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CONCEPTS FOR NEAR-EARTH ASTEROID DEFLECTION USING SPACECRAFT WITH ADVANCED NUCLEAR AND SOLAR ELECTRIC PROPULSION SYSTEMS

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ABSTRACT

The near-Earth object population, composed mostly of asteroids rather than comets, poses an impact hazard to Earth. Space technology is reaching a sufficient level of capability and maturity where the deflection of an Earth impactor may be possible within the next decades. The paper focuses on assessing the maximum deflection capability (minimum response time) that could be achieved with a rendezvous/landed spacecraft, using electric propulsion and nuclear/solar power technologies likely to be available in the near-term, within the constraints of a single heavy launch into low Earth orbit. Preliminary design concepts are presented for large, high-power nuclear and solar electric spacecraft, based on a trade-off analysis of power/propulsion technology options and an optimisation of the complete mission design to the minimise the total response time for a representative impactor/deflection scenario. High specific impulse gridded-ion engines show significantly improved mission performance over Hall effect thrusters due to the high delta-V requirements for Earth spiral out, rendezvous, spin axis re-orientation and deflection. Amorphous silicon thin film solar arrays perform substantially better than conventional high cell efficiency alternatives. It was found that solar electric spacecraft could achieve lower total response times for the deflection than a nuclear electric spacecraft of the same initial mass, if the asteroid perihelion is much lower than the Earth. The comparison is expected to be much closer if the asteroid perihelion is near the Earth. Both systems were found to provide effective deflection capabilities for small/moderate-size impactors.

INTRODUCTION

The Near Earth Object (NEO) impact hazard can be categorized in the same manner as other catastrophic natural disaster phenomenon such as earthquakes and volcanoes. The probability of such events occurring is low and average timescales between events very long, but the consequences to the human population are extreme. Recent studies addressing the long-term hazard from NEO impacts predict, through a combination of population/risk analysis models and empirical survey data, that the hazard is commensurate with other natural hazards[1]. It is dominated by large km-sized bodies with a mean impact interval of over 600,000 years and the potential for global devastation, and also by 150-500 m objects impacting over ten times more frequently but coupled with regional tsunami effects. Unlike other natural hazards however, the advancement of technology is reaching a sufficient level whereby mankind could possess the capability to make this hazard preventable in most cases within the next few decades.

In addressing the NEO threat, ESA has recognised that space missions can play a significant role in assessing the impact risk through both observatory-based survey missions and rendezvous-based in-situ characterisation missions[2]. Within this context, a number of NEO mission studies were prepared, enabling a prioritised mission roadmap for NEO risk assessment and reduction to be planned[3,4]. Looking beyond this critical NEO risk assessment phase, it is prudent that preliminary space system concepts to actually mitigate the risk of impact should be investigated to prepare for the unlikely, but serious event that an Earth impactor is identified.

A number of different methods of asteroid deflection have been proposed[5]: an impulse imparted by an interceptor striking the object at high relative velocity or by a stand-off nuclear blast explosion; other methods involve the action of a longer duration thrust on the object including stand-off laser or solar concentrator surface ablation, and mass drivers or propulsive devices in contact with the asteroid surface.

For any deflection technique to be used, clearly its response time capability must be within the given warning time of an impact. If the warning time is only a few months to a year, then the only possible option would be a mass evacuation of the impact zone. The use of nuclear weapons would be unsuitable, since the dispersion of fragments from the disrupted body would not be sufficient and the hazard would be simply spread over a much wider area of the Earth's surface. For longer warning times of a few years, space-based intercept/impulsive methods are possible but their effectiveness would strongly depend upon the asteroid mass. With only a few revolutions before impact, the required delta-V to be imparted to the body (order 10-20 cm/s) is at least an order of magnitude higher than with warning times of a decade or more[5]. Rendezvous/propulsive methods would not be feasible in this scenario due to the time required for rendezvous and thrusting in addition to the coast time for a miss. Typical warning times for asteroid impact are expected to be on the order of 10-50 years[6] with current optical survey capabilities. Over these timescales, both intercept/impulsive methods and rendezvous/propulsive methods become feasible (assuming that the rendezvous delta-V is not too high).

There are a number of significant challenges associated with the propulsive deflection method. Most asteroids rotate about their principal moment of inertia, but some asteroids have been observed to be tumbling about all three axes, e.g. the slow, excited rotation state of NEA Toutatis[7]. In the latter scenario, it may be very difficult to stabilise and control its attitude motion so that propulsive thrusting for the deflection can occur. Additionally, if the asteroid angular momentum is too large (e.g. it is a fast rotator and/or dense), a high delta-V on-board the spacecraft will be required to re-orient the spin axis by the desired amount prior to deflection thrusting, thus reducing the deflection effectiveness. With irregular (but measurable) rotation states and gravity fields due to inhomogeneous internal mass distributions, a safe landing on the surface of an asteroid may also be difficult operationally, though not impossible[8].

The physical properties of the asteroid population are highly diverse in surface morphology and compressibility, internal structure and cohesion, macroporosity and surface regolith pools. Such diversity poses significant challenges for coupling the spacecraft to the surface in order to impart thrust. Surface attachment devices to be used will strongly depend upon the surface/sub-surface properties of the target body. Hence, a series of precursor missions to characterise these diverse properties across the NEO population would be highly beneficial.

Despite these challenges, propulsive deflection of an Earth-bound asteroid remains very attractive because of its inherent controllability, flexibility and universal application to all asteroid types. Neglecting laser and solar concentrator methods on the basis of being unfeasible on mass and maturity grounds in the near-term, none of the other methods possess these key attributes. KE interceptors and nuclear stand-off blasts will have uncertainties associated with the dynamic response of the asteroid to a large impulse, i.e. the linear and rotational momentum transfer, even if certain properties such as internal structure could be measured in advance. Terminal navigation errors with respect to centre of mass are also possible for such a high-speed impact event. If errors occur in the deflection, then another interceptor attempt may be required at greater cost. With a rendezvous spacecraft, errors can be monitored and corrected immediately and continuously. The non-impulsive nature of the deflection also ensures the structural integrity of the object, which is a concern with impulse techniques.

Electric Propulsion (EP) is an enabling technology for propulsive deflection since chemical propulsion to too mass inefficient. Due to the large mass of the bodies concerned (10-50 Mt) and the likely warning times of an impact, moderate-thrust (N-level) high-specific impulse propulsion systems are needed to meet response time requirements of 10-50 years and rendezvous/deflection delta-V requirements on the order of 20-50 km/s within the constraints of a single launch. For these systems, power requirements are on the order of 100-300 kW, which can only be supplied using either nuclear fission reactor power sources or large deployable solar array structures. The concept of using a nuclear electric propulsion spacecraft for testing asteroid deflection is also currently being studied by the B612 Foundation[9,10] using the NASA Jupiter Icy Moons Orbiter equipped with the experimental VASIMR [11] as a baseline.

In this study, our first objective was to assess the maximum capability (in terms of total response time) for the deflection of hazardous NEAs using both Solar Electric Propulsion (SEP) systems and Nuclear Electric Propulsion (NEP) systems, assuming power/propulsion technologies and launch vehicle performances available in the next decade or two. We also imposed constraints of using a single spacecraft, a single launch, and no on-orbit assembly. Our second objective was to produce preliminary design concepts for SEP and NEP deflection systems, based on an a trade-off of power/propulsion technologies and optimisation of power/propulsion key parameters in order to minimise the total response time for an example asteroid impactor orbit and mass.

EP DEFLECTION MISSION PHASES

When conducting a mission to deflect an asteroid with an electric propulsion spacecraft, a number of different mission phases are involved. Table 1 describes these phases, along with their key mission drivers and typical durations.

Phase	Description	Drivers	Typical Duration
Earth spiral out	Continuous orbit-raising from launch orbit to Earth escape	Launch orbit, s/c thrust-to-mass ratio, specific imp	Several months to a year
Rendez-vous	Long duration thrust & coast arcs in deep space to meet target at low relative speed	Asteroid orbit, s/c thrust-to-mass ratio, specific impulse	Several years
Spin axis re-orient-ation	Landing on rotation pole, thrusting to change spin axis to desired orientation	Asteroid spin state, mass and size, spacecraft thrust	Several months to a year
Push	Thrusting on the asteroid to impart a change in its velocity	Spacecraft remaining fuel, thrust, specific impulse	Several months to years
Coast	Monitoring changes in asteroid trajectory until Earth miss	Asteroid orbit &mass, s/c thrust, push time, miss dist.	Several years to decades

Table 1: EP deflection mission phases

The summation of the durations of all these mission phases is defined as the total response time (it is assumed that such a spacecraft has been developed and is ready for launch). As we can see from the table, the dominant term in the total response time is the coast time. Therefore, the largest gains in reducing the total response time should be directed to reducing the coast phase. The coast time is strongly dependent upon the asteroid orbit/mass and the previous push phase, i.e. the delta-V imparted to the asteroid by the spacecraft (the spacecraft thrust level and push time from the remaining fuel available). Thus, for a given asteroid impactor orbit/mass and required miss distance, the spacecraft design parameters (principally total delta-V budget and specific impulse, since thrust is partially dependent upon the former) can be optimised to find a minimum total response time.

Note that total response time is used here as a performance metric to evaluate different spacecraft design points, and to assess the general maximum capability of this technique using emerging technology. In reality, if an Earth-impacting asteroid were discovered, the total response time would be a hard constraint and the spacecraft would be designed to meet this constraint. Nonetheless, our analysis is useful in drawing boundary conditions beyond which more aggressive measures would be required (e.g. multiple spacecraft, multiple launches, on-orbit assembly of a single larger spacecraft etc.)

MISSION DESIGN & ANALYSIS

Due to the interdisciplinary nature involved in the mission design and the many possible trade-offs, it was necessary to construct dedicated software. The EP Deflection Toolbox was developed internally by the ESA Advanced Concepts Team in Matlab/Simulink, and includes three main modules:

- EP Spacecraft Design Model
- EP Mission Analysis Model
- Asteroid Deflection Model

The user of the software selects the launcher and injection orbit, mass budget margins and mass fractions (excluding power/propulsion), the power and propulsion technologies to use, and specifies the asteroid orbit, size, mass, rotation period, spin axis re-orientation angle, and required miss distance. The control variables are then the spacecraft total delta-V and specific impulse. The main output is the total response time, accompanied by a breakdown of time and delta-V budgets for each mission phase. Other outputs include spacecraft mass budget, power, thrust, number of thrusters, and solar or reactor radiator array area. In addition to manual variation of the control variables, the software can be run within an optimiser.

EP SPACECRAFT DESIGN MODEL

Spacecraft Sizing

A simple analytical model is used in order to derive the main spacecraft parameters of interest to the mission analysis and deflection models, namely the power generated and the thrust produced. The power generated by the power subsystem at Beginning Of Life (BOL) for electric propulsion can be expressed as:

$$P_{BOL} = \frac{M_{p+p}}{\left(\alpha_{thr}\eta_{ppu} + \alpha_{ppu}\right)N_{T,per} + \alpha_{pow}} \quad (1)$$

where α_{thr}, α_{ppu}, and α_{pow} are the specific masses (kg/kW) of the thruster, PPU (thruster Power Processing Unit) and power system respectively. η_{ppu} is the efficiency of the PPU and $N_{T,per}$ is the normalised thrust factor at the minimum perihelion distance (in this case the asteroid's perihelion). $N_{T,per}$ is used in the SEP case to size the propulsion system to accept the higher power available at the spacecraft's closest distance to the Sun, if perihelion is <1 AU (see Power System Technologies section below). For the NEP case, $N_{T,per}$ =1 since power and thrust are constant and independent of solar distance. M_{p+p} is the power and propulsion system mass available after accounting for the propellant, payload and other subsystem masses, and is given by:

$$M_{p+p} = (M_{wet} - M_f)(1 - m_{s/s} - m_{p/l}) - m_{\tan} M_f \quad (2)$$

where M_{wet} is the spacecraft wet mass, $m_{s/s}$ is the mass fraction of other subsystems (ADCS, structure, avionics etc), $m_{p/l}$ is the payload mass fraction, and m_{tan} is the tank mass fraction. Finally, M_f is the fuel mass derived from the rocket equation:

$$M_f = M_{wet}\left(1 - e^{-\Delta V / g_o I_{sp}}\right) \quad (3)$$

where ΔV is the total delta-V budget, g_o is specific gravity at Earth surface, and I_{sp} is the propulsion system specific impulse. Once the BOL power has been calculated, then the BOL thrust produced by the electric thrusters can be determined by:

$$T_{BOL} = \frac{2\eta_{thr}}{g_o I_{sp}} \eta_{ppu} P_{BOL} \quad (4)$$

where η_{thr} is the thrust efficiency, which (as the PPU efficiency) we assume to be constant for a given thruster. Hence, the thrust is entirely dependent on power and specific impulse.

Electric Propulsion Technologies

High power electric thrusters are needed in order to process the high power requirements for this type of mission (order 100-200 kW) without dominating the mass budget. At the present time, individual space-qualified thrusters are limited to a few kW. However, a number of laboratory and engineering models are being developed for much higher power densities. These include:

- Gridded Ion Engines (40-50 kW per thruster)
- Hall Effect Thrusters (40-50 kW per thruster)
- Applied-Field Magneto-Plasma-Dynamic Thrusters (100-200 kW per thruster)
- Pulsed Induction Thruster (100-200 kW per thruster)
- Variable Specific Impulse Magneto-Plasma-Dynamic Rocket (VASIMR) (100 kW to MW)

Due to their relative technological maturity and heritage at lower power, only GIEs and HETs are considered in this study. It has not been possible to include MPDs, the PIT or VASIMR because of a lack of reliable information on either thrust/Isp performance at high powers, or a lack of definition work on their PPU requirements and designs for space use. The latter information is crucial because the PPU specific mass forms a significant part of the overall EP system specific mass. A summary of the expected performances of high power GIE and HET systems (including their PPUs) in space over the next decade or so is given in Table 2, based on recent development and ground test data[12-14]. These values are used in the model described above.

Parameter	**Gridded Ion**	**Hall Effect**
Thrust efficiency, %	79	62
Specific impulse, s	3000-10000	1500-3000
Max. power input per thruster, kW	40	50
PPU efficiency, %	95	95
Thruster specific mass, kg/kW	0.8	1.3
PPU specific mass, kg/kW	6.2	4.75
Overall EP system specific mass, kg/kW	7	6.05

Table 2: Expected performances of high power EP systems used in the study

Note that the PPU specific mass values (derived from an ESA study[15]) are slightly conservative compared to the <4 kg/kW specifications for the NASA Project Prometheus[16] due to differences in design and materials. Engine lifetime is not considered to be an issue here, since many developments in erosion-resistant materials are currently being performed to extend lifetime well beyond mission requirements.

Power System Technologies

For electric propulsion spacecraft, the power system specific mass is an important parameter which, as we can see from equation (1), has a significant influence on the power and thrust available and therefore the mission performance. For solar power systems, developments in small multi-junction and large amorphous cells are improving photovoltaic conversion efficiencies, and research into advanced lightweight array substrates/structures is significantly reducing the areal density of solar arrays. Both of these factors are combining to reduce solar array specific mass considerably. Table 3 summarises these performances for a range of options, based on recent experimental laboratory developments.

Array	**Eff-icie-ncy**	**Areal density kg/m^2**	**Specific area m^2/kW**	**Specific mass kg/kW**
3J GaAs, honeycomb	33%	8	3.4	27.4
3J GaAs, composite	33%	4	3.4	13.7
Amorph Si thin film, CFRP booms	12%	0.5	8.5	4.2

Table 3: Expected performances of solar arrays

Unfortunately, there also needs to be a secondary power system in addition to the solar array in order to manage operational risks, such as the need to perform emergency manoeuvres during eclipse or in case of temporary failures (e.g. solar array drive motor or attitude control) or prior to solar array deployment. In this case, high energy density batteries such as Li-ion at 130 Wh/kg are required. Assuming a requirement to supply full power for thrusting for 1 hour, then the battery subsystem specific mass is 8.5 kg/kW. Thus, overall power system specific mass will range from 12.7 to 35.6 kg/kW. Another major factor to consider with the mission design of SEP spacecraft is the variation of power (and thrust) with solar distance due to solar intensity and cell working temperature effects, as shown in Figure 1 for amorphous Si cells.

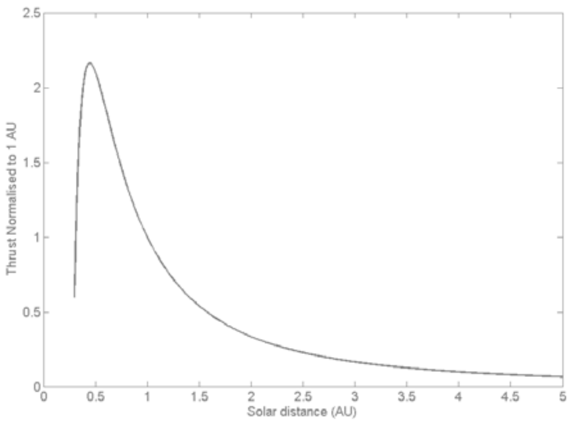

Figure 1: Variation of thrust with solar distance, normalised to 1 AU (amorphous Si array)

Cell efficiency exponentially degrades with increasing temperature closer to the Sun, whilst solar intensity increases as a square law closer to the Sun. The net effect is a peak enhancement in thrust of a factor of 2 at 0.6 AU and an order of magnitude reduction in thrust at 3.6 AU. This effect is likely to have an influence on the deflection time for hazardous asteroids with moderate to high eccentricity and must be accounted for in the model.

Beyond the technology developed and flown since the 1960's with the low-power US Radioisotope Thermoelectric Generators (RTGs) and moderate-power Russian TOPAZ reactors, there has been little advancement in nuclear power systems to much higher power levels. Many high-power space fission reactor concept studies have been performed worldwide however, which enables us to at least get an indication of typical masses at different power levels. Figure 2 shows these values for the different concepts.

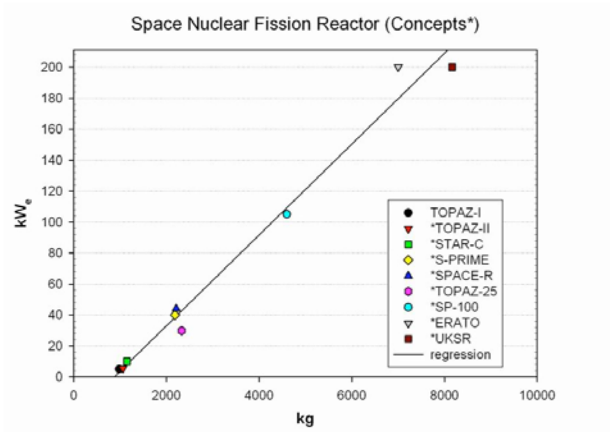

Figure 2: Nuclear reactor mass for different electrical power levels

Despite radically different reactor core, energy conversion and heat rejection techniques used, the distribution of total reactor system mass versus electrical power generated shows a remarkably close linear regression fit (solid line in the figure). Hence, reactor power versus mass can be expressed by an analytic approximation:

$$P(kW_e) = 0.0293\, M(kg) - 25.63 \qquad (5)$$

Using this relationship, the reactor specific mass levels off to 42 kg/kW for power levels above 100 kW, which is significantly higher than for some of the more advanced solar power concepts. However, the NEP spacecraft benefits from constant power and thrust throughout the mission (especially at larger solar distances) and reduced operational constraints.

Technology Trade-off & Selection

Since thrust and delta-V (hence push time) are drivers for the latter phases of the mission where the spacecraft is attached to the asteroid surface, it is desirable to find optimum specific impulse values and select technology options in order to maximise the thrust over a range of possible total delta-V capability for a given launch mass. Power and thrust scale with spacecraft mass, and therefore the highest currently possible launch mass into Low Earth Orbit is used in order to assess a maximum deflection capability. This is a reasonable launch constraint because the Earth spiral out time from LEO to Earth escape will still be small compared to the total response time. Taking the Proton K as the heavy launch vehicle and applying a 3% launch margin then a 20% system margin, we obtain a spacecraft wet mass of 16,220 kg. The following assumptions on mass fractions were made as an input to the EP Spacecraft Design Model:

- Payload mass fraction: 10% of dry mass
- Structure mass fraction: 25% of dry mass
- ADCS mass fraction: 3% of dry mass
- Avionics (incl thermal): 90 kg
- Xenon propellant tanks: 15% of propellant mass

Figure 3 to Figure 5 present the BOL thrust for the different power system and propulsion system options. In Figure 3, we can observe that the optimum specific impulse is constant over the range of delta-V at 3000 s. This is the upper bound imposed on HET technology, implying that the true optimum is above this ceiling. The resultant effect of limiting this value is an increase in propellant mass with increasing delta-V, which in turn reduces available mass for the power system, and causes the thrust to decrease rapidly towards zero irrespective of the solar power technology used.

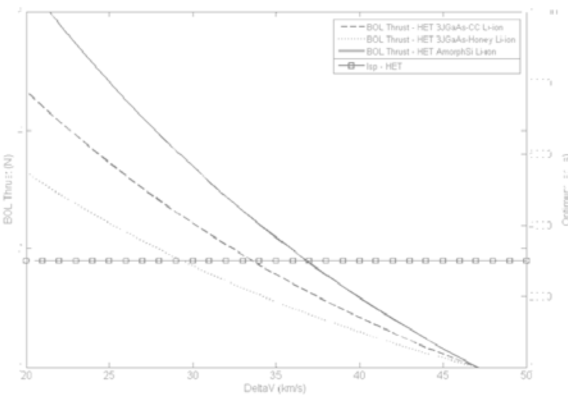

Figure 3: BOL thrust and optimum specific impulse versus delta-V for different array technologies (SEP spacecraft with HETs)

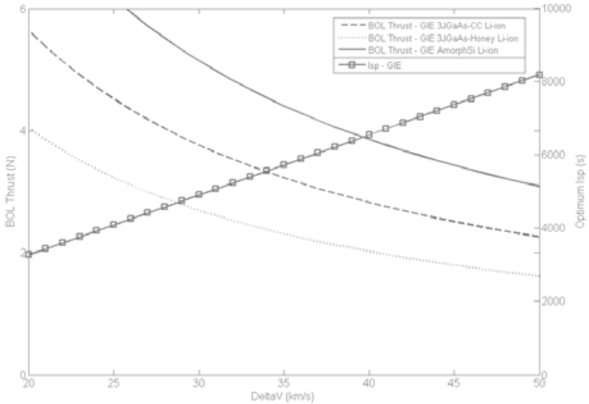

Figure 4: BOL thrust and optimum specific impulse versus delta-V for different array technologies (SEP spacecraft with GIEs)

In the case of GIEs in Figure 4, thrust drops off much more gradually with increasing delta-V as optimum specific impulse is allowed to increase proportionately from 3300 to 8100 s. This maintains a constant propellant mass, and hence power at 167 kW. In general, we can observe that the use of HETs leads to significantly lower thrust levels than GIEs, especially at high delta-V values. Thus, GIEs were selected as the propulsion technology for both NEP and SEP. From Figure 4, there is a significant difference in thrust level according to the power system technology used due to the wide variations in specific mass. The use of amorphous silicon thin film arrays enables a substantially higher BOL thrust, almost a factor of 2 higher than conventional honeycomb solar arrays. It was therefore selected for the SEP power system.

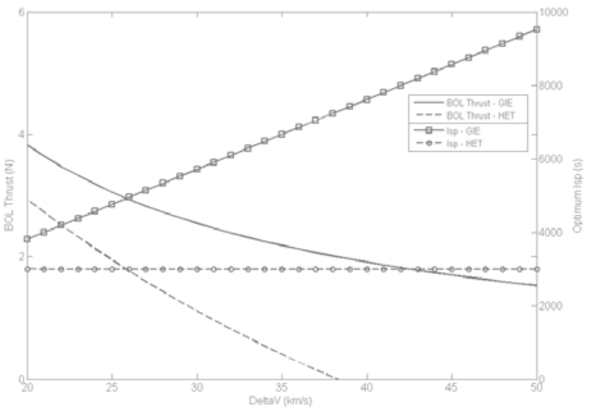

Figure 5: BOL thrust and optimum specific impulse versus delta-V for GIE and HET propulsion on the NEP spacecraft

In Figure 5, it can be seen that the NEP spacecraft has similar thrust to the SEP spacecraft with conventional GaAs honeycomb solar panels due to the fact that their specific masses are very similar. The constant power level across the delta-V range in this case is 95 kW.

EP MISSION ANALYSIS MODEL

Earth Spiral Out

The delta-V and transfer time for this phase of the mission can be modelled analytically to a reasonable accuracy by assuming a constant application of thrust in-plane and tangential to the Earth orbit, i.e. aligned with the velocity vector. Under this assumption, the propellant mass for a low-thrust orbit-raising between two coplanar orbits is given by:

$$M_f = M_1\left(1-\exp\left(\frac{\sqrt{\mu_E}}{g_o I_{sp}}\left(\frac{1}{\sqrt{a_2}}-\frac{1}{\sqrt{a_1}}\right)\right)\right) \quad (6)$$

where μ_E is Earth's gravitational parameter, M_1 is the spacecraft mass at the initial lower orbit and a_1 and a_2 are the respective semi-major axes. Then, from the propellant mass and assumption of constant thrust, we can obtain an estimate of the spiral out time:

$$t_{spiral} = \frac{M_f g_o I_{sp}}{T_{BOL}} \quad (7)$$

Here, we neglect the influence of eclipse duration on the spiral out time for the SEP spacecraft because we have included a battery subsystem in the design to continue thrusting during these ~1 hour events. A transfer to a sub-lunar orbit is assumed, followed by a lunar swingby period lasting 2 months until Earth sphere of influence is crossed and the interplanetary rendezvous phase begins.

Rendezvous

Due to the complexity of modelling interplanetary low-thrust trajectories, we chose to use an external tool. The DITAN (Direct Interplanetary Trajectory ANalysis) software[17], developed for ESA by Politecnico di Milano, was employed to generate and optimise rendezvous trajectory designs for a target asteroid (see below) and a number of reference SEP and NEP spacecraft designs. DITAN used the objective function of minimising propellant mass and transfer time. A fixed maximum thrust was assumed for the NEP reference missions, and a varying maximum thrust with solar distance was used for the SEP reference missions according to the function presented in Figure 1. This enabled the influence of specific impulse and thrust-to-mass ratio on rendezvous delta-V and transfer time to be analysed and curve-fitted for general use by the EP Mission Analysis Model, as a reasonable approximation.

DEFLECTION MODEL

Spin Axis Re-orientation

Unless extremely fortunate, the asteroid will not be rotating in the correct orientation for deflection thrusting to occur and hence some re-orientation will be required. De-spinning the asteroid and obtaining three-axis control of its attitude motion would be desirable, but is unfeasible with the asteroid's very high moment of inertia and the N-level thrust produced by a large EP spacecraft[18]. Therefore, a spin axis control strategy must be used to perform the re-orientation, before deflection thrusting can proceed on the rotating asteroid. Assuming that the asteroid is rotating only about its principal moment of inertia, this strategy can be implemented by landing and attaching the spacecraft to one of the rotation poles. A continuous inertially-fixed thrust is then applied perpendicular to the spin axis until it rotates through the desired angle of re-orientation. In this scenario, the re-orientation time is determined by[18]:

$$t_{spin} = \frac{\theta_{spin} I_{ast} \omega_{ast}}{r_{ast} T_{sc}} \quad (8)$$

where θ_{spin} is the re-orientation angle swept, I_{ast} is the asteroid principal moment of inertia, ω_{ast} is the angular velocity of the asteroid about its spin axis, r_{ast} is the radial distance of the pole surface from centre of mass along the principal axis, and T_{sc} is the spacecraft thrust (variable with solar distance in the SEP case).

Deflection Thrusting

It can be shown that the Earth miss distance induced by applying an external force to an Earth-impacting object in heliocentric orbit is dependent upon the object semi-major axis, a, the Sun's gravitational parameter, μ, the Earth radius, R_E, the start time of the deflection manoeuvre prior to impact, t_s, and the integration of the dot product between the thrust acceleration vector, $\vec{A}$, and asteroid velocity vector, $\vec{v}$, over the duration of the deflection manoeuvre, t_p (the push time):

$$s = \frac{3a}{\sqrt{\mu R_E}}\int_0^{tp}(t_s - t)\vec{A}\cdot\vec{v}\,dt \quad (9)$$

This general asteroid deflection formula, derived and reported in detail by Izzo in a later publication[19], is applicable to both long-duration low-thrust and high-energy impulsive deflection methods[20]. We have used the formula extensively in this study to determine the minimum start time to guarantee a desired miss distance, given an available push time, and adopting a particular deflection strategy to define the acceleration vector in the perifocal reference plane. Available push time is determined by the propellant mass remaining after the spin axis reorientation phase and the propellant mass flow rate in a similar manner to equation 7.

In the NEP case, thrust *T* is fixed and the push time can be easily calculated. Thrust is varying in the SEP case according to solar distance, so the push time must be derived as a result of integration. There are a number of possible strategies for the deflection. These include:

- Inertially-fixed thrust vector (spin axis in-plane)
- Naturally-precessing thrust vector (spin axis rotating in-plane with mean motion)
- Simultaneous ‘push and torque’ (spin axis out-of-plane[18] and forced with mean motion)

A trade-off between these strategies will be carried out in later work[20], but for the purposes of this study, the inertially-fixed strategy was selected as a first step due to its simplicity. Here, we apply the thrust acceleration in a fixed direction which is aligned with the asteroid velocity vector at perihelion in order to maximise the dot product in equation 9. Hence, there will be local maxima in the miss distance when thrusting occurs around perihelion.

ASTEROID DEFLECTION SCENARIO

When assessing the deflection capability of EP spacecraft, it is necessary to set exemplar capability requirements that are relevant to the asteroid impact hazard and population characteristics. After the near-term completion of the Spaceguard survey covering km-sized NEOs, the peak of the remaining residual hazard is predicted to be centred upon 200-300 m-sided objects due largely to tsunami effects[1]. Additionally, the orbital distribution of the Potentially Hazardous Object (PHO) population is such that impulsive rendezvous delta-V requirements vary between 5 and 30 km/s, with the peak of the distribution being around 10 km/s. Hence, the deflection capability is assessed on the basis of being able to deflect Earth-bound asteroids of up to 200 m in size and rendezvous delta-V up to 10 km/s. Larger and/or less accessible objects will thus have longer associated response times than those predicted here.

Based on these capability requirements, it was possible to derive an example asteroid impactor for the deflection model. We searched the PHO population for a ‘representative’ target orbit, with an impulsive rendezvous delta-V close to 10 km/s and orbital elements close to the peaks observed in the population distribution[7]. The orbit of object 2003 GG21 was chosen and modified slightly in argument of perihelion in order to ensure a miss distance of zero with respect to the Earth’s orbit. These modified orbital elements were then assigned to our example asteroid impactor, as seen in Figure 6. Table 4 presents its orbital, physical and rotational properties.

Parameter	Value
Orbital properties	
Semi-major axis, a (AU)	2.143
Eccentricity	0.709
Perihelion, q (AU)	0.623
Aphelion, Q (AU)	3.66
Inclination, i (°)	10.12
Arg. perihelion, ω (°)	95
Ascending node, Ω (°)	13.2
Period (yrs)	3.14
Synodic period (yrs)	1.45
Pre-deflection miss distance (km)	0
Post-deflection miss distance (km)	10,000
Physical properties	
Diameter (m)	200
Density (g/cm^3)	2.4
Mass (Mt)	10
Rotational properties	
Rotation period (hr)	9
Rotation pole wrt plane (°)	40

Table 4: Example Earth-impacting asteroid properties

The orbit has low perihelion and high aphelion in order to introduce thrust variation with solar distance for the SEP spacecraft. Since the distribution of rotation axes amongst the population is random with respect to the orbit plane, we have set a spin axis re-orientation requirement close to the mid-point between minimum (0°) and maximum (90°).

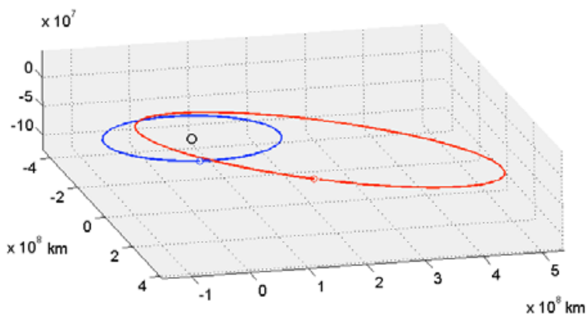

Figure 6: Orbital geometry of the example Earth-impacting asteroid scenario

SIMULATION RESULTS

Using the model described above, the total response times for the successful deflection of the 10 Mt asteroid impactor example can be predicted for a range of EP spacecraft total delta-V capabilities. The thrust-optimised specific impulse values across the delta-V range from 25-50 km/s are taken from the selected curves presented in Figures 4 and 5. Since the push time for the deflection depends upon the remaining propellant left over after all previous mission phases have been completed, this has a direct influence on the total deflection time required (push + coast time to the miss). As we can see from Figure 7 for the SEP spacecraft, the increase of specific impulse over the delta-V range causes the total propellant mass to remain constant, and the mass for all phases prior to the deflection phase to decrease exponentially. The resultant effect is that the propellant available for the deflection increases. A total delta-V budget of 25 km/s would leave almost negligible propellant for the deflection and a short push time (hence delta-V imparted to the asteroid).

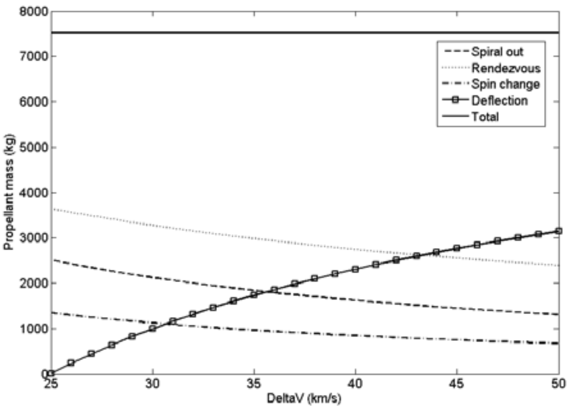

Figure 7: Variation of propellant mass with spacecraft delta-V capability for the SEP deflection of a 10 Mt asteroid impactor by 10,000 km

With a short push time in this case, Figure 8 shows that the deflection phase must start over 90 years in advance of the asteroid impact time in order to induce a miss by the required 10,000 km. As the amount of deflection propellant increases with spacecraft total delta-V, we can observe that the deflection time and total response time drops dramatically in discrete steps to a near constant level after 32 km km/s. These steps are multiples of the orbital period of the asteroid. This coincides with conditions when the spacecraft has sufficient push capability (thrust and push time) to exceed a threshold that enables the inertially-fixed deflection strategy to start on a later perihelion pass (i.e. one or more perihelion passes closer to the impact time). For the SEP spacecraft, minimum deflection times of 6.4 years occur over a wide range of total delta-V from 32 km/s to 48 km/s. At higher delta-V values, the deflection time starts to increase again in the same discrete steps. Again, this is due to the push capability (thrust-push time product) this time falling below the threshold, and so deflection must start on an earlier perihelion pass.

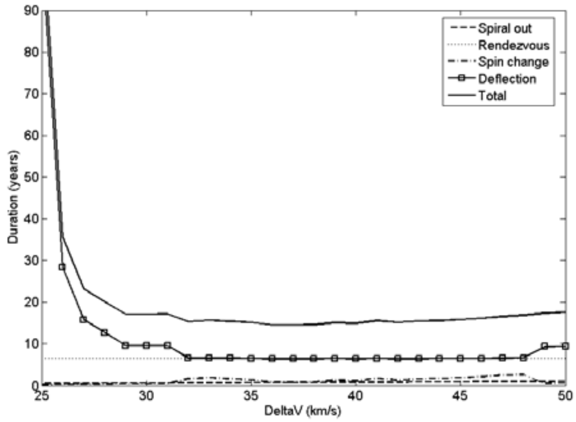

Figure 8: Variation of response time with spacecraft delta-V capability for the SEP deflection of a 10 Mt asteroid impactor by 10,000 km

We can also see from Figure 8 that the spin axis re-orientation time increases to nearly 2 years over this optimum delta-V range, compared to 0.5 years elsewhere. For the SEP case, where thrust is a strong function of solar distance, the duration of this phase is heavily dependent upon the mean anomaly of the start time for the proceeding deflection phase with respect to perihelion (where the maximum thrust occurs). In order to achieve the minimum deflection time on a later perihelion pass, deflection thrusting starts up to 30° in mean anomaly prior to perihelion. As this angle increases, the spin axis re-orientation phase occurs over higher solar distances and hence significantly lower average thrust levels. This leads to longer times according to equation 8. Accounting for these influences, the 14.6-year minimum response time occurs from 36 to 38 km/s. The latter value has been taken as the design baseline for the SEP spacecraft. A breakdown of all phases is presented in Table 5.

Phase	**Delta-V (km/s)**	**Duration (yr)**
Spiral out	6.77	0.81
Rendezvous	13.25	6.46
Spin axis change	4.82	0.89
Deflection push	13.16	1.17
Deflection coast	-	5.27
Total	**38**	**14.6**

Table 5: Delta-V and time budgets for the optimised SEP spacecraft deflection of a 10 Mt asteroid impactor by 10,000 km

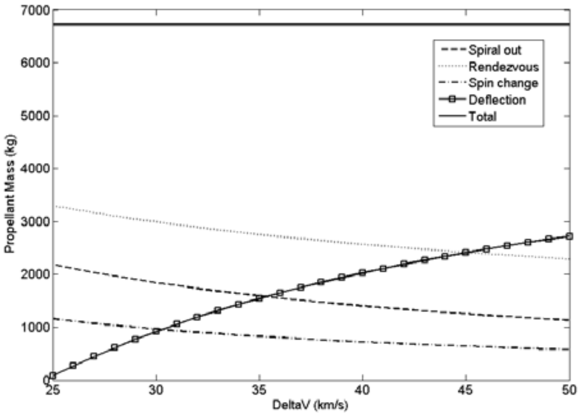

Figure 9: Variation of propellant mass with spacecraft delta-V capability for the NEP deflection of a 10 Mt asteroid impactor by 10,000 km

The NEP spacecraft has a similar minimum total delta-V budget to the SEP spacecraft of 25 km/s. Propellant masses of all mission phases are also similar, but slightly lower than the SEP case, as given by Figure 9. This is due to the higher power system specific mass of the nuclear reactor.

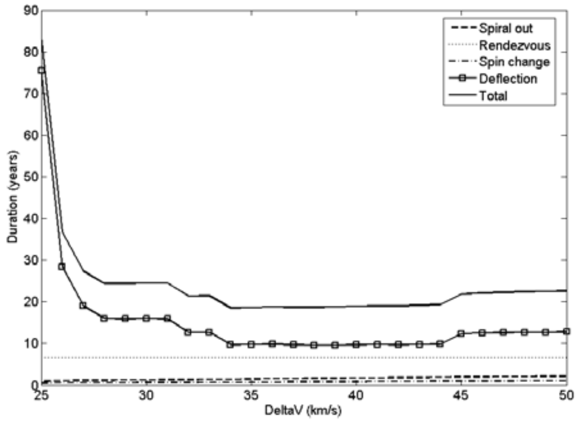

Figure 10: Variation of response time with spacecraft delta-V capability for the NEP deflection of a 10 Mt asteroid impactor by 10,000 km

The NEP spacecraft also has similar total response time trends to SEP. These can be observed in Figure 10. Due to a lower thrust around the perihelion where the optimum deflection occurs, the deflection starts one perihelion pass earlier than the SEP case, giving a minimum NEP deflection time of 9.6 years between 34 and 44 km/s spacecraft delta-V. The time for spin change re-orientation does not fluctuate significantly over the delta-V range because thrust is constant over the entire orbital arc of the asteroid. Only a slight linear increase is evident due to the exponentially decreasing thrust with increasing delta-V displayed in Figure 5. The result of these factors produces a minimum total response time of 18.7 years for the NEP spacecraft with delta-V capability between 34 and 44 km/s. We have selected a delta-V of 38 km/s as the design baseline since it is near the mid-point of this range. The budgets for each mission phase are given in Table 6.

Phase	Delta-V (km/s)	Duration (yr)
Spiral out	6.76	1.65
Rendezvous	14	6.57
Spin axis change	4.62	0.85
Deflection push	12.62	2.07
Deflection coast	-	7.56
Total	**38**	**18.7**

Table 6: Delta-V and time budgets for the optimised NEP spacecraft deflection of a 10 Mt asteroid impactor by 10,000 km

Overall, the minimum achievable response times to deflect a 10 Mt (approx. 200 m size) asteroid by 10,000 km are quite low for both SEP and NEP spacecraft in relation to typical warning times of 10 to 50 years. In this particular asteroid impactor scenario, the use of SEP achieves a shorter response time than NEP by over 4 years. However, this amount is highly dependent upon the perihelion distance of the asteroid. In this case it is quite low at 0.62 AU, leading to higher thrust around perihelion than NEP and hence shorter deflection times by one orbit period. If the asteroid's perihelion was nearer to 1AU, then we might expect the comparison to be much closer.

SPACECRAFT CONFIGURATION

There are a number of driving factors associated with EP deflection missions that significantly influence the configuration of both SEP and NEP spacecraft. Firstly, the spacecraft is attached to the rotation pole of the asteroid, but thrusting to reorient the pole needs to be performed perpendicular to it in inertial space. This implies that the whole spacecraft above the surface attachment device must be despun, thus requiring a drive motor to match the rotation rate of the asteroid. It also implies that the ion thruster assembly must have two degrees of freedom in order to apply thrust along the spacecraft body axis for rendezvous and deflection pushing, and to turn 90° for the spin axis reorientation. The latter could be in any inertial direction according to the initial and desired final states of the spin axis. Hence, the thruster assembly requires 360° movement in azimuth (rotation about the spacecraft z-axis) and 90° in elevation (similar to a telescope mount).

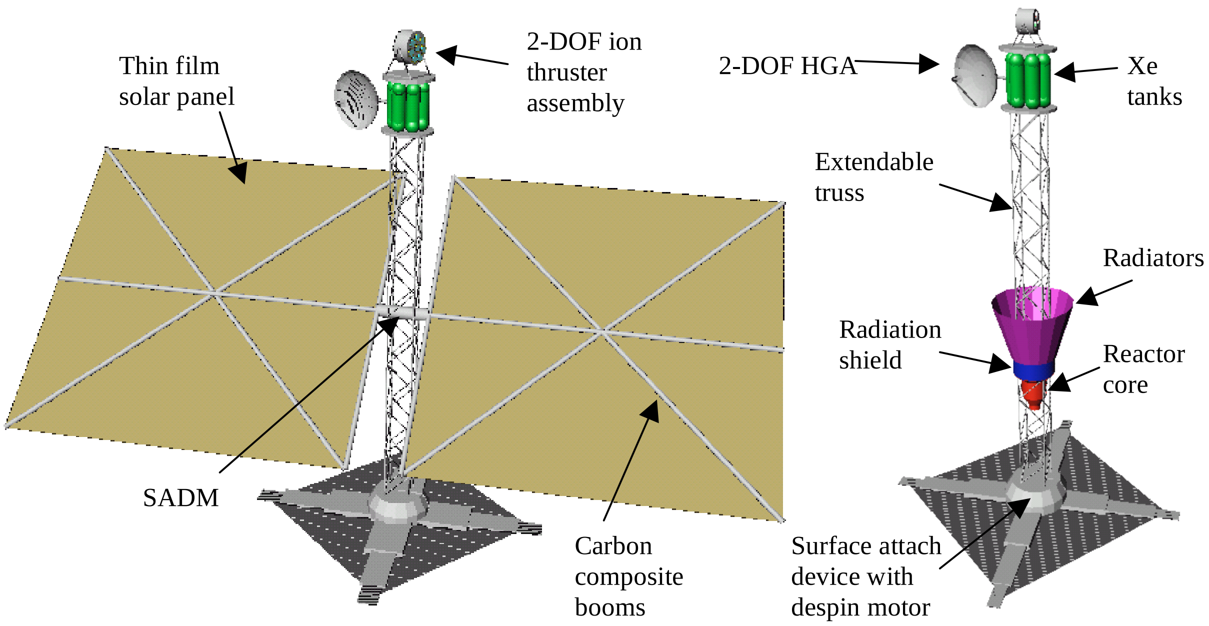

Figure 11: SEP and NEP spacecraft configurations (fully deployed for surface landing and attachment)

Characteristic	Summary	Comments
Launch	Proton K	LEO
Mass	20,270kg wet mass 16,220kg w/o margin	20% system margin
Dimension	35m long	deployed length
Power (SEP option)	165 kW (BOL) 295 kW (perihelion) 19 kW (aphelion)	Deployable Amorph Si Thin Film array with CFRP booms; 2 wings, 26m by 26m each; 1400 m^2 total area
Propulsion (SEP option)	4 N thrust (BOL) 7.3 N thrust (perihelion) 0.5 N thrust (aphelion) 6215 s specific impulse	8 x 40kW Gridded Ion Thruster assembly; 6 cylindrical Xenon tanks storing 1000L at 100bar
Power (NEP option)	95 kW (electric) 950 kW (thermal)	Thermoelectric conversion system; 1-sided carbon radiators with Ti heat pipes, K fluid, 65 m^2 area
Propulsion (NEP option)	2 N thrust (constant) 7235 s specific impulse	3 x 40kW Gridded Ion Thruster assembly; 6 cylindrical Xenon tanks storing 900L at 100bar
Comms	>100kbps @ 2AU range (Ka)	X/Ka-band for radio science, high rate image transmit; 2.5m HGA
Attitude control	3-axis stabilised Absolute 0.01° Relative 10arcsec in 10s	Reaction wheels for fine pointing; 2-DOF gimballed main thrusters used for large and rapid slew manoeuvres
Structure	Lightweight central truss	Telescopic extension of sections to full length
Payload	Surface attachment device High resolution imager Radar tomographer NIR, TIR Spectrometers	Deployable legs and flexible webbing on attach device
Mechanisms	Despin motor Solar array drive motor (SEP option) Gimballed HGA motor Gimballed thruster assembly Attach device leg joints	Despin on attach device/truss interface

Table 7: Baseline spacecraft design specifications

Despite the complexity of this mechanism, it does have the advantage of enabling the thrusters to be used for large and high rate attitude slew manoeuvres in free-flight mode. The large solar array wings in the SEP spacecraft design also need complete 360° rotation about the spacecraft x-axis in order to maintain normal incidence to the Sun (pre- and post-landing) without experiencing plume impingement from the ion thrusters. They must also be above the surface attachment device in order to avoid grounding on the asteroid surface after landing. The implication is that the solar array must be entirely between the ion thruster assembly and the attachment device, and hence a long truss structure is needed which is longer than the height of the solar array wings (of the order of 20 m for amorphous silicon thin film arrays).

Despite the radiation shield placed close to the nuclear reactor, the NEP spacecraft also requires a long truss structure to separate the reactor at a sufficient distance from the on-board electronics and instrument payload housed in the propulsion tank module (such as an optical imaging camera). This is in order to achieve an acceptably low radiation dose to prevent permanent damage and temporary upsets. The required separation distance is also on the order of 20 m. The long truss structure needs to be in a number of sections nested within one another and deployed telescopically after launch in order to be stowed within the launcher fairing.

The surface attachment device must be able to fit to the contours of the asteroid surface, which can be highly variable, otherwise the spacecraft may become unstable. This would suggest that multi-jointed legs should be used. The device should also extend laterally enough to counteract the torque during surface-parallel thrusting for spin axis reorientation and prevent tip-over of the spacecraft. In order to avoid the legs sinking into regolith pools and surface rubble during the application of thrust into the surface for the deflection push, it is suggested to use a lightweight, tough, flexible webbing material. This would reduce the exerted pressure on the surface. However, this is only a very preliminary design and this particular area needs to be studied in much greater depth in order to find a set of solutions that can be used on a range of different asteroid surfaces.

After taking account of these driving factors and the results of the system design optimisation presented in the previous section, the preliminary configurations for both the SEP and NEP spacecraft are presented in Figure 11. A summary of the preliminary design specifications is provided in Table 7.

CONCLUSIONS

The study has assessed the maximum deflection capability achievable with spacecraft using high-power, high specific impulse electric propulsion and advanced power system technologies likely to be available within the next decade or so. Using the maximum current launcher lift capacity into low Earth orbit, a 15-20 ton-class spacecraft can perform rendezvous, spin axis re-orientation and deflection of a 10 megaton (approx. 200 m) Earth-impacting asteroid within a minimum response time of 10-20 years. Hence, it can be concluded that electric propulsion deflection is very effective for this class of asteroid, considering that typical warning times are of the order of 10-50 years. Larger asteroids of 300 m have a mass over three times larger and therefore it can be expected that response times would be in the 30-60 years range, which is still reasonable. It should be noted that these sizes of asteroids would represent the peak of the impact hazard once existing surveys have retired the risk from km-sized bodies with potential to cause global devastation.

Exact response times are dependent upon the orbit and rotation of the asteroid, but these are considered to be representative predictions taking account of possible variations from the example chosen. Optimum deflection occurs close to the asteroid perihelion when thrust and velocity vectors are aligned. For asteroids with perihelion distance close to the Sun, solar electric propulsion would produce significantly better deflection performance than nuclear electric propulsion spacecraft of the same launch mass (assuming that the propulsion system is sized for the higher solar powers experienced at perihelion). For asteroids with perihelion close to the Earth, the comparison between solar and nuclear electric propulsion for deflection is expected to be much closer.

A number of spacecraft technologies would need to be developed and space-qualified for this capability to become feasible. They include large gridded ion thrusters and associated power processing units able to process up to 40-50 kW power and operate at specific impulses of 6000-7500s, large deployable amorphous silicon thin film solar arrays with high packing density, 100 kW nuclear reactor system, lightweight strong extendible truss structures, large two degree of freedom gimbal mechanisms for thrust vectoring, and deployable articulated surface attachment devices with hold-down mechanisms. Fortunately, the power and propulsion technologies have good synergy with those being developed for ambitious future planetary science missions using electric propulsion.

OUTLOOK

This preliminary analysis on the deflection of Earth-bound asteroids with EP spacecraft has provided some insight into the maximum capability that could be reached in the next decade or so with a dedicated development programme (or benefiting from similar NEP/SEP spacecraft developments for science missions). Further work is needed to investigate the sensitivity of response time to different asteroid orbit and mass classes to gain a broader picture. It would also be useful to look further ahead in terms of what could be achieved using more advance propulsion systems, once their specific mass performances become clearer. Apart from performing trade-off analyses between different deflection thrusting strategies, and between EP and KE interceptors, further research needs to be conducted into the definition of robust, stable and secure attachment devices for different asteroid surface environments.

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