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NEO THREAT MITIGATION SOFTWARE TOOLS WITHIN THE NEOSHIELD PROJECT AND APPLICATION TO 2015 PDC

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Protecting Earth from the threat implied by the Near Earth Objects (NEO) is gaining momentum in recent years. In the last decade a number of mitigation methods have been pushed forward as a possible remedy to that threat, including nuclear blasts, kinetic impactor, gravity tractors and others. Tools are required to evaluate the NEO deflection performances of each of the different methods, coupled with the orbital mechanics associated to the need to transfer to the target orbit and maybe rendezvous with it. The present suite of tools do provide an integral answer to the need of determining if an asteroid is to collide with Earth (NIRAT tool), compute the required object deflection (NEODET tool) and assess the design features of the possible mitigation space missions (RIMISET tool). The tools are presented, their design analyzed as well as the methods and architecture implemented. Results are provided for the hypothetical NEO 2015 PDC proposed for this conference.

INTRODUCTION

Currently, there are a number of institutions worldwide that contribute to the discovery, tracking, identification, cataloguing and risk characterization of asteroids in general, and NEOs in particular. However, there is no currently an integrated set of tools that cover in a complete manner the assessment of the impact risk mitigation actions that can be taken to prevent the impact of a NEO on Earth and to allow helping the dimensioning of space missions to address such problem.

Within the EC funded NEOShield project[*] started in 2012 the following set of utilities has been developed by Elecnor Deimos to allow covering the abovementioned activities:

- NEO Impact Risk Assessment Tool (NIRAT)
- NEO Deflection Evaluation Tool (NEODET)
- Risk Mitigation Strategies Evaluation Tool (RIMISET)

[*] See www.NEOShield.net

NIRAT, the first tool, allows evaluating the projection of the b-plane dispersion at the dates of possible impact for likely impactors and also the presence of keyholes that would enable future collision opportunities. This tool allows characterizing the impact probability for the different opportunities and, together with the knowledge of the asteroid features, the evaluation of the risk in terms of the Palermo Scale and the Torino Scale. This tool resembles current performances achieved by NEODyS[1] and Sentry[2], but does not intend to represent the same level of accuracy in the obtained results. The services provided by this tool are required by the next other tools.

The second tool, NEODET, allows assessing the required optimal change in asteroid velocity (modulus and direction) at any given instant prior to the possible impact epoch that would allow shifting the dispersion ellipse out of the contact with the Earth. This by means of impulsive mitigation options (one or several impacts) and by the accumulated effect that slow-push techniques (e.g. gravity tractor) could impose on the asteroid orbit to achieve optimal deflection.

Finally, the RIMISET tool allows evaluating how each of the possible impulsive and slow-push mitigation techniques would meet the required changes in asteroid state to obtain the searched for deflection and the requirements that this could impose on the design of the mitigation mission. Each technological solution would be simulated to allow ascertaining the efficiency in achieving the goal deflection by any of the proposed means (impact, explosive, gravity tractor and possible combinations of those). Ultimately, it serves to dimension the required mitigation space systems and solutions.

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THE NEO RISK ASSESSMENT TOOL - NIRAT

The NEO Risk Assessment Tool (NIRAT) is a piece of software aimed at the identification of potential future collision threats from Near Earth Objects to Earth. When a new asteroid is discovered and its orbit is estimated, relevant uncertainties on the accuracy of the orbit determination (OD) solution may be present[4,5]. This means that the actual evolution of the NEO orbit could deviate significantly from the reference solution. If intermediate planetary encounters are present, they can contribute in amplifying the size of the uncertainty region at the epoch of the threat, possibly increasing the risk on Earth. It is thus fundamental to evaluate as many as possible different trajectories compatible with the uncertainty domain, in order to identify the ones which may collide with the Earth and to provide a statistical evaluation of the risk.

The state of the art tools for asteroid OD and collision risk monitoring are CLOMON2[6,7], managed by university of Pisa together with other institutions and Sentry[8], operated by JPL. NIRAT is not meant to achieve the same level of accuracy and completeness of those systems, but aims at evaluating, by the means of simple algorithms and with minimum intervention by the user, the risk of possible future impacts, providing a quick assessment tool to support system-level studies for hazard mitigation missions.

NIRAT software is an integrated tool combining some well-known astrodynamical techniques in a simple and easy to use environment. In the following paragraphs the main aspects of the adopted techniques are briefly summarized. Among those techniques stand the Monte Carlo sampling and the Line of Variations sampling.

The core of NIRAT tool consists in a Cowell orbit propagator, based on a variable step Runge-Kutta-Fehlberg 7(8) integration scheme[9]. The center of integration is the Sun, but whenever the sphere of influence of a planet is entered, the central body is automatically changed to preserve numerical accuracy. Gravity from the planets can be treated with point-mass or spherical harmonics models, and their positions and velocities are obtained by ephemeris reading and interpolation (no integration of planetary motion). The gravitational effect of the major asteroids is also considered in the propagation. A basic modeling of relativistic effects is included, considering the Sun monopole term only[10]. Solar radiation pressure is treated with a simplified radial model. To enhance tool flexibility, instantaneous changes of velocity or continuous accelerations on the asteroid can be applied to simulate fast and slow push mitigation techniques.

Monte Carlo Sampling

The Monte Carlo (MC) engine permits to propagate a multitude of possible asteroid orbits, or Virtual Asteroids (VAs), all compatible with a certain OD solution. An OD solution consists in a set of orbital elements given at a specified epoch, and an associated uncertainty region, which in general can have a curved shape, but for many practical applications can be well approximated by a 6-dimensional ellipsoid defined through a covariance matrix. The MC sampling method implements a random perturbation of all (or some) of the orbital elements, based on a multivariate normal distribution obtaining the covariance matrix from the mentioned OD solution. Perturbed solutions are derived by using a normal random generator, based on a Mersenne twister algorithm[11] and a Box-Muller transform[12], which is considered to have a sufficiently accurate statistical behavior. Custom scaling can be applied to the covariance matrix prior to the calculation of eigenvalues/eigenvectors, to improve numerical conditioning. Different parameterization can be used for the definition of initial state and covariance, such as Keplerian elements, position/velocity state vector, cometary and equinoctial elements, to allow compatibility with different formats of OD solutions. During propagation, all planetary close encounters are tracked, for b-plane analysis, keyhole identification and risk probability computation for those trajectories ending with a collision, named Virtual Impactors (VIs).

Line of Variations Sampling

Although very simple and effective, Monte Carlo sampling is also very intensive in terms of computational resources if statistical accuracy is required. With impact probabilities lower than 10^{-4} and tens of years propagations the problem results too heavy to be practically affordable on standard processors. The concept of Line of Variations[6,13] is based on the idea that the uncertainty region associated with an OD solution has typically an elongated shape, especially when the asteroid observations are sufficiently separated in time. When propagated over very long time spans, different solutions belonging to the uncertainty domain spread in true anomaly, and the uncertainty region becomes a very elongated, curved and thin tube which may even include the entire orbit. A sampling along a one-dimensional subspace of the initial uncertainty region, following the weakest direction of the OD solution, may be sufficient to capture the essence of the multiplicity of the solutions. This one-dimensional space is called Line of Variations (LOV), it is generally curved and it can have different mathematical formulations[6], but in many practical cases it is well approximated by the major axis of the covariance ellipsoid of the reference OD solution. This last simplified version of LOV is implemented in NIRAT, and can be efficiently sampled with a limited number of points for the search of virtual impactors.

THE NEO DEFLECTION TOOL - NEODET

The second tool in the set, NEODET, focuses on the exploration of orbital dynamics constraints in a possible deflection mission of an asteroid that has previously been identified as a threat by NIRAT. The tool supports two different types of deflection attempts:

1. Impulsive problems: a nearly instantaneous change in the asteroid velocity without a change in position. This case models events such as the crash of a kinetic impactor or a nuclear blast.
2. Continuous problems: a force applied over a period of time long enough to span a significant part of the NEO orbital period. Examples are the gravity tractor and the ion-beam shepherd methods.

In each case, the program allows the evaluation of either the direct or the inverse problem. In the former, the input consists of a description of the introduced perturbation (e.g. the imparted $\Delta\mathbf{v}$ for an impulsive problem) while the output is the associated b-plane deflection. For inverse problems, however, the input is the desired b-plane displacement and the output is an optimal perturbation that produces the requested deflection. Furthermore, an additional inverse case is provided for impulsive deflections, see below.

Most of the logic in the tool revolves around the propagation of selectively perturbed virtual impactors, often within the context of an optimization of some function of the perturbation parameters. The particular algorithm is different for each problem, as described in the following sections.

To the effects of this tool, all NEOs are modeled as dimensionless points, with inertial mass but no gravity of their own. The main propagation model is the same used in NIRAT. An alternative propagation model is provided for continuous thrust problems, based on the analytical solution described by Bombardelli and Baù[14]. The NEODET implementation employs the simple expression for the spatial lag $\Delta\zeta$, which provides a good enough level of approximation; but includes expansions up to order 8 of the MOID $\Delta\xi$. The implementation of some functions required the computation of complete elliptical integrals of the first and second kinds, based on the method described by Adlaj[15].

The Impulsive Problem

The perturbation is modeled as an instantaneous velocity change $\Delta\mathbf{v}$ over the state vector $\mathbf{x}$, applied to the NEO at a certain point in time t_b. All problem subtypes are based on the *b-plane displacement function*, $\Delta\mathbf{b}(\Delta\mathbf{v}, t_b)$, which evaluates the change in the b-plane representation of the close approach (CA) when the perturbation is applied:

$$t_0, \mathbf{x}_0 \rightarrow \begin{cases} \qquad\qquad t_{CA0}, \mathbf{x}_{CA0} \Rightarrow \mathbf{b}_0 \\ t_b, \mathbf{x}_b \rightarrow t_b, \tilde{\mathbf{x}}_b \rightarrow t_{CA}, \mathbf{x}_{CA} \Rightarrow \mathbf{b} \end{cases}$$
$$\Delta\mathbf{b}(\Delta\mathbf{v}, t_b) = \mathbf{b}(\Delta\mathbf{v}, t_b) - \mathbf{b}_0 \quad [1]$$

The impulsive direct problem is a simple application of the above definition, as the input consists of the impulse to apply and the time of the deflection. In the hybrid case, the input is an admissible impulse magnitude $\Delta\mathbf{v}$ applied at a time t_b, and the result is the orientation that will produce the largest b-plane displacement. In other words, an optimization problem on the impulse **u**, where the magnitude of $\Delta\mathbf{b}$ from [1] is the function to maximize:

$$\max_{\mathbf{u}} \left\| \Delta\mathbf{b}(\mathbf{u}; t_b) \right\| \quad \Big/ \quad \left\| \mathbf{u} \right\| = \Delta v_{req} \quad [2]$$

A similar reasoning applies to the inverse case, where the input is a desired b-plane displacement Δb produced by an impulse at a time t_b, and the result is the smallest impulse **u** fulfilling the constraint:

$$\min_{\mathbf{u}} \left\| \mathbf{u} \right\| \quad \Big/ \quad \left\| \Delta\mathbf{b}(\mathbf{u}; t_b) \right\| = \Delta b_{req} \quad [3]$$

In both cases, the optimization procedure employs a non-linear recursive quadratic programming method.

<u>The Continuous Problem</u>

The perturbation is modeled in this case as a force **F** applied between two instants of time t_b and t_f. The treatment and form of the force in the numerical integrator uses a generic model:

$$\mathbf{F}(\mathbf{x}, t) = F(r_{Sun}(\mathbf{x}))\, \mathbf{A}_{i0}(t) \cdot \mathbf{u}_{\mathbf{F}} \quad [4]$$

Where $\mathbf{A}_{i0}$ is the rotation matrix from one of the defined reference frames (see below) and $\mathbf{u}_{\mathbf{F}}$ is a constant unit vector. There are essentially three generic degrees of freedom:

1. The model of the thrust magnitude, either constant or dependent on a power of the distance to the Sun.
2. The orientation of the force, given as constant right ascension/declination.
3. The base frame, which may be the ICRF, the orbital perifocal frame or the trajectory intrinsic frame.

In the continuous case, the b-plane displacement function for a particular force model is defined as $\Delta\mathbf{b}$ (t_f, t_b). The analytic model provides explicit formulas for ($\Delta\xi$, $\Delta\zeta$), while in the numerical propagator the force described above is incorporated as another perturbation to the integration. The function is evaluated in a similar way as in [1], and with the same special cases mentioned there. Also, like in the impulsive case, the continuous direct problem is solved by the direct application of the b-plane displacement function.

The definition of the "inverse case", on the other hand, is not unique because several parameters could be chosen as outputs. The formulation in NEODET returns the minimum time t_f (for a fixed t_b) that is needed to successfully deflect the NEO by the requested amount on the b-plane. The implementation *assumes* that $|\Delta\mathbf{b}|$ is roughly monotonic in t_f, removing the need for the optimization. Instead, the tool solves:

$$\left\|\Delta\mathbf{b}(t_f;t_b)\right\| = \Delta b_{req} \qquad [5]$$

For t_f values between t_b and the forecast impact date t_{CA0}. The implementation employs a bisection-like algorithm, which allows the tool to determine *a priori* (under the mentioned assumption of monotonicity) whether the requested deflection is at all possible and save time if it is not. The mentioned assumption of monotonicity in $\Delta\mathbf{b}(t_f, t_b)$ amounts to stating that the secular drift terms are the main drivers of the solution and that continued pushes keep affecting the b-plane in roughly the same direction. If the NEO swings by a massive body in the thrust arc, the latter condition may not hold, since the modified orbit may cause pre- and post-flyby pushes to affect the b-plane in significantly different directions.

THE RISK MITIGATION STRATEGIES EVALUATION TOOL - RIMISET

The last tool in the risk mitigation suite, RIMISET, is designed to evaluate the performance of different mitigation methods in a particular deflection situation and compare their results based on certain figures of merit that shall allow deriving requirements for the definition of a dedicated spacecraft. Each mitigation method is supplied with information about the NEO, its trajectory, the available transfers to reach it from Earth and other method-specific information. The program allows two different problems to be defined, each with a different set of figures of merit:

1. Direct problems, where the methods are allotted a given Earth escape spacecraft mass for each case, and return the maximum attainable NEO interaction with such a mission. In other words, methods based on impulsive deflections return the largest $\Delta\mathbf{v}$; while those that cause continuous deflections return the longest push time T_p that may be sustained.
2. Inverse problems, where the methods are asked to produce a certain orbital interaction on the NEO given by either of the measures named above, and return the smallest Earth escape spacecraft mass of the mission that will fulfill the requirements.

The design is thus complementary to NEODET, which computed how to deflect a threatening object from the standpoint of *orbital mechanics*. In other words, NEODET quantifies the deflection requirements and RIMISET examines the performances of the methods available to actually create such a deflection. Unlike other tools of the suite, RIMISET has no orbital propagation capabilities, relying on data from other tools and focusing on the implementation of the mitigation methods.

NEO Modeling

In the first two tools, the target object is represented only by its extended state vector ($\mathbf{r}$, $\mathbf{v}$, m). However, most mitigation methods need more detailed information on the NEO. As a minimum, the additional information shared between methods consists of the ephemerides of the unperturbed NEO trajectory, the asteroid mass and a model of the object size based on two co-centered spheres of radii r_a and R_a, which mark the inner and outer extents of the asteroid surface. Other information, like data on the properties of the surface material, is method-specific and will be described as required on the relevant subsections.

Deflection and Orbital Data

All mitigation options in the tool require the output of at least one NEODET case targeting the object in question, as the program is designed to operate over a series of dates which have associated deflection specifications obtained from NEODET. Some methods employ such data even if the program is configured for a direct problem, e.g. the impulse-based methods often use the direction of the optimal impulse in their computations. RIMISET allows the specification of several NEODET results files, which may be assigned to different methods.

Other important information is the specification of the spacecraft state at arbitrary points in time, e.g. to compute the available solar power needed for the operation of some mitigation options (e.g. those using solar electric propulsion). However, given that RIMISET has no orbital propagation facilities, the tool assumes that the S/C orbit follows the same path as the unperturbed NEO. This information is supplied by NEODET in a simplified binary ephemerides format: as the use of this information does not require high levels of precision in this context, the tool merely performs linear interpolation on the ephemerides data.

Earth–NEO Transfer Handling

The fact that RIMISET must examine how to impart a particular deflection requires the knowledge of the Earth–NEO transfer trajectory; particularly the flight time, the arrival conditions and the amount of fuel consumed by the en-route maneuvers. Given that the tool lacks the features to perform transfer propagations, the task is left to an external Elecnor Deimos trajectory optimization tool, which produces a large number of transfer solutions to the target NEO including multiple planetary swing-bys and maneuvers. As with deflection files, RIMISET allows the selection of several solution files which are then assigned to different methods using aliases.

Spacecraft Propulsion Model

Slow-push mitigation methods interact with the threatening object over a long period of time, so the mission spacecraft will need to fly a certain trajectory near the object, which requires a dedicated propulsion subsystem as part of the mission payload. RIMISET models the thrust level from the propulsion subsystem by a constant value k times a time-varying part $f(t)$. Two models are available: a constant-thrust engine and a solar-powered engine, both with constant specific impulse I_{sp}. In the former k is the thrust itself and $f \equiv 1$; while in the latter k is the thrust at 1 AU from the Sun and f is the inverse of the square of the distance to the Sun (in AU). Using the mentioned model, the propellant mass that is spent in a given push time T_p is:

$$\Delta m_p(T_P) = \frac{k_T}{I_{sp} g_0} \int_{t_0}^{t_0+T_P} f(t)\,\mathrm{d}t \quad [6]$$

Where the sub-index T in the engine reference value stands for the *total* thrust; that is, the combined output of all thrusters using the same model. However, the full engine block includes other elements which are represented in the power plant term m_{pp}:

$$m_{pp}(T_P) = \alpha_{pp} F_T \frac{I_{sp} g_0}{2} \frac{1}{\eta_T} \qquad [7]$$

Where α_{pp} is the *inverse power density* of the power plant and η_T is the thruster efficiency in converting electric to kinetic power. Usually, [7] would be defined as the maximum value of that expression in the interval of interest, since both α_{pp} and F_T vary with time. However, in both current models either both are constant, or their form makes the product a constant: in solar engines $F_T \sim r^{-2}$ and $\alpha_{pp} \sim r^2$, with r the distance to the Sun. Finally, the full propulsion subsystem consists of all the above terms, plus the propellant tanks, represented as a fraction κ_t of the spent mass:

$$m_{PL} = m_{pp} + (1+\kappa_t)\Delta m_p \qquad [8]$$

<u>Deflection by Kinetic Impactor</u>

The principle of a kinetic impactor is simple and well documented in the literature[16]: an object, termed the impactor, crashes at hypervelocity into the NEO causing an impulsive change in its momentum, possibly enhanced by ejected asteroid mass. The impactor properties are straightforward, but the cratering model has been the subject of thorough research[17], with the general consensus stating that both contributions to the momentum are aligned under certain conditions. The model describes the obtained deflection as:

$$\Delta \mathbf{v}_a = \beta \frac{m_{SC}}{m_a} \mathbf{v}_{SC} \qquad [9]$$

Where β is the *momentum multiplication factor*, m_{SC} is the S/C mass, m_a is the asteroid mass and $\mathbf{v}_{SC}$ is the relative S/C arrival velocity to the asteroid. β models the additional impulse caused by the mass ejection and is computed by RIMISET using a power law model based on research by Housen and Holsapple[18]:

$$\beta = 1 + \left(\frac{v_{imp}}{v_{esc}}\right)^{3\mu-1} \cdot f\left(\frac{Y}{\rho v_{esc}^2}, \frac{g R_{crater}}{v_{esc}^2}\right) \qquad [10]$$

The constant μ is between $^1/_3$ and $^2/_3$, and represents the absorption of part of the delivered energy by plastic deformations such as crushing of voids in the material. Porous materials exhibit lower μ, and thus lower β. The other parameters represent the regime of the impact: for smaller NEOs, the expelled material is governed by the material specific strength (Y/ρ) while for larger targets the dominant force is gravity, with $g \cdot R_{crater}$ representing a rough measure of the involved potential energy.

However, the authors find that, in most cases, such modulating functions are essentially constants because a large part of the momentum comes from the central region where the impact shock pressures dominate. Thus, practical models employ a

K value multiplying the impact-to-escape velocity ratio. The value of β computed by RIMISET uses the simplified formula described in the kinetic impactor section, with K and μ provided by the user and the escape velocity computed from the NEO mass and geometry data.
Finally, for inverse problems the determination of the computed impulse $\Delta\mathbf{v}_{obt}$ needs to ensure that the NEODET optimal impulse $\Delta\mathbf{v}_{req}$ is achieved. However, since the direction of the impact trajectory is fixed by the transfer, the target impulse is scaled so that its projection in optimal direction fulfils the requirement.

$$\Delta v_{obt}^{inv} = \frac{\Delta\mathbf{v}_{req}}{\cos\alpha} \cdot \frac{\mathbf{v}_{SC}}{v_{SC}} \qquad [11]$$

Where α is the misalignment between the impact trajectory and the optimal direction.

Deflection by Nuclear Blast

The obvious next step for impulsive deflections when other methods would not suffice is a nuclear blast. For this method the program employs the model described by Solem[19]. The detonation vaporizes part of the NEO, creating a large crater and causing mass ejection similar to the kinetic impactor.

According to the model, the ejected mass is defined by a power law of the released energy with two constants *A* and *B* (α and β in the reference), following experimental study of cratering processes. The global ejecta kinetic energy is modeled with another user-provided constant called the energy coupling constant Δ (δ in [19]). Thus, RIMISET computes the deflection as:

$$\Delta v_a = \frac{\mathrm{A}\Delta}{m_a} \cdot \left(\varphi m_b\right)^{\frac{\mathrm{B}+1}{2}} \qquad [12]$$

Where φ is the yield-to-mass ratio of the bomb. Solem presents this model as valid for any kind of nuclear blast deflection, with different constant values for a surface, buried or stand-off detonation.

The implemented nuclear method accepts both impact transfer trajectories and rendezvous transfers with the NEO: the former case is handled like the kinetic impactor, using [11] in the inverse case to compute the actual deflection target; while in the latter it is assumed that the spacecraft will position itself so that the resulting deflection is fully aligned with the optimal direction. Note that the variable used by the nuclear deflection method is not the S/C mass at arrival but only the mass of the bomb, that is, the mission payload. Unlike in the kinetic impactor, the structural mass does *not* help towards the deflection target. The transfer handling routines take this distinction into account and add the needed structural mass as required.

Deflection with an Ion Beam Shepherd

This slow-push deflection method was proposed as recently as 2011 and it uses a pair of nearly balanced thrusters to hover at a stable distance from the NEO, either leading or trailing it along its orbit. The object is hit by the exhaust plume of a thruster and is consequently pushed by it. The model put forward by Bombardelli and

Peláez[20] assumes that the spacecraft is far enough from the NEO for the mutual gravitational interaction to be negligible. This is a strong point of the design because it relaxes the stringent control requirements that are characteristic of gravity tractor designs, as noted in later sections.

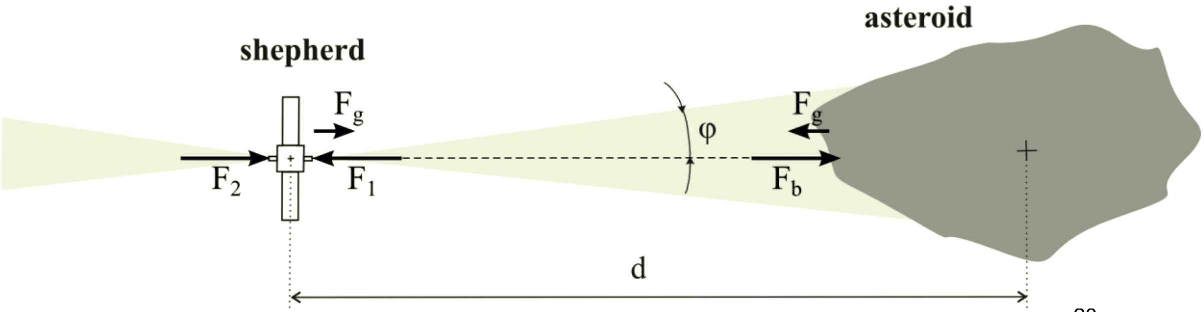

Fig. 1: Diagram of the ion beam shepherd concept (figure taken from [20])

The beam force is generated by ionizing interactions at the NEO surface that stop the incoming plume, thus absorbing its momentum with a very high efficiency $\eta_B \sim 1$. Thus, the total thrust employed by the spacecraft is:

$$F_T = \frac{2}{\eta_B} F_a(t) \qquad [13]$$

Where F_a is the same force profile given to generate the NEODET file; the force that the NEO is supposed to receive to produce the desired deflection.

The tool implements the ion beam shepherd by using [13] to find out the total thrust requirements, and then uses the engine model as defined in [6], [7] and [8] to either obtain the required payload mass (for inverse problems) or compute the maximum mission duration given the available payload mass (for direct problems). The use of the transfer routines bridges the gap between the Earth escape mass and the payload mass. Once the mission has been deemed viable from a payload mass standpoint, the geometric feasibility must be verified. The distance to the NEO must be large enough that the gravitational attraction is negligible; but small enough that the beam is fully intercepted:

$$\sqrt{\frac{\mu_a}{k_g}\left[\frac{m_{SC}}{F_T}\right]_{max}} \leq d \leq \frac{r_a}{\tan\varphi} \qquad [14]$$

Where k_g is an arbitrary small constant (1% in RIMISET) relating the gravitational and beam forces.

<u>Deflection with a Hovering Gravity Tractor</u>

Gravity tractor (GT) deflection missions are based on the idea of using a massive spacecraft as a contact-less tow-ship. This requires the S/C to be in close proximity to the NEO and operated by a low-thrust propulsion subsystem; typical distances in proposed designs are *at most* around 2-3 times the asteroid radius. At such a close range, two important problems arise:

1. The complexity of the collision avoidance and operational control systems increases, as most NEOs are not spheroidal but markedly triaxial.
2. The exhaust plumes from the thrusters must not impinge on the asteroid surface. A failure to keep this condition would result in transference of momentum to the NEO in the opposite direction, partially (or even completely) counteracting the desired force.

The latter concern has been addressed by two separate GT designs: hoverers and displaced orbiters. The first model, described in this section, was proposed by Lu and Love[21]. It puts the spacecraft either leading or trailing the target in its orbit, with a system of *n* symmetric thrusters exerting a total force F_T. The thrusters, which expel exhaust cones of semi-angle φ, are all canted away from the NEO orbit tangential direction at an angle δ (fixed at construction) to clear the NEO surface. The resulting configuration is shown in fig. 2 (left); note that the exact tangency of the thruster exhaust cones is *not* a requirement a priori. In the model, the instantaneous distance between the spacecraft and the NEO is:

$$d = \sqrt{\frac{\mu_a}{k_T \cos\delta} \frac{m_{SC}(t)}{f(t)}} \quad [15]$$

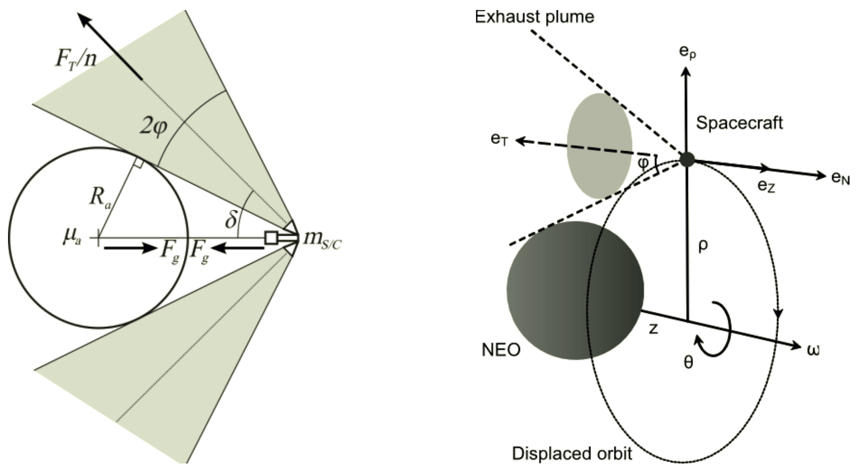

Fig. 2: Hovering (left, as from [21]) and orbiting (right, as from [22]) gravity tractors in the tangent exhaust configuration

The presence of the canting angle δ in this formula (and most others) is problematic because it couples the system together. In fact, the applied force profile determines the reference value for the *useful* thrust $k_u = k_T \cos \delta$, instead of k_T itself. This carries over to the mass equations [6] and [7], which require the total thrust and so are left depending on the choice of δ. As a solution, RIMISET considers that in the worst case (minimum relative distance), the thruster exhaust cone will be *exactly tangent* to the NEO outer surface:

$$d_{\min}^{geom} = \frac{R_a}{\sin(\delta - \varphi)} \quad [16]$$

With the above, the tool forms an equation for δ by holding that the minimum distance given by the geometric constraint in [16] must match the actual minimum of the trajectory in [15]. Thus, [15] equals [16] at a time $t = t^*$ to be determined.

The other constraint in the system is mass-related: the payload mass and the mission running time are linked by [8]. The problem is solved with a single variable (which is T for direct or m_{PL} for inverse problems) used to drive a bisection solver over the geometric equation. The value of δ for each iteration comes from applying the requested force profile and the current value of the bisection variable to the payload mass definition, which in the case of gravity tractors includes an additional deadweight to be determined. The value of this deadweight is such that the total arrival mass is minimal ($dm_{Arr}/d\delta = 0$, or the lowest mass allowed by user-set constraints like the minimum distance to the NEO).

Finally, analysis of [15] shows that the minimum distance occurs at the time t^* with the minimum m_{SC}/f ratio. It is straightforward for missions with a constant-thrust engine: the S/C mass decreases linearly and $f = 1$, so the minimum separation occurs at $t^* = t_f$. However, the use of a solar-powered engine complicates the problem of finding t^* significantly. Differentiation of d in that case produces an equation for t^* that depends on the distance and radial velocity to the Sun. The tool sweeps the interval of interest, looking for the global minimum of the mutual distance by detecting changes of sign in that equation and computing the distance to compare between two local minima.

Deflection with an Orbiting Gravity Tractor

The second GT design[22] (see fig. 2 right) avoids the canting of the S/C thrusters, instead establishing a displaced circular orbit of radius ρ and displacement z around the NEO so the secular gravitational force is in the same direction as before. The design choice in this case consists of keeping the exhaust cone tangent to the NEO outer radius at all times, which produces:

$$
\begin{aligned}
R_a &= \rho\cos\varphi - z\sin\varphi \\
F_T &= \mu_a m_{SC}\frac{z}{\left(\rho^2+z^2\right)^{2/3}}
\end{aligned}
\qquad [17]
$$

The first equation, which represents the thruster cone tangency condition, links the orbit radius and displacement. Since the force profile to apply is known a priori ($F_T = F_u$), the size of the power plant and the amount of propellant needed can be known just from the thrusting time, like the ion beam shepherd. The arrival mass, however, needs to be topped up with some deadweight like in the hovering GT in order to provide the required force level in all cases.

Eliminating ρ from the expressions in [17] results in an equation for the trajectory, that is, it links z and the instantaneous mass m_{SC}. As the latter is simply the arrival mass minus any spent propellant, this can be the basis of an equation for m_{Arr}. In order to obtain the arrival mass, this equation is applied at the critical point z* of minimum distance, and is completed with either the condition of minimal arrival mass ($dm_{arr}/dz^* = 0$) or the minimum NEO distance constraint ($\rho^{*2}+z^{*2} \geq d_{min}^2$).

Analysis of the constraints reveals that the existence of a solution for a given point in time is determined by the value of the gravitational parameter $\mu_a \cdot m_{SC}/F$. As displayed on fig. 3, this dependency is monotonic, so the worst case that is being sought will occur when this parameter is smallest. This reduces to finding the minimum m_{SC}/F, so RIMISET uses the same code as in the hovering GT to find the worst instant t^*.

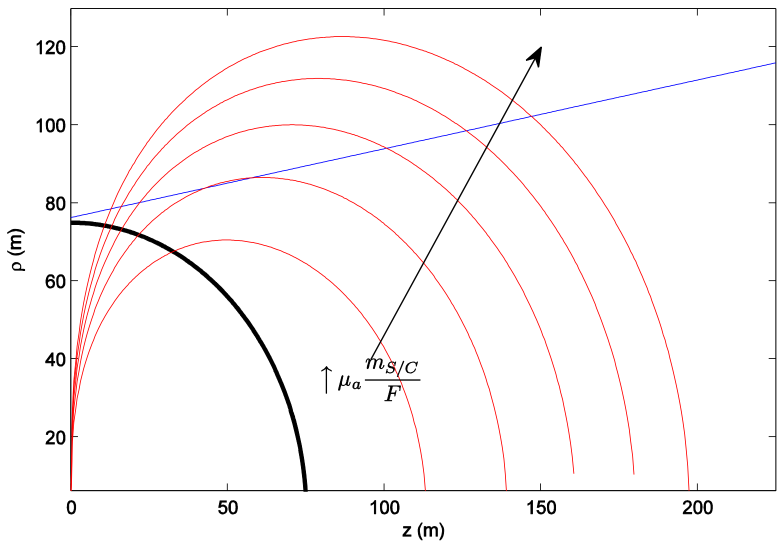

Fig. 3: Geometric constraints in the orbiting GT

AN END-TO-END USE CASE FOR THE HYPOTHETICAL NEO 2015 PDC

In this section, we present an application case of the suite of tools to the hypothetical NEO that was proposed for the conference, i.e. 2015 PDC[23,24]. We find out the impact characteristics with NIRAT, compute deflection requirements with NEODET and estimate feasible missions with RIMISET using Earth-NEO transfers computed with an external tool. Previous to the analysis of the mitigation opportunities we analyze the possibility of reaching the target asteroid by rendezvous missions and by impact missions.

Summary of NEO Information

Orbital information on 2015 PDC most likely trajectory is initially known from JPL provided data[24]. Little information is known on the asteroid features among which the relevant uncertainty in the NEO size and composition means that the mass of the target is very poorly determined: the lightest case can be 250 times less massive than the heaviest (see Table 1).

In any case, the large mass of the worst-case NEO is a severe complication that compounds with the short length of time available that will likely rule out complete deflections by slow-acting methods such as an ion-beam shepherd. The combination of these circumstances considerably complicates any effective deflection effort, no matter what trajectory or deflection action is taken.

	Light case	Heavy case
Equivalent diameter (m)	100	500
Mean density (kg/m^3)	1500	3000
Total mass (kg)	$7.85 \cdot 10^8$	$1.96 \cdot 10^{11}$

Table 1: Extreme mass values compatible with the JPL data

Rendezvous Missions to 2015 PDC

Due to the fact that the target NEO will be mostly non-visible in the time frame between late 2015 and early 2022, the only way to ascertain whether the asteroid is going to collide or not with Earth before it is too late to react is by sending a sounding probe as early as possible. This means that any launch opportunity has to be profited as soon as a sounding spacecraft is ready for launch. Even if a flyby mission could be enough to determine some of the required information, the most interesting mission will be that of rendezvousing with the target in order to get as much information as possible on the asteroid physical properties, further than just on its orbit data.

Rendezvous missions to the target asteroid have been investigated by means of a ballistic branch-and-prune algorithm combining swingbys with Venus, Earth and Mars (up to a maximum of two) before arrival to the NEO. No assumption is made on the launcher vehicle other than it can provide a roughly constant maximum C_3 of 10 km^2/s^2, so the mission ΔV is computed as the sum of the remaining escape excess velocity plus the intermediate maneuvers and the arrival burn to the asteroid. It is assumed that the same engine that will provide the operating phase thrust will be used for the additional escape ΔV.

The transfers to the asteroid have been analyzed from mid-July 2015 until close to the possible impact with Earth in 2022. A time grid on the departure date, swing-by dates and arrival date has been chosen to investigate the solution space. The following plots summarize the obtained information:

- Fig. 4, which shows the map of departure dates vs. arrival dates for the reported trajectories, while Fig. 5 depicts the total flight time for them. Feasible arrivals are more frequent close to the pericentre, although the swing-bys provide significant leeway in the arrival date
- Fig. 6, which displays the total transfer Δv for such trajectories. Trajectories with swing-bys provide a marked improvement over the direct transfer arcs, achieving transfer costs in the order of 6 km/s. However, many of these good transfers arrive less than one year before the impact
- Fig. 7 and Fig. 8, which plot the Earth departure and NEO rendezvous velocities, respectively, for the reported trajectories. Many trajectories allow reaching the NEO near the 2017 and 2020 pericentre passages at relative velocities of 2–3 km/s, and even lower values in 2022

A closer look at the solutions that are around or before the first pericentre (in late 2017) reveals a trade-off between two families: the ones departing in 2016 require a significantly larger rendezvous Δv_{arr} (at least 3.5 km/s) than those launched in 2015 (Δv_{arr} from 2.0 km/s). However, a launch in 2015 would take place only months after the NEO discovery, which is very challenging in terms of spacecraft availability (even for the 2016 launch case).

Therefore, the earliest arrival to the asteroid could take place at the end of 2017 with still five years to take any action against the asteroid. If the 2016 launch opportunity is missed, then the next arrival series would be the ones of the 2020 perihelion pass (from mid 2019 to mid 2020) with three years at most for action.

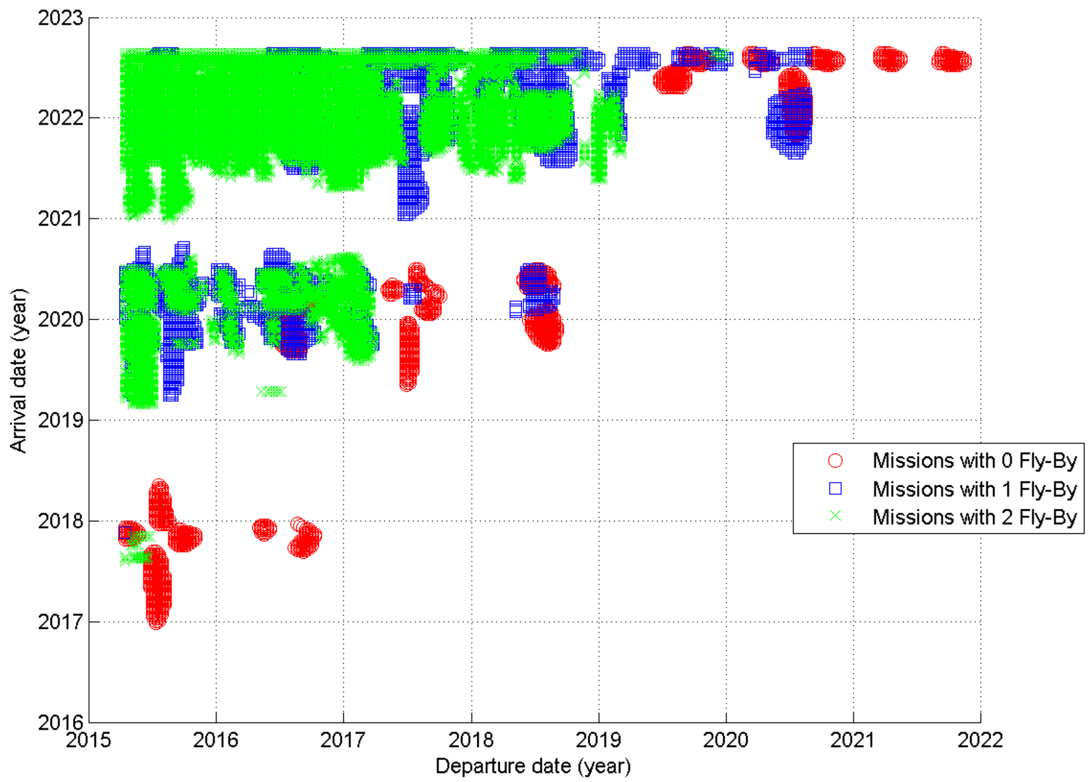

Fig. 4: Dates of departure and arrival of rendezvous missions with swing-bys

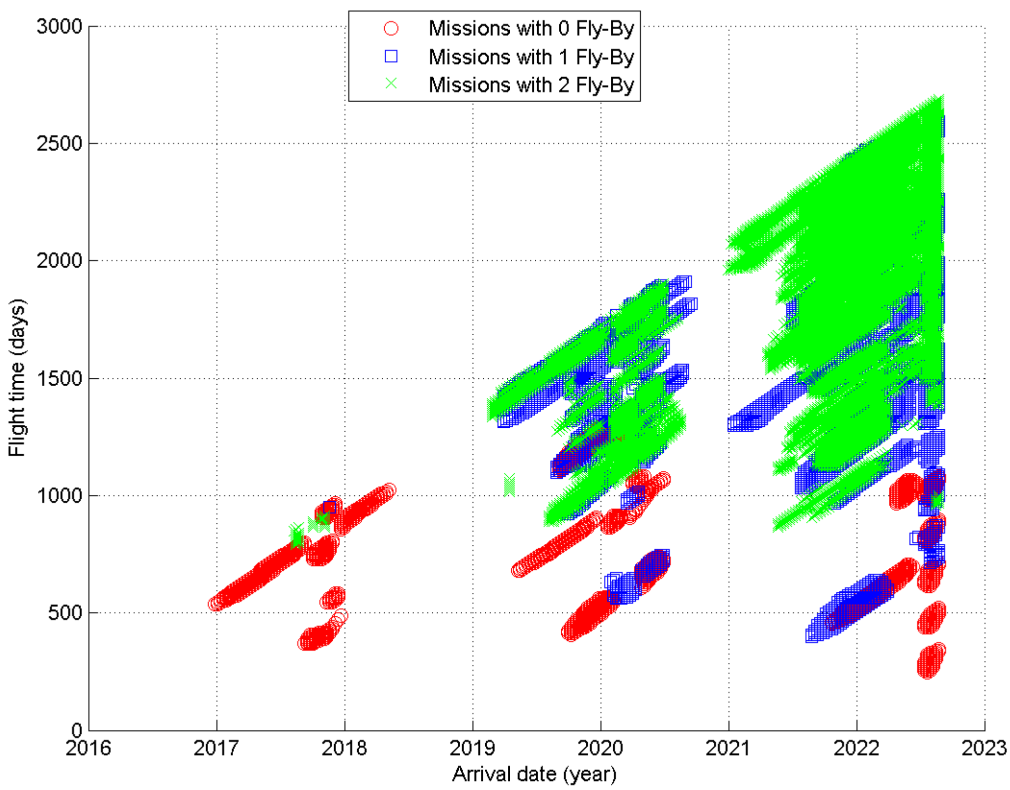

Fig. 5: Flight times of rendezvous missions with swing-bys

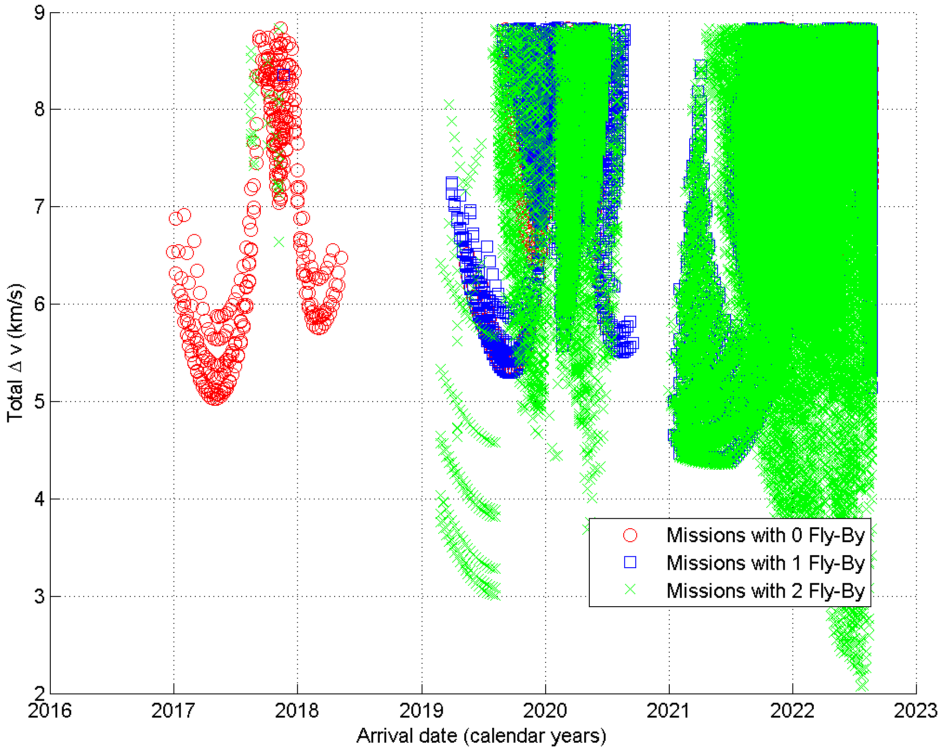

Fig. 6: Total transfer Δv of rendezvous missions with swing-bys

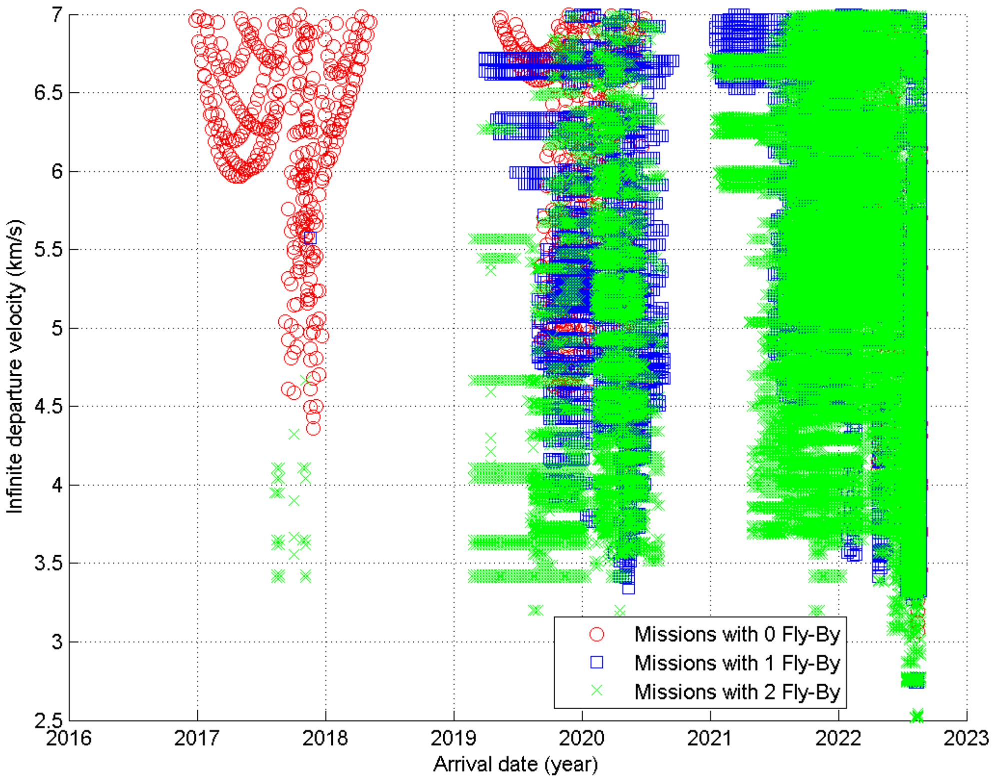

Fig. 7: Earth departure velocity of rendezvous trajectories with swing-bys

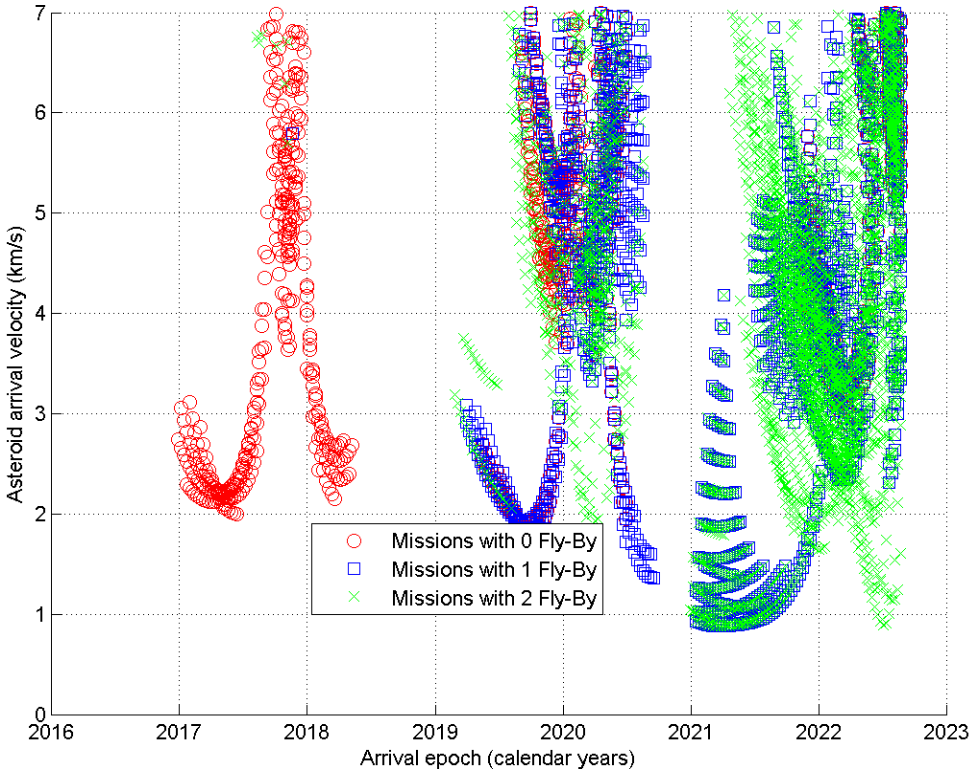

Fig. 8: NEO arrival velocities of rendezvous trajectories with swing-bys

Rendezvous missions can already provide means for deflection if so designed for, in addition to allowing a detailed characterization of the orbit and properties of the asteroid. For example, they could carry a nuclear device for a stand-off explosion or act as a means of slow-push by means of ion-beam shepherding, laser beaming or gravity towing. In these last cases, low-thrust transfers to the asteroid would probably be the most applicable transfer options (such transfers are not analyzed in this paper).

Impact Missions to 2015 PDC

In case a high relative velocity mission is needed for the mission (e.g. for kinetic impact solutions or nuclear explosive devices detonated at large arrival velocities), ballistic transfers to the asteroid without rendezvous need to be investigated.

The results of a similar branch-and-prune algorithm over this type of transfer as for the rendezvous case are depicted in next figures including up to one swingby:

- Fig. 9 shows the date map of departures and arrivals for the reported trajectories, while Fig. 10 depicts the total flight time of such trajectories. Like in the rendezvous case, arrivals cluster mostly around the NEO perihelia
- Fig. 11 displays the Earth departure velocities for the impactor trajectories. Earth escape velocities as low as 0.7 km/s are available
- Fig. 12 shows the obtained impact velocity for trajectories with and without a swing-by. Both types of trajectories provide options to reach the asteroid at a relative velocity of up to 15 km/s
- Fig. 13 represents a figure of merit (FOM) that has been constructed to approximate the effect of the transfer trajectory on the expected deflection performance. This FOM is the obtainable NEO tangential deflection Δv_a,

multiplied by the NEO-to-S/C mass ratio for each case, which is a value that could conceivably range between 10^4 and 10^8. The latter ratio uses the S/C mass at launch, thus the figure already considers the mass loss in the trajectory. Most trajectories in the set exert a negative tangential deflection, thus slowing the NEO down

- Finally, Fig. 14 plots the Sun-NEO-S/C angle at arrival, thus representing the illumination conditions that an impactor S/C would face for its terminal navigation phase. In principle, the closer to zero, the better, but even angles larger than 90 deg could be accepted under certain conditions such as the previous presence of an orbiter that could provide a tracking signal. Missions with angles larger than 145 deg have been disregarded altogether

This type of impact missions could be considered in two different schedule combinations with the rendezvous missions. In cases where there is time enough for any mitigation action, one could think of launching the impact mission after arrival of the rendezvous spacecraft to the target and once this has determined whether the impact with Earth is certain or not. However, in cases when the time available for mitigation is very tight, as in the case of 2015 PDC, the international mitigation community might decide to advance the launch of the mitigation mission before the first probe has arrived to the asteroid such to advance any mitigation response. If the impact can be ruled out once the first probe is at the NEO, then the second mission can be made to avoid action, and contrary to that, actually act in case the threat is confirmed.

As seen in Fig. 9 the impact missions can be launched any time in the considered period with arrival times around the asteroid perihelion passes at end of 2017, start of 2020 or mid 2022. As it will be shown later, the earlier the impact the most effect on the asteroid trajectory.

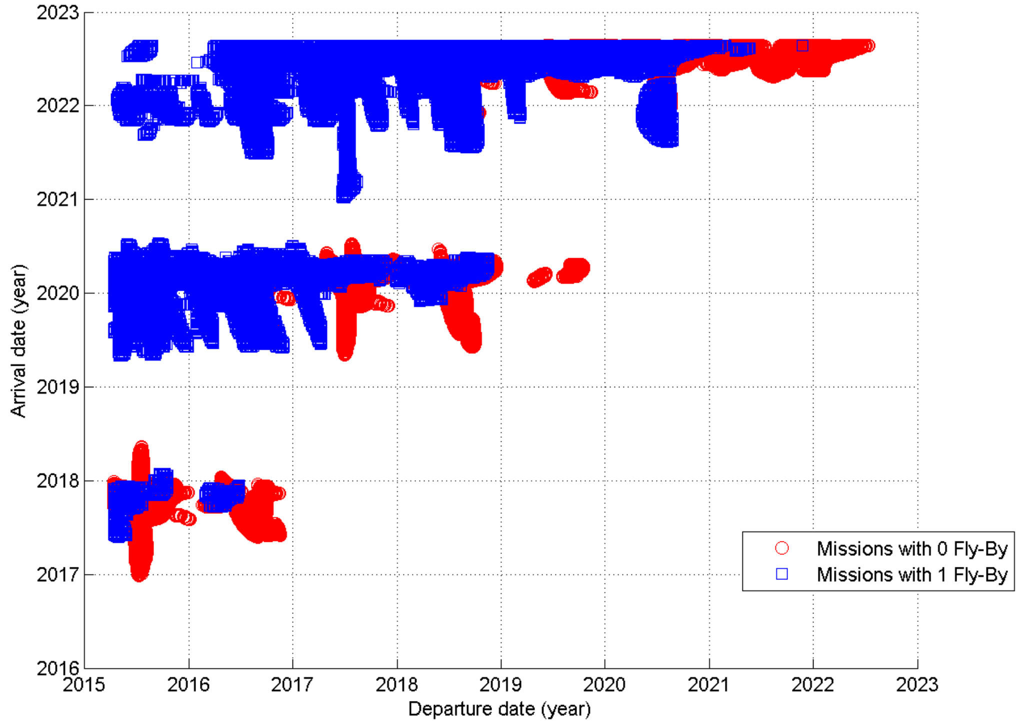

Fig. 9: Dates of departure and arrival of impactor missions, with & without swing-bys

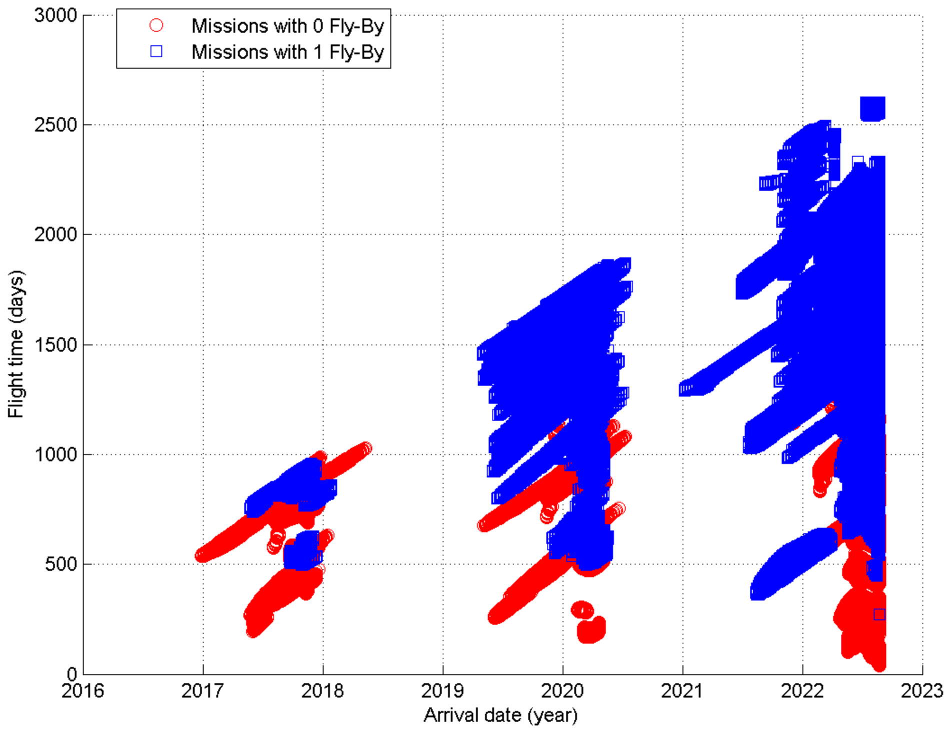

Fig. 10: Flight times of impactor missions with and without swing-bys

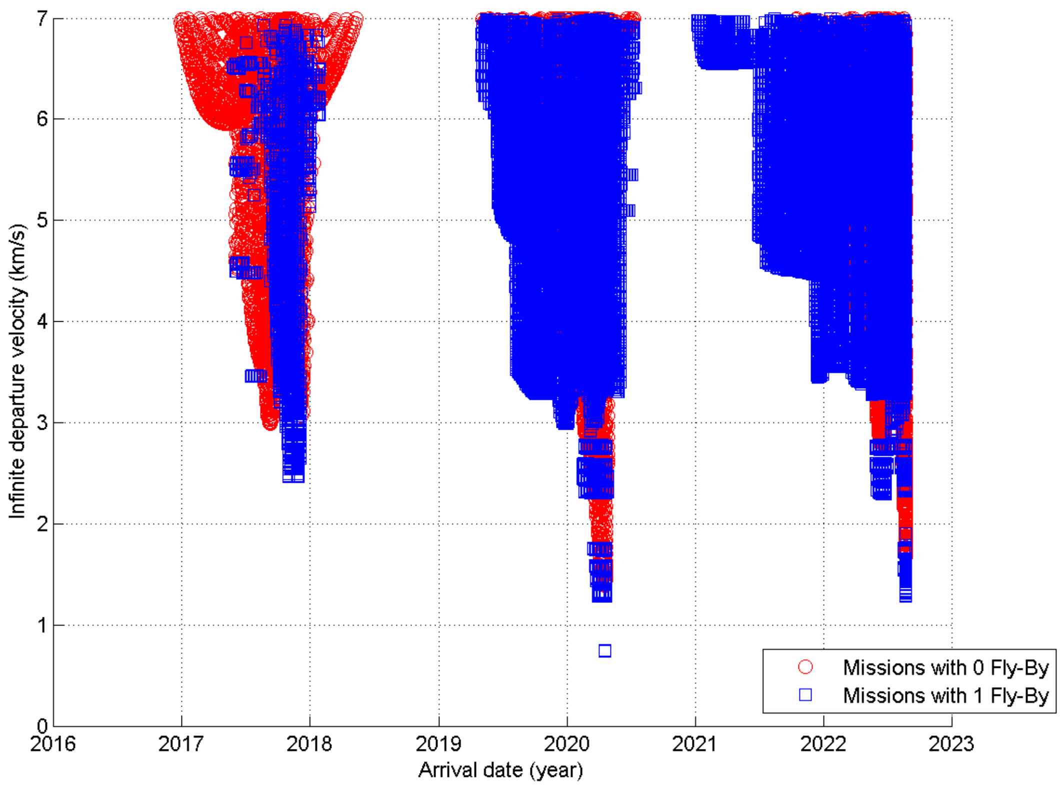

Fig. 11: Earth departure velocity of impact trajectories with and without swing-bys

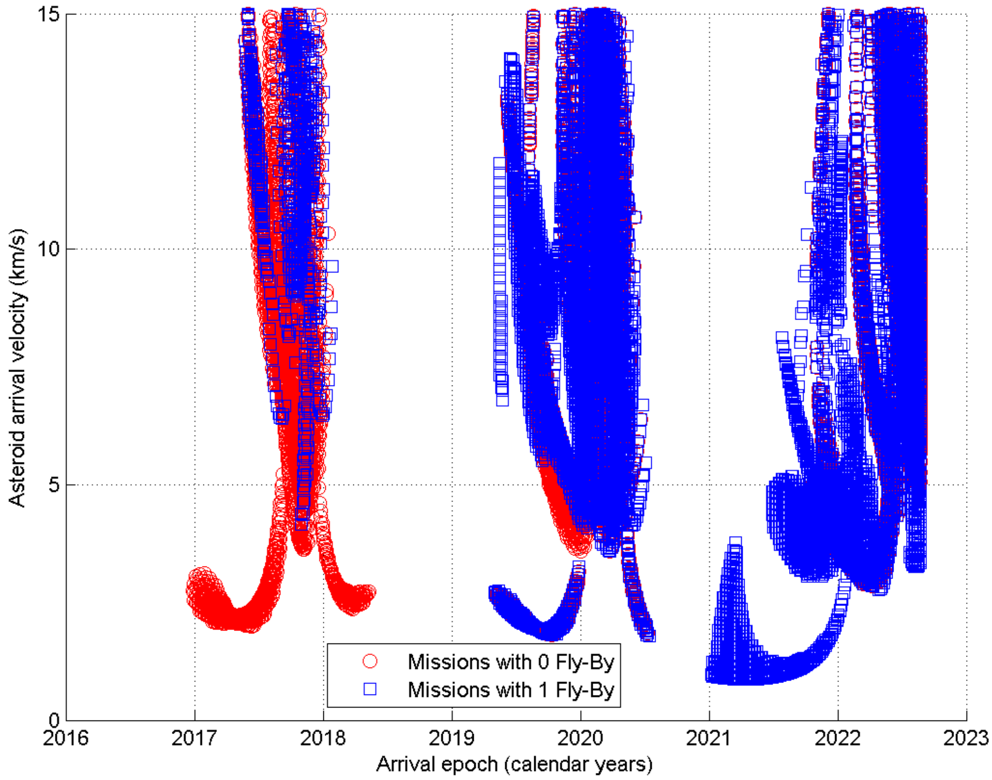

Fig. 12: NEO arrival velocities of impact trajectories with and without swing-bys

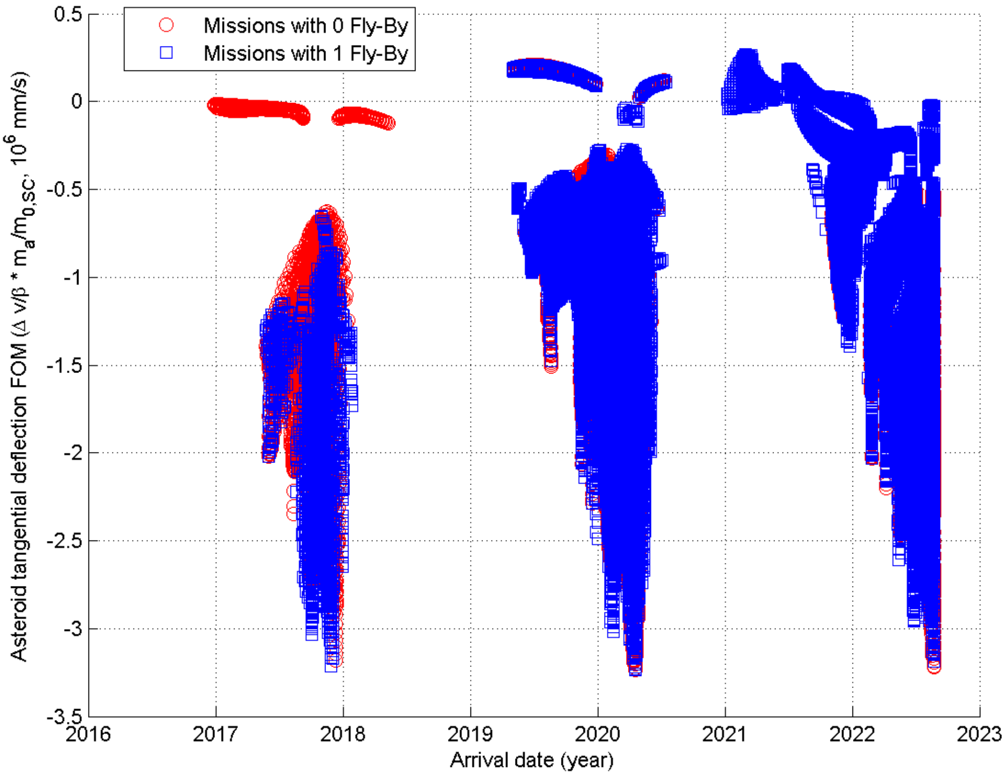

Fig. 13: Impact figure of merit (notional Δv independent of NEO and S/C masses)

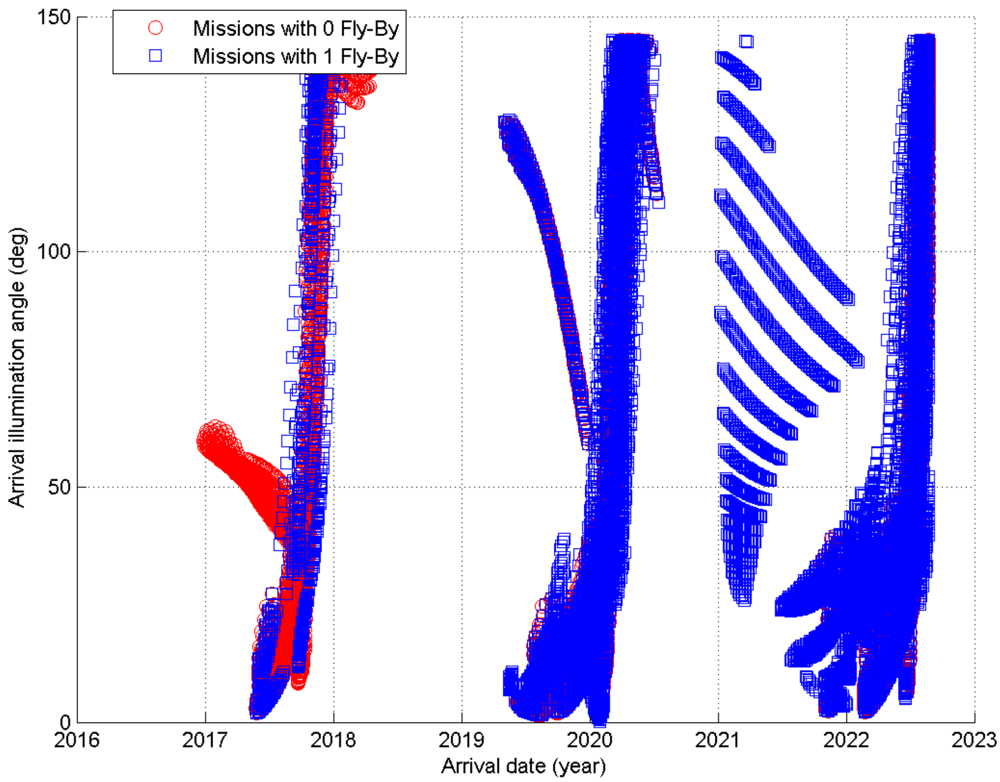

Fig. 14: Sun-NEO-S/C angle on arrival of trajectories with and without swing-bys

<u>NIRAT: Predicted Impact Details</u>

The information provided by JPL does not include the initial uncertainty matrix, so the possibility of replicating the full b-plane plot in NIRAT is curtailed. However, the single initial condition given was introduced into NIRAT for a detailed analysis of the forecast impact circumstances. The employed initial conditions were obtained through the Horizons system:

- Reference frame and epoch: ICRF, JD2457125.5 = 2015-Apr-13 00:00:00 (CT)
- Position: (-1.531026557357406e+8, -8.955183563006687e+7, -5.402472874782888e+7) km
- Velocity: (2.593397015055142e+1, -1.452683659248620e+1, -6.957773322866355e+0) km/s

Propagation settings included point-mass gravity fields of all the major planets, plus the Moon, Ceres, Vesta and Pallas, along with the first-order expansion of the relativistic effects of the Sun. Fig. 15 shows that NIRAT was able to replicate the impact, although the minimum orbital intersection distance (MOID, ξ coordinate in the b-plane) is about 25 km off the line traced by the JPL impactors. This difference is at over 6 times the ξ variation shown in the JPL VI set. However, it amounts to less than 0.3% of the scaled Earth radius; see the blue square against the red line of JPL impactors.

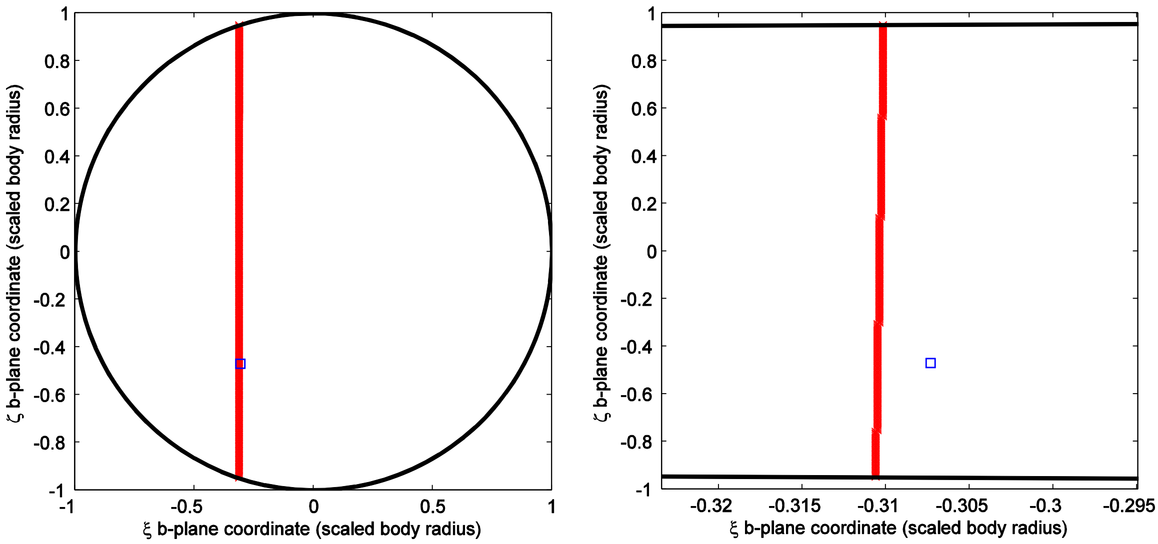

Fig. 15: B-plane coordinates of the provided virtual impactors

NEODET: Dynamical Requirements for Deflection

The presented hypothetical NEO is problematic for any deflection method, as stated above, mainly due to the short warning time from discovery to impact. This implies that secular effects from e.g. a change in the asteroid semi-major axis will have little opportunity to accumulate and amplify the effect of any applied deflection. For instantaneous methods mitigation will imply large actions to actually achieve the required deflection.

Impulsive Deflections

In addition to the mentioned lack of time for the deflection to accumulate through the secular drift in the true anomaly, the absence of other planetary close encounters in the NEO trajectory from 2015 up to the predicted impact in 2022 prevents any deflection mission from profiting from the deflection amplification effect that these close encounters often produce. In order to assess the feasibility of a complete deflection – making the NEO miss Earth completely – we asked NEODET to generate specifications for a 2 Earth radii deflection. That way, even though the virtual impactor used in the computations corresponds only to the blue square in Fig. 15, every other impacting trajectory should be safely deflected off the impact course with Earth.

The combination of the close proximity of the impact event (7 years from the discovery) and the mentioned lack of any other close encounters that could have an amplifying effect on the applied impulse places a heavy requirement on any deflection missions, see Fig. 16. This figure provides the delta-V requirements for a full one Earth diameter deflection (mapped in the b-plane), where it can be observed that for most of the time (with the exception of the last months prior to collision with Earth) a tangential action on the asteroid is the most effective mitigation action. Delta-V needs on the asteroid would range between 2 cm/s to 10 cm/s for most of the time. Discarding the mid-2015 Δv minimum as unrealistic for any mission to reach due to the small preparation time of only a few months, the first opportunity appears in late 2017 and requires an impulse of nearly 2 cm/s. The next (and final) minimum shows up in early 2020, and a mission targeting the NEO at that point would need 4 cm/s to completely deflect it off its collision course. From 2021, actions in the direction perpendicular to the tangential direction start gaining importance while the overall required action starts rising sensibly. As expected, only in the last weeks before collision acting in the direction perpendicular to the NEO orbit would be an option.

Even if the required delta-V values are large, these are not in principle unfeasible and so the feat might be possible depending on the available Earth-NEO transfers, the spacecraft mass and mainly the NEO mass, which are explored in the RIMISET section.

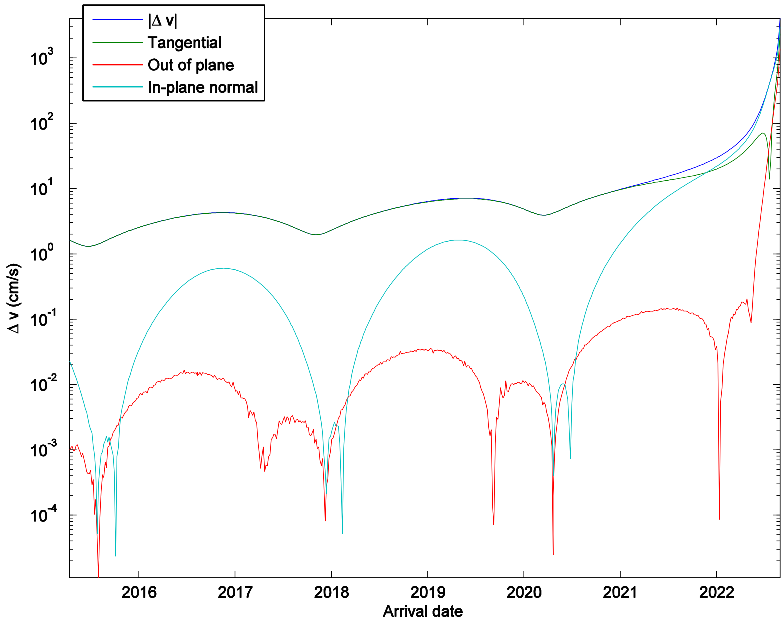

Fig. 16: Optimal impulsive deflection for a 2 R_E deflection of the virtual impactor

Continuous Deflections

As mentioned before, the short time from the NEO discovery to the forecast impact does not permit a full deflection by means of slow-push methods, that is, an action that would move the trajectory of the proposed asteroid off its collision course with Earth. However, the possibility remains to use the slow-acting deflection methods to at least impart a partial deflection that would at move the forecast NEO impact point in order to avoid particularly populated zones.

The 2015 PDC virtual impactor that is described by the initial conditions given by the Horizons system would fall in the South China Sea, close to the Philippines and Hong Kong. However, the trace of virtual impactors offered by NASA covers even more populated areas, going dangerously close to several Indian cities or to Tehran. We have considered a possible deflection along the ζ direction on the b-plane, which corresponds to the phasing of the Earth encounter event. Then, moving the impactor along that direction on the b-plane without achieving a complete deflection roughly amounts to “sliding” the impact point along the red VI trace. Thus, political considerations would be of the utmost importance, since the risk area would literally crawl across the surface of the Earth while the deflection mission is in progress.

Two deflection scenarios are considered: 100 and 1000 km from the impact point, which have been correlated using the b-plane data in the JPL VI impactor list to b-plane deflections in the ζ direction. The deflection specifications have been generated using NEODET for a hypothetic spacecraft that would generate a tangential force on the NEO of 250 mN or 500 mN using an engine of I_{sp} ~ 4200 s, characteristics that are similar to those planned for the NASA Evolutionary Xenon Thruster[25]. Table 2 contains the resulting b-plane deflections and last possible dates of arrival to the NEO for a valid deflection with that level of thrust exerted upon over the asteroid, considering two alternative thrust models: a constant force and a force that scales with $r_{Sun}^{-1.7}$. The former may represent a thruster powered by a nuclear reactor, while the latter models a solar-powered engine with the stated level of thrust occurring at a distance of 1 AU.

NEO size/deflection case	Small/Small	Small/Large	Large/Small
Ground deflection (km)	100	1000	100
b-plane $\Delta\zeta$ (km)	±107	±1112	±107
Net thrust applied (mN)	250	250	500
Last possible arrival date (const. thrust)	Jun/2022	Apr/2021	Oct/2017
Last possible arrival date (thrust~$r_{Sun}^{-1.7}$)	Jul/2021	Apr/2020	Jul/2015

Table 2: Required deflections for slow push methods

Results are shown graphically on Fig. 17, which represents the required thrust time for the considered deflections. In these cases the deflection action is always assumed to occur in the direction of the asteroid velocity about the Sun. The relatively large force values applied to the small NEO result in small required thrust times of only a few days for the case corresponding to the lesser deflection of 100 km. On the other hand, the large NEO needs to be deflected for years (with twice the thrust) even for the smallest displacement. Solutions disappear at the black line, which represents that thrusting action would be required right until the impact date. For example, in the case of the largest NEO and the 100 km deflection with a constant thrust the action would need to start before 2018. The solar-powered S/C would show larger oscillations of the push time, depending on the rendezvous point in the NEO orbit.

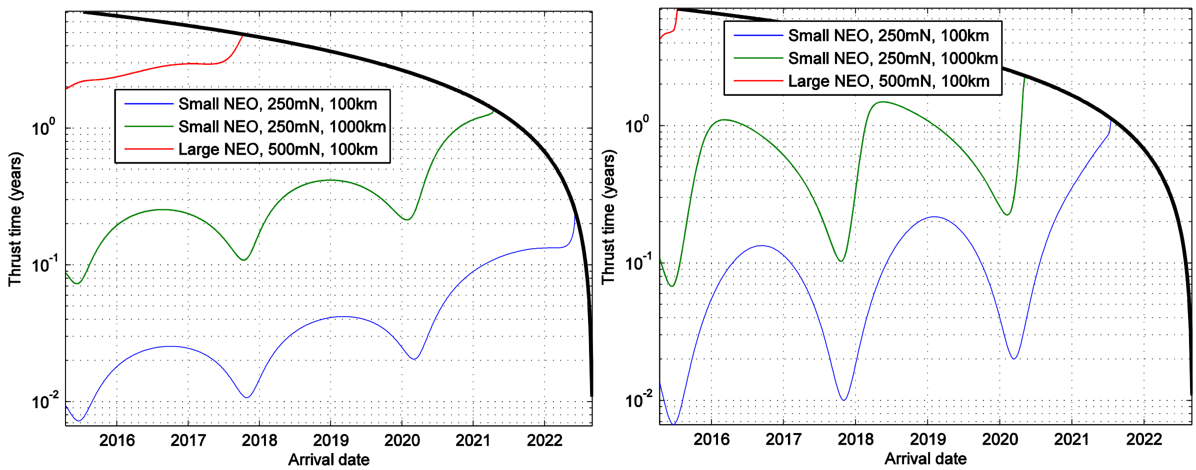

Fig. 17: Required thrust time for continuous partial deflections (left: constant thrust, right: solar thrust)

RIMISET: Mission Requirements for Deflection

In order to apply the third and final tool in this mitigation mission design end-to-end process, viable transfers from Earth to the target need to be considered in addition to the deflection specifications generated by NEODET. These computations are the ones that have been already presented in previous sections both for rendezvous and impact missions.

Impulsive Mission Requirements

With the above trajectory information, if the largest NEO is hit by a kinetic impactor with a large mass of 10,000 kg, crashing into the target at 15 km/s and profiting from a momentum multiplication factor (β) of 5, the resulting target Δv would be in the order of 4 mm/s. This value is *5 to 10 times smaller* than the minimum requirements described in the NEODET section (see Fig. 16), remarking that the NEO mass extreme case would be a very demanding scenario.

A preliminary analysis based on the transfer options above can use the impact figure of merit of Fig. 13, for which a pessimistic value of the momentum multiplication factor β = 1 was used. The NEO-to-S/C mass ratio thus covers a wide range from just below 10^5, for a large S/C of 10,000 kg and the smallest NEO, to 10^8 for a "small" S/C of 1,000 kg and the heaviest asteroid case. The value of the achievable tangential deflection will range through five orders of magnitude, see Table 3.

Deflection case	Best	Worst
S/C Earth escape mass (kg)	10^4	10^3
NEO mass (kg)	$7.85{\cdot}10^8$	$1.96{\cdot}10^{11}$
NEO-to-S/C mass ratio	$7.85{\cdot}10^4$	$1.96{\cdot}10^8$
Momentum multiplier β	5	1.5
Max. tangential Δv (cm/s)	-204	-0.0245

Table 3: Maximum achievable deflections for the best and worst cases

The more detailed analysis performed by RIMISET makes use of the transfer conditions to the asteroid and models the hypervelocity impact behaviour captured in the β parameter, following the theories put forth in [18], where the authors recommend values for the model parameters K and μ for several materials. The analysis described here used a matrix of four cases: two mass cases with the smallest and largest NEO, and two different target materials with constants that are chosen to match the "river rock" and "sand" materials from the figure at the end of [18], see Table 4 for details on the model constants.

NEO material	"Sand" (porous NEO)	"River rock" (solid NEO)
Log-slope μ	0.5	0.667
K factor (light NEO)	$3.23{\cdot}10^{-3}$	$4.90{\cdot}10^{-5}$
K factor (heavy NEO)	$8.55{\cdot}10^{-3}$	$3.36{\cdot}10^{-4}$
Momentum multiplier β	1.4-2.5	1.6-12.0

Table 4: Kinetic impactor model constants and resulting β for the RIMISET analysis

As an additional consideration, the delivery of a nuclear bomb for a stand-off explosion has also been simulated in both cases, for which RIMISET uses the model described by Solem[19]. Unlike the case with the kinetic impactor, only part of the final S/C mass (that of the bomb itself) participates in the deflection, so the program has been configured to consider an additional arrival S/C mass of 200 kg on top of the required bomb mass. The model constants employed are those quoted by Solem in the reference for stand-off nuclear blasts: the cratering factor A and exponent B take the values $1.5 \cdot 10^{-6}$ $(cm/s)^{-1}$ and 1 respectively, while the energy coupling factor $\delta = 0.3$ implies that 4.5% of the bomb energy is converted to ejecta kinetic energy. Finally, the bomb yield to mass ratio is assumed to be near 1 kton of TNT per kg of bomb assembly.

Results are shown on Fig. 18, where the left side represents deflection attempts for the light NEO case and the right side those for the heavy case. For each launch date, the plot shows the minimum initial mass of a kinetic impactor mission that will deflect the target a full one Earth diameter. Missions are chosen from the range of impacting trajectories presented above. Points are dots if the deflection is considered feasible; circles if it is feasible but carries the risk of fragmenting the NEO or Xs if the deflection is not achievable with a reasonable initial mass of 10,000 kg. Fig. 19 shows that such missions arrive mostly around the NEO perihelion.

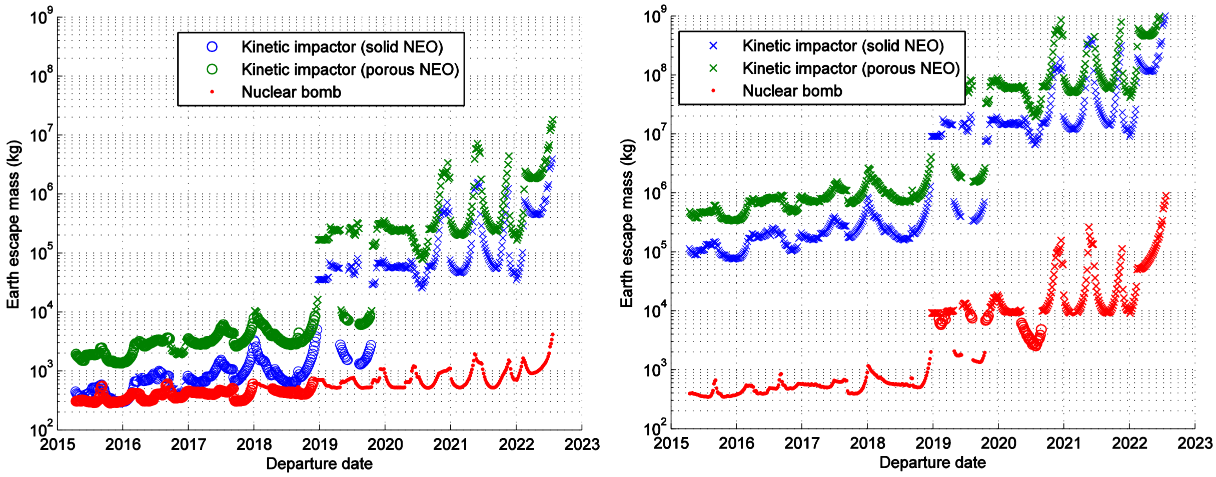

Fig. 18: Best single-shot impulsive deflection requirements by date of departure for the smallest (left) and largest (right) NEOs

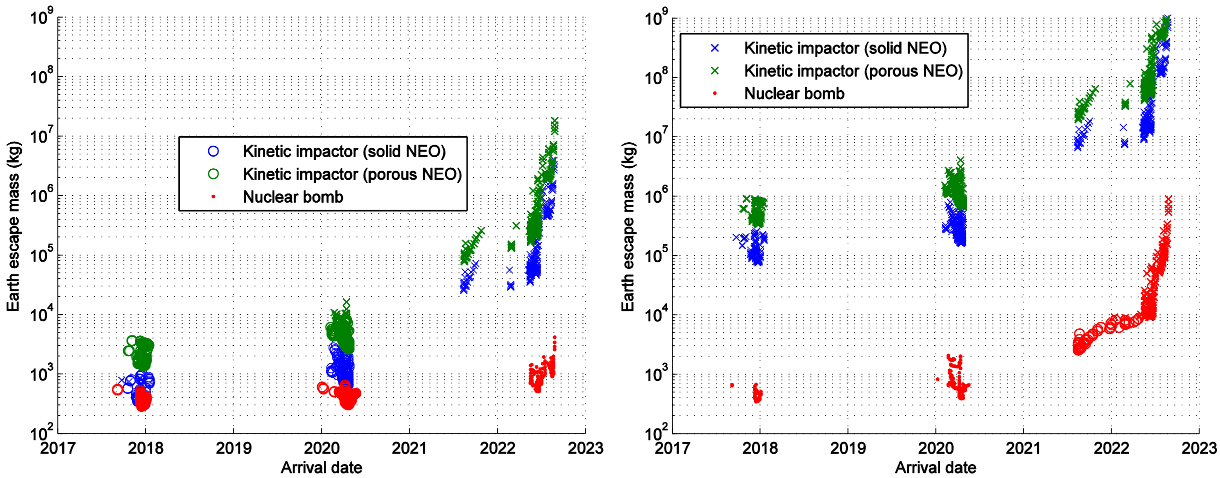

Fig. 19: Corresponding arrival dates of the previous missions for the smallest (left) and largest (right) NEOs

The differences between the respective solid/porous cases are significant, close to an order of magnitude due to the large change in the β values, which range from 1.4

to 2.5 (depending on the impact velocity) in the porous case but from 1.6 to 12 for the “river rock” solid NEO. It is also evident that the heaviest NEO cannot be deflected with a single kinetic impactor shot because the initial mass requirements are over 80,000 kg. In addition to the nuclear bomb, other options could include multiple kinetic impactor launches.

On the other hand, as predicted in Table 3, there would be plenty of opportunities for the lightest NEO to be deflected with even a small mission of initial mass in the 300–500 kg range. The problem in this case lies in the possible fragmentation of the NEO: the induced Δv is in the same order of magnitude than the surface escape velocity of the target, which according to Gennery[26] is likely to cause full-scale fragmentation of the target *without* causing its full disruption and dispersal. This eventuality may subject Earth to a cloud of debris – basically shrapnel from the partial destruction of the NEO – that could be even more dangerous (due to the much larger threatened area) than a single, bigger impact.

In conclusion, the presented asteroid would not be a good candidate for kinetic impact deflection for all options: the large uncertainty in its size, which according to the visibility data is not likely to improve in the short term, represents a significant problem for mission planning in this case. If subsequent orbital determination ends up confirming the threat, delivering a nuclear payload to the target seems the most effective method of avoiding the impact, either by completely disrupting the NEO if the mass is in the lower range, or by acceptably deflecting it if in the higher mass band.

Slow-push Mission Requirements

The results from NEODET, which include only partial deflections, were introduced in RIMISET along with the rendezvous trajectories. The program was configured to ascertain the viability of using an ion-beam shepherd mission equipped with two counter-acting NEXT ion engines with the following characteristics:

Net thrust on NEO (N)	0.25
Beam divergence (deg)	10
Maximum momentum transfer efficiency η_B	90%
Thruster efficiency (electric-to-kinetic)	70%
Power plant inverse power density (kg/kW)	15
Xenon tank mass fraction	7%
S/C structural mass (kg)	600

Table 5: Ion-beam shepherd analysis parameters

In addition, the viability of using a hovering or orbiting gravity tractor was also investigated. The main characteristics of either model are outlined in Table 6, which shows that the thrusters and power plant values are the same as those used in the IBS. Note that the structural mass is now just a minimum, since the program may add extra deadweight to the S/C in order to obtain the required amount of gravitational force.

Net thrust on NEO (N)	0.25
Thruster exhaust divergence (deg)	10
Thruster efficiency (electric-to-kinetic)	70%
Power plant inverse power density (kg/kW)	15
Xenon tank mass fraction	7%
Minimum S/C structural mass (kg)	600
Minimum distance to the NEO (m)	100
Number of thrusters (hovering GT only)	2

Table 6: Hovering gravity tractor analysis parameters

Results on Fig. 20 show that the ion-beam shepherd is able to provide valid deflections in both the small and large displacement cases for the small NEO case, although in a significantly reduced date range compared to impulsive deflection methods. However, the small deflection of ~100 km on Earth surface remains possible until very late, arriving at mid-2021 for the solar-powered S/C or even early 2022 for the nuclear-powered mission. This circumstance allows a last-ditch mission to avoid a direct impact of a small NEO case on a populated area even after other missions like a kinetic impactor fail.

The masses for gravity tractor missions are infeasible in every case for both the hovering and orbiting GT. The main issue facing these methods is the relatively large value of the net thrust used to deflect the NEO, which requires a large S/C mass in order to exert that amount of gravitational force while holding the NEO distance constraint. Thrusting time could be traded, if available, to reduce the thrust requirement and thus reduce the required spacecraft mass.

In the particular case of the hovering GT, and even with the current net thrust constraints, the mathematical model of the system would accept much smaller masses (though still larger than 10^4 kg) if both the “number of thrusters” and “minimum distance to the NEO” constraints were removed. However, in that case the solutions would call for a patently unrealistic thruster canting angle of 88.27 deg.

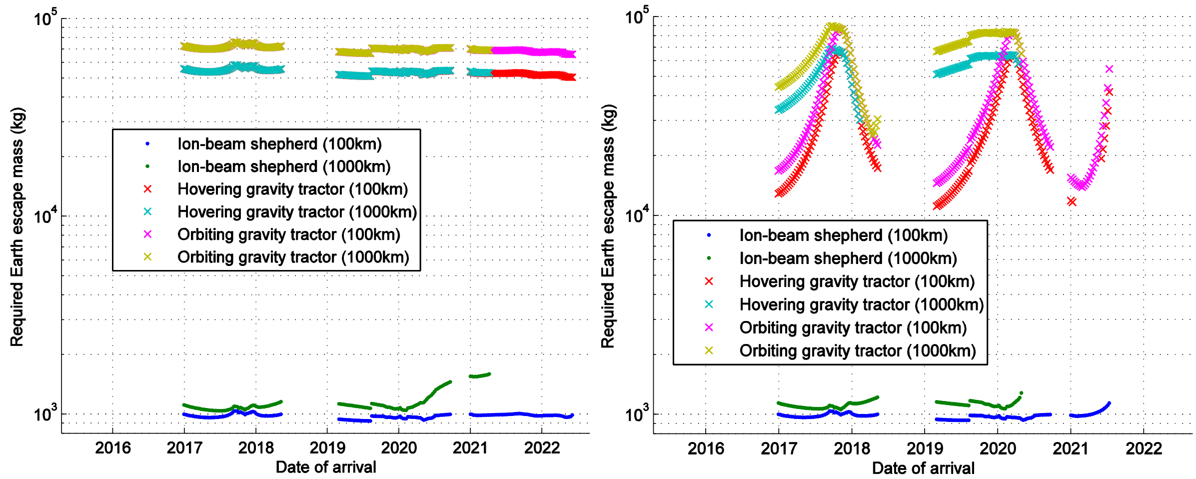

Fig. 20: Slow-action deflection requirements for the small NEO case (left: constant thrust, right: solar thrust)

Note that both such constraints are purely geometric and limit the maximum canting angle δ, so only the strictest one applies. In all instances here, the minimum NEO distance is the limiting factor, forcing a maximum canting angle of 40 deg. If this constraint was removed the canting angle could go up to 60 deg, resulting in escape masses in the order of “only” 25 tons. However, in that case the minimum S/C hovering altitude would be slightly over 15 m *above the assumed NEO radius* of 50 m. Taking into account that the *actual* shape of the NEO is unknown and could be markedly oblong, this is likely to be a very unsafe distance to hover at.

The large difference between the masses predicted for solar-powered and nuclear-powered gravity tractors may appear striking at first. However, a look at Fig. 17 shows that the push durations of the small (100 km) deflections can be in the order of days, which is very small compared to the NEO period. Since the net force to be applied to the NEO scales with $r_{Sun}^{-1.7}$, those missions that arrive far from the pericentre apply actual maximum forces that are significantly below the nominal “250 mN at 1 AU”, thus requiring a smaller mass for a mission concept that uses *gravity* to pull on the NEO. In contrast, the variations in the solar-powered deflection missions for the large (1000 km) deflections are visibly weaker, since the mission is more likely to go through a larger fraction of the NEO period and thus to cover the full range of forces.

CONCLUSIONS

This paper presents the suite of tools developed by Elecnor Deimos in the frame of the NEOShield FP7 project for the analysis of mitigation solutions and mission design options required to alleviate the risk posed by threatening NEOs. This suite is composed by three tools that respectively allow determining if an asteroid is to collide with Earth (NIRAT tool), compute the required object deflection (NEODET tool) and assess the design features of the possible mitigation space missions (RIMISET tool).

Design solutions, methods and algorithms employed and the overall set-up of the tools have been presented. Results from all the tools have been obtained by a chained execution of the different tools in application to the hypothetical NEO proposed for the conference, 2015 PDC. The obtained results allow comparing the design and performance requirements associated to different deflection methods and thus favor the selection of the best option for a space mitigation mission.

The large uncertainty present in the asteroid mass makes it very difficult for just one method to provide an ensuring mitigation response in all circumstances. Whereas kinetic impactor could cover a large envelope of cases, only the nuclear blast seems to be effective in all cases. Regarding slow-push methods, these can be used to ensure that some deflection is achieved in order to prevent the asteroid from falling on particularly high population density areas.

Finally, and in relation to the comparison between the IBS and GT missions in Fig. 20; we would like to make clear that GT deflection missions with lower total mass for each arrival date could exist. Such missions would use a lower thrust level but longer deflection times than those in Fig. 17 since, once the S/C can move no closer to the NEO, reducing the required force increases the push time *but* the additional propellant mass used is more than compensated by removing deadweight that was being used as a gravity source. An optimization process could be envisioned where a minimal mass could be found for each arrival date.

However, the applied force was selected when NEODET was run, and cannot be changed or optimized for in RIMISET alone. This is obviously an artifact of the design of the tool set that keeps the orbital requirements (NEODET) separate from the mission requirements (RIMISET). In addition, this is thought to be a fair comparison between the GT and IBS *concepts* at equal force levels; thereby highlighting the advantages of the simplicity of the ion beam shepherd design.

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