecraft mass is used as propellant, whereas in the other case only a part of it can be used, depending on the system design.

CONCLUSIONS

The asteroid deflection formula, derived in the framework of studies on long duration low-thrust asteroid deflection strategies, is shown to retain its accuracy for the evaluation of concepts based on high energy impulsive methods, allowing for a direct trade-off of the two concepts. An optimisation of the heliocentric trajectories has been performed for two different mission profiles exploiting these ideas: a kinetic impactor and a “rendezvous and push” spacecraft. In the case of the asteroid 2003 GG21, and assuming an advanced spacecraft design, the resulting miss-distance has then been evaluated via the asteroid deflection formula. The results revealed , from a dynamic point of view, the extreme efficiency of a kinetic impactor strategy coupled with an electric propulsion spacecraft that allows for high optimal relative velocities to be reached. Due to the high impact energies involved, though, the structural integrity of the asteroid may be endangered and a less disruptive mission may still result to be favourable.

REFERENCES

1. C.Chyba, P.Thomas, K.Zahnle: "**The 1908 Tunguska Explosion: Atmospheric Disruption of a Stony Asteroid**", Nature 361, pp.40-44, 1993.

2. A.Carusi, G.B.Valsecchi, G.D'Abramo, A.Boattini: "**Deflecting NEOs in Route of Collision with the Earth**", Icarus, No..159, pp. 417-422, 2002.

3. T.J.Ahrens, A.W.Harris: "**Deflection and fragmentation of near-Earth asteroids**", Nature, No.360, pp. 429-433, 1992.

4. D.J.Scheeres, R.L.Schweickart: "**The Mechanics of Moving Asteroids**", Paper AIAA 2004-1446, Planetary Defense Conference, Orange County, California, February 24-26, 2004.

5. D.Izzo,: "**On the deflection of Potentially Hazardous Objects**", Paper AAS 05-141, Copper Mountain, Colorado, January 23-27, 2005.

6. R.Walker, D.Izzo, C.de Negueruela, L.Summerer, M.Ayre, M.Vasile: "**Concepts for Near-Earth Asteroid deflection using spacecraft with advanced nuclear and solar electric propulsion systems**", Paper IAC-04-Q.5.08, 55th International Astronautical Federation Congress, Vancouver, Oct.4-8, 2004.

7. G.B.Valsecchi, A.Milani, G.F.Gronchi, S.R.Chesley: "**Resonant return to close approach: Analytical theory**", Astronomy and Astrophyisics, No. 408, pp 1179-1196 2003.

8. A.W.Harris, W.Benz, A.Fitzsimmons, A.Galvez. S.F.Green, P.Michel, G.B.Valsecchi: "**Recommendations for an ESA Initiative to Further Our Understanding of the Near- Earth Object Impact Hazard**", COSPAR 2004, Paris, France, July 2004.