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Astrodynamic Fundamentals for Deflecting Hazardous Near-Earth Objects*

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Abstract

The John V. Breakwell Memorial Lecture for the Astrodynamics Symposium of the 60th International Astronautical Congress (IAC) presents a tutorial overview of the astrodynamical problem of deflecting a near-Earth object (NEO) that is on a collision course toward Earth. This lecture focuses on the astrodynamic fundamentals of such a technically challenging, complex engineering problem. Although various deflection technologies have been proposed during the past two decades, there is no consensus on how to reliably deflect them in a timely manner. Consequently, now is the time to develop practically viable technologies that can be used to mitigate threats from NEOs while also advancing space exploration.

1 Introduction

Despite the lack of a known immediate threat from a near-Earth object (NEO) impact, historical scientific evidence suggests that the potential for a major catastrophe created by an NEO impacting Earth is very real. It is a matter of when, and humankind must be prepared for it.

74 million years ago, a 2-km-wide asteroid struck in central Iowa, creating the Manson Crater. Now covered with soil, it is the largest crater in North America at more than 38 km across. 65 million years ago, a larger 10-km-wide asteroid struck near the Yucatan Peninsula in Mexico and created the 170-km-across Chicxulub Crater. Most scientists now believe that a global climate change caused by this asteroid impact may have caused the dinosaur extinction. On June 30, 1908, an asteroid or comet estimated at 30 to 50 m in diameter exploded in the skies above Tunguska, Siberia. Known as the Tunguska Event, the explosion flattened trees and killed other vegetation over a 500,000-acre area with an energy level equivalent to about 500 Hiroshima nuclear bombs.

In the early 1990s, scientists around the world initiated studies to prevent NEOs from striking Earth [1]. However, it is now 2009, and there is no consensus on how to reliably deflect them in a timely manner even though various mitigation approaches have been investigated during the past two decades [1-8]. Consequently, now is the time for initiating a concerted R&D effort for developing practically viable deflection technologies before any NEOs are discovered heading toward Earth.

A 2-km NEO is known to be capable of causing catastrophic alteration of the global ecosystem which may lead to the end of civilization. Ocean impacts of even smaller objects are of some concern because the destructive potential caused by the resulting tsunamis may be above that from a same-size object's land impact. The probability of a major impact to cause the extinction of humanity is extremely low, but it is not zero. Unlike many other natural disasters, such as earthquakes, tsunamis, hurricanes, and tornadoes, which cannot be prevented, the impact threat posed by NEOs can be mitigated given adequate warning time. The impact of an object smaller than 30 m in diameter is often naturally mitigated by the Earth's atmosphere. As typical small meteoroids enter the atmosphere, they often burn up or explode before they hit the ground. If they burn up, they are called meteors; if they explode, they are called bolides.

A near-Earth asteroid (NEA) refers to any asteroid with a perihelion of less than 1.3 AU. If comets are included, then we speak of near-Earth objects (NEOs). If an NEA's perihelion is less than that of Earth, and its

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aphelion is greater than that of Earth, it is referred to as an Earth-crossing asteroid (ECA). All asteroids with an Earth Minimum Orbit Intersection Distance (MOID) of 0.05 AU or less and an absolute magnitude of 22.0 or less are considered Potentially Hazardous Asteroids (PHAs). Asteroids that cannot get any closer to the Earth than 0.05 AU ($\approx 117R_\oplus$) or are smaller than about 150 m in diameter are not considered PHAs. A comet sometimes experiences net thrust caused by evaporating ices; this thrust varies significantly as a function of radial distance from the sun, the comet's rotational axis and period, and the distribution of ices within the comet's structure. The precise trajectories of comets are thus less predictable, and an accurate intercept is correspondingly more complex.

Both high-energy nuclear explosions and low-energy non-nuclear alternatives will be discussed in this paper. The non-nuclear alternatives include kinetic impactors, slow-pull gravity tractors, and use of the Yarkovsky effect.

2 Asteroid 99942 Apophis

In this section, the basic orbital characteristics of asteroid 99942 Apophis are briefly described. Throughout this paper, we will use Apophis as an "example" target asteroid to illustrate the astrodynamic principles for deflecting NEOs.

Asteroid 99942 Apophis, previously known by its provisional designation 2004 MN4, was discovered on June 19, 2004. It is currently predicted to swing by at around 32,000 km from the Earth's surface in 2029 with a probability of 1 in 45,000 for a keyhole passage in 2029 to result in a resonant return to impact the Earth in 2036 [9, 10]. Keyholes are very small regions of the first encounter b-plane such that if an asteroid passes through them, it will have a resonant return impact with the Earth [11, 12]. Further accurate observations of its orbit are expected when it makes fairly close flybys at 0.1 AU from Earth in 2013 and 2021.

Apophis is an Aten-class asteroid with an orbital semimajor axis less than 1 AU. Its mass is estimated to be 4.6×10^{10} kg and its size is currently estimated to be about 270 m in diameter. It has an orbital period of 323 days about the sun. After its close flyby of the Earth in 2029, it will become an Apollo-class asteroid.

An extremely small amount of impact velocity change, $\Delta V \approx 0.05$ mm/s, in 2026 is required to move Apophis out of a 600-m keyhole area by approximately 10 km in 2029, in case it is going to pass through a keyhole, to completely eliminate any possibility of its resonant return impact with the Earth in 2036.

In Table 1, the six classical orbital elements of Apophis in the JPL 140 (heliocentric ecliptic J2000 reference frame at epoch JD 2455000.5 (2009-June-18.0) TDB) are provided. Its other orbital properties are estimated as: perihelion r_p = 0.74606 AU, aphelion r_a = 1.09881 AU, perihelion speed V_p = 37.6 km/s, aphelion speed V_a = 25.5 km/s, perihelion passage time t_p = JD 2454894.9 (2009-Mar-04.41), the mean orbital rate $n = 2.2515 \times 10^{-7}$ rad/sec, and the mean orbital speed = 30.73 km/s.

Table 1: Orbital elements of asteroid Apophis at Epoch 2455000.5 (2009-June-18.0) TDB. Source: JPL's small-body database.

Orbital Elements	Value
Semimajor axis a, AU	0.92243
Eccentricity e	0.19120
Inclination i, deg	3.33142
Perihelion argument ω, deg	126.404
Right ascension longitude Ω, deg	204.442
Mean anomaly M_0, deg	117.468

3 Standard Dynamical Model

Future space missions to deflect or disrupt the trajectory of an NEO on an impending Earth impact trajectory will require the accurate prediction of its orbital trajectory.

A so-called Standard Dynamical Model (SDM) of the solar system for trajectory predictions of NEOs normally includes the gravity of the Sun, nine planets, Earth's Moon, and at least the three largest asteroids. An extended SDM with additional gravitational perturbation terms, including the Earth's oblateness, must be used for the accurate orbit prediction of asteroid Apophis for its close flyby of the Earth and for its possible passage through a 600-m keyhole in 2029.

However, significant orbit prediction error can result when we use even the extended SDM due to the unmodeled or mismodeled non-gravitational perturbation acceleration caused by the Yarkovsky effect and solar radiation pressure. The Yarkovsky effect is the thermal radiation thrust due to the anisotropic radiation of heat from a rotating body in space [13-15]. Its possible application to NEO deflection has also been proposed in [16]. Any low-energy deflection approach must consider significant orbital perturbations caused by the Yarkovsky effect as well as solar radiation pressure. Consequently, the detection and measurement of the Yarkovsky effect via precise orbit determination of NEOs based on precise radar astrometry is also of current practical interest [10].

3.1 Basic Orbital Dynamics

The standard n-body problem in celestial mechanics is the problem of determining, given the initial positions, velocities, and masses of n bodies in space, their subsequent motions as governed by Newton's laws of motion and Newton's law of gravity. In general, the equation of motion of the ith body B_i of mass m_i in space can be written as [17, 18]

$$m_i \ddot{\vec{R}}_i = \vec{F}_i - \sum_{j=1}^{n} \frac{G m_i m_j}{{R_{ij}}^3} \vec{R}_{ij} \ ; \quad i = 1, 2, ..., n \quad (1)$$

where $\vec{R}_i$ is the position vector of B_i from the inertial origin, $\ddot{\vec{R}}_i$ is the inertial acceleration of the ith body B_i, $\vec{F}_i$ is the *external* force acting on B_i, G is the universal gravitational constant, and $R_{ij} \equiv |\vec{R}_{ij}|$ where $\vec{R}_{ij} = \vec{R}_i - \vec{R}_j$ is the position vector of B_i from B_j. In n-body system simulations, the equations of motion of a system of n bodies under the influence of their mutual gravitational forces are integrated numerically without any simplifying approximations.

However, the orbital motion of an asteroid in a heliocentric orbit is often simply described by

$$\ddot{\vec{r}} + \frac{\mu}{r^3}\vec{r} = \vec{f} \quad (2)$$

where $\vec{r}$ is the position vector of the asteroid from the center of the sun, $\mu \approx \mu_\odot$ = 132,715E6 km^3/s^2, and $\vec{f}$ is the sum of all perturbing force vectors (per unit mass) acting on the asteroid.

The orbital equation of motion in vector form, Eq. (2), can be expressed in rectangular coordinates as follows:

$$\ddot{X} = -\frac{\mu X}{r^3} + f_X \quad (3a)$$

$$\ddot{Y} = -\frac{\mu Y}{r^3} + f_Y \quad (3b)$$

$$\ddot{Z} = -\frac{\mu Z}{r^3} + f_Z \quad (3c)$$

where $r = \sqrt{X^2 + Y^2 + Z^2}$ and (f_X, f_Y, f_Z) are the perturbing acceleration components along the inertial (X, Y, Z) coordinates.

A set of six first-order differential equations, called Gauss's form of the variational equations, in terms of six osculating orbital elements, are given by [17, 18]

$$\dot{a} = \frac{2a^2}{h}[eR\sin\theta + T(1 + e\cos\theta)]$$

$$\equiv \frac{2a^2}{h}[eR\sin\theta + \frac{pT}{r}] \quad (4a)$$

$$\dot{e} = \sqrt{\frac{p}{\mu}} \ [R\sin\theta + T(\cos\theta + \cos E)] \quad (4b)$$

$$\equiv \frac{1}{h}\{pR\sin\theta + [(p+r)\cos\theta + re]T\} \quad (4c)$$

$$\dot{i} = \frac{r\cos(\omega+\theta)}{h}N \quad (4d)$$

$$\dot{\Omega} = \frac{r\sin(\omega+\theta)}{h\sin i}N \quad (4e)$$

$$\dot{\omega} = -\frac{r\sin(\omega+\theta)}{h\tan i}N + \frac{1}{eh}[-pR\cos\theta + (p+r)T\sin\theta] \quad (4f)$$

$$\dot{\theta} = \frac{h}{r^2} + \frac{1}{eh}[pR\cos\theta - (p+r)T\sin\theta] \quad (4g)$$

where

$$p = a(1-e^2); \quad h = \sqrt{\mu p} = na^2\sqrt{1-e^2}; \quad n = \sqrt{\mu/a^3}$$

$$r = \frac{p}{1+e\cos\theta} \equiv a(1 - e\cos E)$$

and (R, T, N) are the components of all perturbing accelerations along the radial, tangential, and normal directions.

The n-body system simulation is simple in principle because they merely involve the integration of the 6n ordinary differential equations describing the n-body motions in Newtonian gravitational field. However, it becomes a non-trivial problem when a precise orbit prediction is required in the presence of various physical modeling uncertainties. Consequently, accurate physical/computational models of various perturbation terms, contributing to (R, T, N), for the efficient and accurate n-body system simulation of NEO trajectories for impact threat mitigation studies is of current practical interest.

3.2 N-Body Simulation Example

An example is discussed here to illustrate the inherent uncertainty issue associated with typical n-body system simulations of an NEO trajectory for impact threat mitigation studies. Consider a fictional asteroid deflection problem created by AIAA for the 2004/2005 AIAA Space Design Competition. A similar fictional asteroid deflection problem, called the Defined Threat (DEFT) scenarios, has been also created for the 2004 Planetary Defense Conference. One of the four DEFT scenarios is about mitigating a fictional 200-m Athos asteroid with the predicted impact date of February 29, 2016.

The fictional asteroid mitigation problem of AIAA is briefly described as follows. On July 4, 2004, NASA/JPL's Near Earth Asteroid Tracking (NEAT) camera at the Maui Space Surveillance Site discovered a 0.205-km diameter Apollo asteroid designated 2004WR. This asteroid has been assigned a Torino Impact Scale rating of 9.0 on the basis of subsequent observations that indicate there is a 95% probability that 2004WR will impact the Earth. The expected impact will occur in the Southern Hemisphere on January 14, 2015 causing catastrophic damage throughout the Pacific region. The

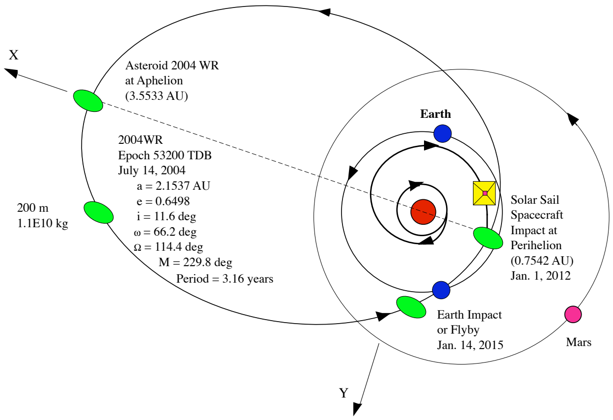

Figure 1: Illustration of the asteroid deflection problem of a fictional asteroid 2004WR [18].

mission is to design a space system that can rendezvous with 2004WR in a timely manner, inspect it, and remove the hazard to Earth by changing its orbit and/or destroying it. The classical orbital elements of 2004WR are given in the J2000 heliocentric ecliptic reference frame as follows:

$$
\begin{aligned}
\text{Epoch} &= 53200 \text{ TDB (July 14, 2004)} \\
a &= 2.15374076 \text{ AU} \\
e &= 0.649820926 \\
i &= 11.6660258 \text{ deg} \\
\omega &= 66.2021796 \text{ deg} \\
\Omega &= 114.4749665 \text{ deg} \\
M_0 &= 229.8987151 \text{ deg}
\end{aligned}
$$

The STK 5.0.4 software package, with a 9th-order Runge-Kutta integrator with variable stepsize and the planetary positions from JPL's DE405, was used by AIAA to create this set of orbital parameters of 2004WR.

Other orbital parameters of 2004WR in an ideal Keplerian orbit can be found as

$$
\begin{aligned}
r_p &= 0.7542 \text{ AU (perihelion)} \\
r_a &= 3.5533 \text{ AU (aphelion)} \\
v_p &= 44 \text{ km/s (perihelion speed)} \\
v_a &= 9.3 \text{ km/s (aphelion speed)} \\
P &= 3.16 \text{ year (orbital period)}
\end{aligned}
$$

An ideal Keplerian orbit simulation of 2004WR indicates that its closest approach to Earth is about 0.035 AU, which is less than the MOID of 0.05 AU of a PHA. It has a close encounter with Mars by 0.1 AU. Orbit simulation results using n-body software packages, including JPL's Horizons and CODES, all utilizing JPL's DE405 ephemeris data for the planetary positions indicate that 2004WR misses Earth by $1.6R_\oplus$ ($\approx 10,000$ km from the Earth center). This Earth miss-distance of approximately 10,000 km is in fact caused by the various modeling uncertainties inherent with the complex n-body orbital simulation problem.

4 The Yarkovsky Effect

This section contains a brief overview of the Yarkovsky effect. The Yarkovsky effect is the thermal radiation thrust due to the anisotropic radiation of heat from a rotating body in space [13-15]. Although it has been investigated extensively in the past, most thermal models of the Yarkovsky effect available in the literature are not well suited for typical n-body system simulations of the SDM.

The standard heat conduction in a solid body is simply described by

$$\rho C \frac{\partial T}{\partial t} = K \nabla^2 T \tag{5}$$

where T = the temperature throughout the body at any time t, ρ = the body's density, C = the specific heat, K = the thermal conductivity, and ∇^2 = the Laplace operator. The unit of this energy conservation equation is W/m^3. The boundary condition on the surface of the body is described by

$$\epsilon \sigma T^4 + K(\vec{n} \cdot \nabla T) = \alpha \mathcal{E} \tag{6}$$

where ϵ = the surface emissivity, σ = the Stehan-Boltzmann constant, α = the absorptivity of the asteroid surface (complementary to albedo), $\vec{n}$ = the unit vector normal to the body surface, and $\mathcal{E}$ = the external radiation flux. A general solution to this heat conduction problem is difficult to obtain even for a simple spherical body.

However, a simple model of the Yarkovsky force per unit mass of a rotating spherical body along the orbital flight direction is described in [13] as

$$f = \frac{2}{\rho R} \frac{\epsilon \sigma T^4}{c} \frac{\Delta T}{T} \cos \gamma \tag{7}$$

where R = the radius of the spherical body, c = the speed of light, T = average temperature of the body, ΔT = temperature change as the body rotates, and γ = the obliquity of the spin axis.

A more complex model of the diurnal variant of the Yarkovsky effect is described by Vokrouhlicky et al. [14] as

$$\vec{f} = \frac{4\alpha}{9} \frac{\Phi(r)}{1+\lambda} G[\sin\delta \frac{\vec{r} \times \vec{s}}{r} + \cos\delta \frac{\vec{s} \times (\vec{r} \times \vec{s})}{r}] \tag{8}$$

where $\vec{s}$ = the unit vector of the spin axis. The standard radiation force factor $\Phi(r)$ is defined as

$$\Phi(r) = \frac{\mathcal{E}(r)}{4R\rho c} \tag{9}$$

where $\mathcal{E}(r)$ = the solar radiation flux on a body at a distance r from the Sun described by

$$\alpha\mathcal{E}(r) = \epsilon\sigma T^4(r) \tag{10}$$

The magnitude G and the phase angle δ in Eq. (10) are defined as

$$Ge^{i\delta} \equiv \frac{A(X) + iB(X)}{C(X) + iD(X)} \tag{11}$$

where $i = \sqrt{-1}$, $X = \sqrt{2}R/\ell_s$, $\ell_s = \sqrt{K/(\rho_s C\omega)}$ = the thermal length, and ρ_s = the surface density. The auxiliary functions employed in Eq. (11) can be found in [14].

An analytical estimation of the semi-major axis drift due to the diurnal variant of the Yarkovsky effect is given in [14] as

$$\frac{da}{dt} \approx -\frac{8\alpha}{9n}\Phi(a)\frac{G\sin\delta}{1+\lambda}\cos\gamma \tag{12}$$

where $\lambda = \Theta/X$ and the diurnal thermal parameter Θ is defined as

$$\Theta = \frac{\sqrt{K\rho_s C\omega}}{\epsilon\sigma T^3(r)} \tag{13}$$

More details of this thermal model can be found in [14]. Note that the diurnal acceleration is perpendicular to the spin axis and that the surface thermal conductivity is the principal unknown parameter of this thermal model.

As described in [15] for numerical evaluation of the Yarkovsky effect on an irregularly-shaped body, a body can be divided into discrete cells and the heat equation can be solved using a finite-difference method. After the thermal state of the body is evaluated for a given time step, the net radiative reaction force, $\vec{F}$, can be found as

$$\vec{F} = \sum_{i=1}^{n} \vec{f}_i A_i \tag{14}$$

where $\vec{f}_i$ = the force per unit area of the ith cell. Finally, the net Yarkovsky force vector becomes

$$\vec{F} = \sum_{i=1}^{n} \frac{2}{3}\frac{\epsilon_i\sigma T_i^4}{c}\vec{n}_i A_i \tag{15}$$

The rates of change of the orbital elements can then be computed by decomposing this Yarkovsky force vector into (R, T, N) in Eq. (4) and then numerically integrating the Gauss's form of the variational equations.

A recent study by Giorgini et al. [10] shows that the Yarkovsky effect and solar radiation pressure can cause 20 - 740 km of position change of Apophis over the next 22 years leading into the Earth flyby in 2029. Furthermore, this change will result in a 520,000 - 30,000,000 km (0.0035 - 0.2 AU) position change in 2036. It was also found in [10] that small uncertainties in the masses and positions of the planets and the Sun can cause up to 23 Earth radii of prediction error for Apophis by 2036. Although new observations of Apophis prior to 2029 could reduce such large orbit prediction errors, further physical characterization of Apophis and its accurate orbit simulation will certainly help refine the 2036 impact probability estimation.

It is important to note that any NEO deflection effort must produce an actual orbital change much larger than predicted orbital uncertainties from all sources. An accurate estimation/characterization of the various uncertainties will be essential for determining whether a space mission to disrupt or deflect the trajectory of Apophis is warranted. Precise orbit determination of Apophis will be needed using precise astrometry data from the Arecibo radar, which might be available during its 2012-2013 apparition.

5 Asteroid Deflection Formulas

In this section we employ Clohessy-Wiltshire-Hill equations to discuss the fundamentals of asteroid deflection dynamics.

Consider the Clohessy-Wiltshire-Hill equations of motion of a target asteroid (in an assumed heliocentric circular orbit) described by [7, 17, 18]

$$\ddot{x} = 2n\dot{y} + A_x \tag{16}$$
$$\ddot{y} = -2n\dot{x} + 3n^2 y + A_y \tag{17}$$

where (x, y) are the in-plane coordinates of an asteroid with respect to the origin of a circular orbit reference frame and (A_x, A_y) are the perturbation acceleration components acting on the asteroid. The x-axis is along the negative orbital flight direction and the y-axis is along the radial direction. The out-of-plane orbital motion is not considered here. A simple case with $A_x = A$ = constant and $A_y = 0$ is further assumed here without loss of generality because the asteroid deflection effect of a nonzero A_y is often negligible.

Integrating the x-axis equation, we obtain

$$\dot{x}(t) = \dot{x}(0) + 2ny + At \tag{18}$$

where $\dot{x}(0)$ denotes the along-track velocity at $t = 0^-$. All other initial conditions will be ignored here. For a kinetic energy impactor problem, the initial impact ΔV along the x-axis direction becomes $\dot{x}(0)$.

Substituting Eq. (18) into the y-axis equation, we obtain

$$\ddot{y} + n^2 y = -2n\dot{x}(0) - 2nAt \tag{19}$$

Its solution can be found as

$$y(t) = -\frac{2}{n}\dot{x}(0)(1 - \cos nt) - \frac{2}{n}A\left(t - \frac{1}{n}\sin nt\right)$$

We then obtain

$$
\begin{aligned}
x(t) = & -\dot{x}(0)\left(3t - \frac{4}{n}\sin nt\right) - \frac{3}{2}At^2 \\
& + \frac{4}{n^2}A(1-\cos nt) \\
\approx & -3\dot{x}(0)t - \frac{3}{2}At^2 \quad \text{for large t}
\end{aligned} \tag{20}
$$

The orbital "amplification" factor of three can be seen from the preceding equation. Note that the positive values of $\dot{x}(0)$ and A slow down the asteroid and reduce its orbital energy. Consequently, its along-track position becomes negative (i.e., ahead of its unperturbed virtual position in a circular reference orbit).

Consider an asteroid with the accelerated time of t_a by a constant acceleration A and the additional coasting time of t_c with $A = 0$. It is assumed that $\dot{x}(0) = 0$ here. A new set of initial conditions at the end of accelerated period become:

$$
\begin{aligned}
x_0 = & -\frac{3}{2}At_a^2 + \frac{4}{n^2}A(1-\cos nt_a) \\
\dot{x}_0 = & -3At_a + \frac{4}{n}A\sin nt_a \\
y_0 = & -\frac{2}{n}A\left(t_a - \frac{1}{n}\sin nt_a\right) \\
\dot{y}_0 = & -\frac{2}{n}A(1-\cos nt_a)
\end{aligned}
$$

The final position changes at the end of the coasting phase can then be found as

$$
\begin{aligned}
\Delta x = & \, x_0 + (6ny_0 - 3\dot{x}_0)t_c + \frac{2\dot{y}_0}{n}(1-\cos nt_c) \\
& + \left(\frac{4\dot{x}_0}{n} - 6y_0\right)\sin nt_c
\end{aligned} \tag{21}
$$

$$
\begin{aligned}
\Delta y = & \, 4y_0 - \frac{2\dot{x}_0}{n} + \left(\frac{2\dot{x}_0}{n} - 3y_0\right)\cos nt_c \\
& + \frac{\dot{y}_0}{n}\sin nt_c
\end{aligned} \tag{22}
$$

Finally, we obtain

$$
\Delta x \approx -\frac{3}{2}At_a(t_a + 2t_c) \tag{23}
$$

which is the low-thrust deflection formula for NEOs disturbed by a constant acceleration A for the period of t_a and also with the additional coasting period of t_c with $A = 0$ [4, 7]. Note that $\Delta y \approx 0$ compared to Δx.

Equation (23) can be rewritten as

$$
\Delta x = -\left(\frac{3}{2}At_a^2 + \Delta V t_c\right) \quad \text{where } \Delta V = 3At_a \tag{24}
$$

Note that Δx is caused by various initial conditions including $\dot{x}_0$ and y_0 as can be seen in Eq. (21). Such a combined effect of $\dot{x}_0$ and y_0 results in the term $\Delta V t_c$ (not $3\Delta V t_c$ as one might expect) in Eq. (24).

For an impulsive ΔV along the x-axis direction, the resulting deflection Δx after a coasting time of t_c is simply given by

$$
\Delta x = -3\Delta V t_c \tag{25}
$$

For a kinetic impactor approach, ΔV can be estimated as

$$
\Delta V \approx \beta\frac{m}{M+m}U \approx \beta\frac{m}{M}U \tag{26}
$$

where β is the impact efficiency factor, m the impactor mass, M the target asteroid mass, and U the relative impact velocity.

The Clohessy-Wiltshire-Hill equations of motion of a target asteroid in an elliptical orbit are also given by

$$
\ddot{x} = 2\dot{\theta}\dot{y} + \ddot{\theta}y + \dot{\theta}^2 x - \frac{\mu}{r^3}x + A_x \tag{27}
$$

$$
\ddot{y} = -2\dot{\theta}\dot{x} - \ddot{\theta}x + \dot{\theta}^2 y + \frac{2\mu}{r^3}y + A_y \tag{28}
$$

$$
\ddot{r} = r\dot{\theta}^2 - \frac{\mu}{r^2} \tag{29}
$$

$$
\ddot{\theta} = -\frac{2\dot{r}\dot{\theta}}{r} \tag{30}
$$

where (x, y) are the relative coordinates of the target asteroid with respect to a reference point of its nominal elliptical orbit, r is the radial distance of the reference orbit from the sun, θ is the true anomaly, and μ is the gravitational parameter of the sun. Furthermore, we have

$$
r = \frac{p}{1 + e\cos\theta} \tag{31a}
$$

$$
\dot{r} = \sqrt{\mu/p}\,(e\sin\theta) \tag{31b}
$$

$$
\dot{\theta} = \sqrt{\mu/p^3}\,(1 + e\cos\theta)^2 \tag{31c}
$$

where $p = a(1-e^2)$.

For Apophis with e = 0.1912 and with an assumed perturbation acceleration of $A_x = 3.8 \times 10^{-10}$ mm/s^2, $A_y = 5.4 \times 10^{-10}$ mm/s^2, t_a = 5 years, and t_c = 3 years, the eccentricity effect on the relative orbital distance x is evident in Fig. 2.

To illustrate the significant effect of a large eccentricity, consider a 200-m asteroid, with $M = 1.1 \times 10^{10}$ kg, a = 2.1537 AU, e = 0.6498, $A_x = 1.7 \times 10^{-9}$ mm/s^2, $A_y = 2.5 \times 10^{-9}$ mm/s^2. Simulation results for t_a = 10 years and t_c = 12 years are shown in Fig. 3. The significant effect of a large eccentricity (e = 0.6498) on the relative orbital distance x is evident in this figure.

6 NEO Deflection Options

Early detection, accurate tracking, reliable precision orbit calculation, and characterization of physical properties of NEOs are prerequisites to any space mission of

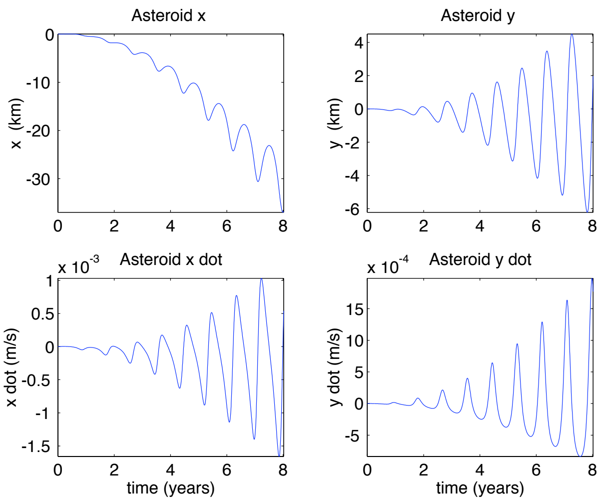

Figure 2: Long-term deflection simulation of Apophis with e = 0.1912, a perturbation force of 30 mN, t_a = 5 years, and t_c = 3 years.

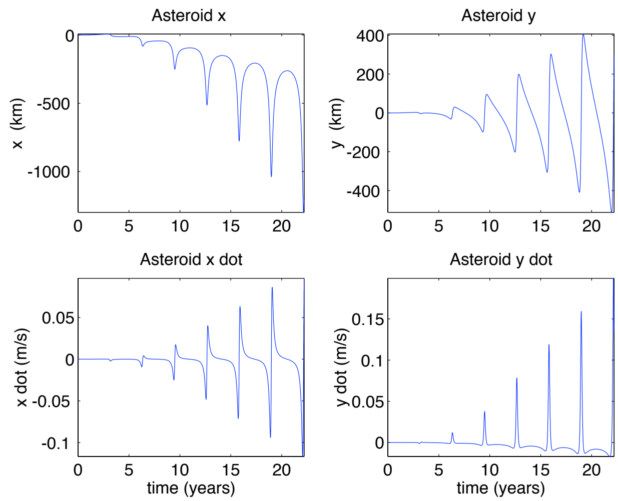

Figure 3: Long-term deflection simulation of a fictional 200-m asteroid with e = 0.6498 and a perturbation force of 30 mN.

deflecting NEOs. The early discovery of NEOs prior to impact using current ground-based optical sensors is not assured, and detection/tracking of small (500 m or less) NEOs is a difficult task given their low albedo and small size. Various concepts and approaches for advanced ground-based as well as space-based detection systems are being developed to allow for adequate warning time.

Assuming that NEOs on a collision course can be detected prior to impact with a mission lead time of at least 10 years, however, the challenge becomes eliminating their threat, either by destroying the asteroid, or by altering its trajectory so that it will miss Earth. A variety of schemes have been already extensively investigated in the past for such a technically challenging, asteroid deflection problem [1-8]. The feasibility of each approach to deflect an incoming hazardous object depends on its size, spin rate, composition, the mission lead time, and many other factors.

6.1 Nuclear Standoff Explosions

In practice, deflection methods of sufficiently high energy density are preferred and need to be prepared in advance of an expected impact date with the Earth. One of these methods utilizes a nuclear explosion at a specified standoff distance from the target NEO to cause its velocity change by ablating and blowing off a thin layer of the surface. The basic physical fundamentals of such nuclear standoff explosions can be found in Ref. 1 (pp. 897-1033) and Ref. 5 (pp. 113-140). System design aspects of employing nuclear standoff explosions for NEO deflection can also be found in [19-23].

Nuclear standoff explosions are often assessed to be much more effective than the non-nuclear alternatives, especially for larger asteroids with a short mission lead time [6]. Other techniques involving the surface or subsurface use of nuclear explosives are also assessed to be more efficient, although they may run an increased risk of fracturing the target asteroid. However, the nuclear approach needs more rigorous studies to verify its overall effectiveness and determine its practical viability. The nuclear standoff explosions require an optimal standoff distance for a maximum velocity change of a target asteroid. Therefore, we have to determine how close the nuclear explosion must be to effectively change the orbital trajectories of asteroids of different types, sizes, and shapes. The precise outcome of a NEO deflection attempt using a nuclear standoff explosion is dependent on myriad variables. Shape and composition of the target NEO are critical factors. These critical properties, plus others, would need to be characterized, ideally by a separate mission, prior to a successful nuclear deflection attempt. High-fidelity physical models to reliably predict the velocity change and fragmentation caused by a nuclear standoff explosion will need to be developed.

A simple model that can be used to assess the effectiveness of a nuclear standoff explosion approach was recently examined in [24]. Geometric principles and basic physics were used in [24] to construct a simple model which can be augmented to account for icy bodies, anisotropic ejecta distributions, and effects unique to the nuclear blast model. Use of this simple model has resulted in an estimation of NEO velocity change of about 1 cm/s on the same order as other complex models, and has correlated data for optimal standoff distance of about 200 m for an ideal spherical model of a 1-km NEO. More rigorous physical modeling and simulation, including hydrodynamic codes and other forms of computer modeling, will be necessary to account for changes in material properties under the realistic conditions of the nuclear blast. Possible fracturing of the asteroid and other anticipated outcomes of a nuclear blast are also needed for a further study.

6.2 Kinetic Impactors

A non-nuclear approach does currently exist for an impulsive velocity change, caused by the targeted kinetic impact of a spacecraft on the target asteroid's surface. If applied correctly without causing fragmentation of a large asteroid into smaller pieces and if applied long enough prior to a projected Earth impact, the effect of such an impulsive ΔV would magnify over decades (or even centuries), eliminating the risk of collision with Earth. To be most effective, the impacting spacecraft would either have to be massive, or be moving very fast relative to the asteroid. Since current launch technology limits the mass (including propellant) that can be lifted into an interplanetary trajectory, we are therefore led to consider designs that would maximize impact velocity, and which would not require large amounts of fuel.

The recent success of NASA's Deep Impact mission in 2005 has significantly enhanced the practical viability of the kinetic-impact concept. Its mission goals were to explore the internal structure and composition of the nucleus of comet Tempel 1 before, during, and after impacts, and to return the observations to Earth. The Deep Impact spacecraft was launched by a Delta II launch vehicle on January 12, 2005 and released a 370-kg impactor spacecraft which collided with Tempel 1 on July 4, 2005 to create a large crater on the surface of the 5-km target comet. The crater is estimated to be 20-m deep and 100-m wide. In fact, the 5-km comet with a heliocentric speed of 29.9 km/s crashed into the 370-kg impactor which was moving at a slower heliocentric speed of 22.4 km/s. This resulted in a rear-end collision of the impactor spacecraft at a 10 km/s impact speed but with an impact approach angle of 15 deg. The kinetic energy of the impactor was $1.9\ \times 10^{10}$ J and the resulting impact ΔV was practically zero. The Deep Impact mission was not intended to deflect the orbit of such a large 5-km comet. The attitude/position of the impactor spacecraft after being released from the flyby spacecraft was precisely controlled by the autonomous optical navigation system to achieve a 300-m targeting accuracy.

A somewhat futuristic, solar sailing mission concept utilizing a 160-m solar sail to deliver a 150-kg kinetic impactor into a heliocentric retrograde orbit was studied in [25-28]. Such kinetic impactors will result in a head-on collision with a target asteroid at its perihelion (as illustrated in Fig. 1), thus increasing its impact velocity to at least 70 km/s. The NEAR Shoemaker study of asteroid Mathilde and the Japanese Hayabusa mission for exploring the asteroid Itokawa suggest that many asteroids are essentially "rubble piles." Consequently, a practical concern of any impulsive approaches employing kinetic impactors or nuclear explosions is the risk that such high-energy deflection attempts could result in the fragmentation of NEOs, which could substantially increase the damage upon Earth impact.

6.3 Gravitational Binding Energy

In astrophysics, the energy required to disassemble a celestial body consisting of loose material, which is held together by gravity alone, into space debris such as dust and gas is called the gravitational binding energy.

The gravitational binding energy of a spherical body of mass M, uniform density ρ, and radius R is given by

$$E = \frac{3GM^2}{5R} = \frac{3G}{5R}\left(\frac{4\pi\rho R^3}{3}\right)^2 = \frac{\pi^2\rho^2 G}{30}D^5 \quad (32)$$

where G = 6.67259×10^{-11} N·m^2/kg^2 is the universal gravitational constant and $D = 2R$ is the diameter of a spherical body. The escape speed from its surface is given by

$$V_e = \sqrt{\frac{2GM}{R}} \quad (33)$$

For example, for a 200-m (diameter) asteroid with a uniform density of ρ = 2,720 kg/m^3 and a mass of $M = 1.1 \times 10^{10}$ kg, its gravitational binding energy is estimated to be 4.8×10^7 J. Since the kinetic energy of a 150-kg impactor at an impact velocity of 70 km/s is 3.7×10^{11} J, one may expect that a gravity-dominated, 200-m asteroid would be disrupted and dispersed by such a high-energy impactor. However, its escape velocity of 12 cm/s is about 120 times the impact ΔV of 0.1 cm/s. This large ratio of the escape velocity to the impact ΔV may suggest that if the asteroid disperses, the resulting fragments might scatter around their deflected center of mass [19].

In Ref. 1 (pp. 897-927) and Ref. 5 (pp. 135-136), the disruption energy per unit asteroid mass is predicted to be 150 J/kg for strength-dominated asteroids. This indicates that a strength-dominated, 200-m asteroid would not be disrupted by a 150-kg impactor at a high impact velocity of 70 km/s. Also in Ref. 5 (pp. 135-136), the energy (per unit asteroid mass) required for both disruption and dispersion of a 1-km asteroid is predicted to be 5 kJ/kg. Thus, the feasibility of the most kinetic-impact approaches for either disrupting or deflecting an incoming NEO depends on its size and composition (e.g., solid body, porous rubble pile, etc.), as well as the time available to change its orbit. An accurate determination of the composition of the target asteroid is a critical part of the kinetic-impact approaches, which may require a separate inspection mission.

A further study is also needed to optimize impactor size, relative impact velocity, and the total number of

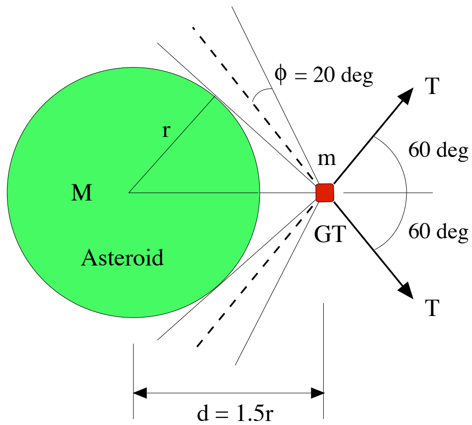

Figure 4: A geometrical illustration of the gravity tractor (GT) concept for towing an asteroid. A simple spherical body is considered for the concept illustration purpose without loss of generality.

impactors as functions of asteroid size and composition, to ensure a deflection attempt does not cause fragmentation/dispersal.

6.4 Gravity Tractors

Lu and Love [29] have proposed a low-energy asteroid deflection concept utilizing the mutual gravitational force between a hovering spacecraft and a target asteroid as a towline as illustrated in Fig. 4. To avoid exhaust plume impingement on the asteroid surface, two ion engines are properly tilted outward and the hovering distance is accordingly selected as: $d = 1.5r$ and $\phi = 20$ deg. This illustrative combination yields an engine cant angle of 60 deg, and the two tilted thrusters (each with a thrust T) then produce a total towing thrust T as illustrated in Fig. 4.

Although a large 20-ton gravity-tractor (GT) spacecraft propelled by a nuclear-electric propulsion system is considered in [29], a smaller 1000-kg GT spacecraft is capable of towing a certain class of near-Earth asteroids such as asteroid 99942 Apophis [7, 30]. It is interesting to notice that such a gravitational coupling/towing concept has been previously proposed for somewhat science-fictional, astronomical problems by Shkadov [31] in 1987 and also by McInnes [32] in 2002.

A simplified dynamical model of the GT for towing asteroid Apophis (with an assumed diameter of 320 m) is given by

$$M\frac{\Delta V}{\Delta t} = \frac{GMm}{d^2} = T \tag{34}$$

or

$$\frac{\Delta V}{\Delta t} = \frac{Gm}{d^2} = \frac{T}{M} = A \tag{35}$$

where $G = 6.6695 \times 10^{-11}$ N·m^2/kg^2, $M = 4.6 \times 10^{10}$ kg, $m = 1000$ kg, r = 160 m. d = 240 m, $T = 0.053$ N, $A = 1.1579 \times 10^{-9}$ mm/s^2 is the characteristic acceleration, and

$$\Delta V = A\Delta t \tag{36a}$$

$$\Delta X = \frac{1}{2}A(\Delta t)^2 \tag{36b}$$

where ΔV and ΔX are, respectively, the resulting velocity and position changes for the total towing period of Δt. For example, we have ΔV = 0.036 mm/s and ΔX = 575 m for Δt = one year.

Including the orbital "amplification" effect as discussed in Section 5, we have

$$\Delta V = 3A\Delta t \tag{37a}$$

$$\Delta X = \frac{3}{2}A(\Delta t)^2 \tag{37b}$$

Consequently, we have ΔV = 0.1 mm/s and ΔX = 1.7 km for one-year towing by a 1000-kg GT.

Including an additional coasting time of t_c, we have the total position change (i.e., the Earth miss-distance) given by

$$\Delta X = \frac{3}{2}A\Delta t(\Delta t + 2t_c) \tag{38}$$

Thus, one-year towing in 2026 with an additional coasting time of 3 yrs will cause a total position change of approximately 12 km in 2029, which may be considered to be sufficient to safely move Apophis out of its 600-m keyhole in 2029. However, it is important to note that any NEO deflection effort must produce an actual orbital change much larger than predicted orbital uncertainties from all perturbation sources, including the Yarkovsky effect.

The propellant amount required for maintaining a desired hovering altitude of 80 m for a 1000-kg GT can be estimated as

$$\Delta m \approx \frac{2T\Delta t}{g_o I_{sp}} \approx 0.3 \text{ kg per day } \approx 114 \text{ kg per year}$$

where $T = 0.053$ N, $g_o = 9.8$ m/s^2, and $I_{sp} = 3000$ sec (assumed for typical ion engines).

Therefore, a 1000-kg GT spacecraft equipped with ion engines can be considered as a viable option for a pre-2029 deflection mission for Apophis. However, it is emphasized that a 1000-kg spacecraft, colliding with Apophis at a modest impact velocity of 10 km/s in 2026, will cause a much larger, instantaneous velocity change of at least 0.22 mm/s for Apophis, resulting in an orbital deflection of 62 km in 2029. Such a higher-energy kinetic impactor approach may not be applicable to highly porous, rubble-pile asteroids, while a GT spacecraft mission may need an additional large ΔV to rendezvous with a target asteroid. Consequently, further system-level tradeoffs on various practical issues, such as the total mission ΔV requirement, low-thrust gravity towing vs. higher-energy kinetic impact, and asteroid dispersal/fragmentation concern, need to be performed.

6.5 Multiple Gravity Tractors (MGTs) in Halo Orbits

Because a GT spacecraft hovering in a static equilibrium standoff position requires canted thrusters to avoid plume impingement on the NEA surface, McInnes [33] has proposed a GT spacecraft flying in a displaced non-Keplerian orbit (also often called a halo orbit) for a possible fuel-efficient way of towing asteroids. Such a GT spacecraft (not requiring canted thrusters) in a displaced orbit in fact has a fuel-efficient advantage over a single hovering GT spacecraft requiring two canted thrusters. However, a GT in a displaced orbit will require a much heavier spacecraft (about 2.8 times heavier than a single hovering GT) if its orbital displacement is the same as the standoff distance of a hovering GT. Or it will need to be placed much closer to the target asteroid (at about 59% of the standoff distance of a hovering GT) if its mass is the same as the mass of a hovering GT. Despite such drawbacks, a practical significance of a displaced orbit is that it simply allows many gravity tractors near a target asteroid, resulting in a larger total ΔV capability, multi-spacecraft redundancy, and mission design flexibility with smaller satellites equipped with lower-risk propulsion systems. Consequently, a system of multiple gravity tractors flying in halo orbits near a target asteroid can be considered as a viable near-term option for deflecting a certain class of NEAs such as asteroid 99942 Apophis or other highly porous, rubble-pile asteroids

An MGT system consisting of several GTs in a primary halo orbit as well as in a secondary/backup halo orbit is illustrated in Fig. 5. More detailed discussions of the MGT system can be found in [7, 34].

6.6 A Hovering Solar-Sail Gravity Tractor (SSGT)

Utilizing the same physical principle of gravitationally "anchoring" the spacecraft to the asteroid, without physical contact between the spacecraft and the asteroid, we may employ solar sails/reflectors rather than nuclear- or solar-electric propulsion systems to produce the required continuous low-thrust force. Such a solar-sail gravity tractor, described in [7, 35], exploits the propellantless nature of solar sails/reflectors for towing asteroids, despite its inherent drawback of requiring an offset hovering position of 55 deg from an asteroid's flight direction.

For a solar sailing kinetic impactor mission proposed in [26-28], its large lightweight solar sail is to be deployed at the beginning of an interplanetary solar sailing

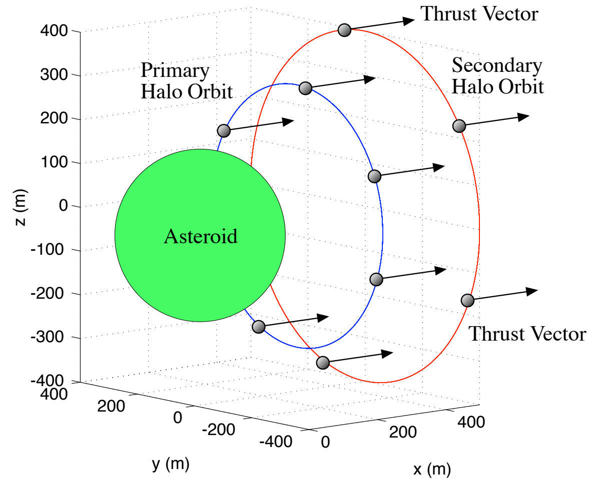

Figure 5: A conceptual illustration of a system of multiple gravity tractors (MGTs) in halo orbits.

flight toward a target asteroid and the impactor spacecraft will be separated from the solar sail prior to impacting a target asteroid. For an SSGT spacecraft mission, its large solar sail is to be deployed after completing a rendezvous with a target asteroid; however, its large solar sail/reflector is not required to be lightweight.

A simple planar model of the hovering dynamics of an SSGT spacecraft towing a target asteroid is illustrated in Fig. 6. Utilizing the Clohessy-Wiltshire-Hill equations of motion, we can obtain the orbital equations of motion of the asteroid-SSGT system orbiting around the sun as follows:

$$\ddot{x}_1 = 2n\dot{y}_1 + Gm_2 \frac{x_2 - x_1}{r^3} \tag{39}$$

$$\ddot{y}_1 = -2n\dot{x}_1 + 3n^2 y_1 + Gm_2 \frac{y_2 - y_1}{r^3} \tag{40}$$

$$\ddot{x}_2 = 2n\dot{y}_2 - Gm_1 \frac{x_2 - x_1}{r^3} + \frac{1}{m_2}(T_x + F_x) \tag{41}$$

$$\ddot{y}_2 = -2n\dot{x}_2 + 3n^2 y_2 - Gm_1 \frac{y_2 - y_1}{r^3} + \frac{1}{m_2}(T_y + F_y) \tag{42}$$

where (x_1, y_1) are the coordinates of the target asteroid with respect to an orbiting reference frame, (x_2, y_2) the coordinates of the SSGT spacecraft, (T_x, T_y) solar pressure thrust components, (F_x, F_y) control thrust components, $r = \sqrt{(x_2 - x_1)^2 + (y_2 - y_1)^2)}$, G = the universal gravitational constant, m_1 the asteroid mass, m_2 the SSGT spacecraft mass, and n the orbital rate of the reference frame. For simplicity, a circular orbital motion of the reference frame is assumed here.

As an example, for Apophis, we may need a 2500-kg SSGT spacecraft, equipped with a 90×90 m solar sail of a 0.03-N solar thrust with a 35-deg sun angle as

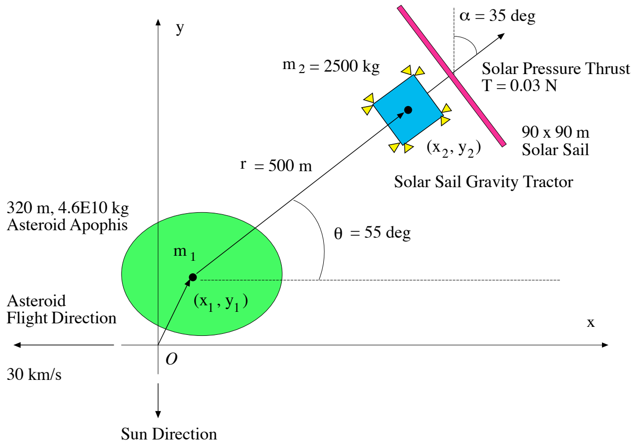

Figure 6: A simplified dynamical model for hovering control design of a solar-sail gravity tractor (SSGT) spacecraft [7, 35].

illustrated in Fig. 6. Such a particular hovering position with an offset angle of $\theta = 55$ deg from an asteroid's flight direction is necessary because of the 35-deg sun angle requirement of a typical solar sail/reflectorto produce a maximum solar pressure thrust. A larger (2500-kg) SSGT, compared to a 1000-kg GT equipped with ion engines, is to be placed at a higher altitude of 350 m because of its large solar sail/reflector. This 2500-kg SSGT produces an along-track acceleration of $A_x = 3.8 \times 10^{-10}$ mm/s^2 and a radial acceleration of $A_y = 5.4 \times 10^{-10}$ mm/s^2 of the target asteroid Apophis. However, its radial acceleration component A_y has a negligible effect on the asteroid deflection. This fact can be considered as an inherent drawback of the SSGT although a solar sail/reflector is a propellantless propulsion system. More detailed discussions of the solar sail/reflector gravity tractors can be found in [7, 35, 36]. Again, it is emphasized that the solar sail/reflector is not required to be lightweight.

6.7 Recent Studies on Asteroid Exploration, Deflection, and Fragmentation

Recent study results on a crewed exploration mission to NEOs, an Interplanetary Ballistic Missile (IPBM) system architecture for NEO deflection, and orbital dispersion simulation of NEOs fragmented by nuclear explosions can be found in [36-40].

7 Conclusions

This paper for the John V. Breakwell Memorial Lecture at the 60th International Astronautical Congress (IAC) has presented a tutorial overview of the astrodynamical problem of deflecting a near-Earth object (NEO) that is on a collision course toward Earth. This paper has focused on the astrodynamic fundamentals of such a technically challenging, complex engineering problem. Now is the time to develop practically viable planetary defense systems to assure a high quality of life for future generations.

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