DEFLECTION OF NEAR EARTH OBJECTS BY MEANS OF TETHERS

Vladimir Chobotov, Ph.D.
The Aerospace Corporation, U.S.A
Vladimir.Chobotov@aero.org

Nahum Melamed, Ph.D.
The Aerospace Corporation, U.S.A
Nahum.Melamed@ aero.org

Once an Earth collision threat posed by a NEO (Near Earth Object), has been determined and the corresponding required Earth collision avoidance deflection delta V obtained, the problem of implementing an orbit modification maneuver remains to be identified. One approach may be termed a soft collision impact where tethers or net type devices attach a deflection projectile to the NEO to modify its orbit. This study examines an approach which involves anchoring a tether with a tip mass, the relative velocity of which is rotated in an optimum direction relative to the velocity of the NEO. After release of the tip mass, the NEO velocity is retarded similar to that of a gravity assist maneuver. The corresponding loss of NEO momentum represents a diversion delta V that is of a sufficient magnitude to alter the flight path of the NEO and thus avoid a collision with Earth.

The study identifies the required delta V for a representative NEO that may be obtained by Kevlar and carbon nanotube tether materials. Maximum NEO size and mass characteristics that can be deflected are also derived.

1. INTRODUCTION

The current study examines the capability of a tether-assisted method to deflect a Near Earth Object (NEO) and mitigate its Earth collision threat. The idea of using space tethers to perform trajectory modifications of space objects is not new. Ref's. 1 and 2 examine using asteroids to deflect a spacecraft trajectory in a manner similar to a gravity assist maneuver. The approaching spacecraft would deploy and anchor a tether to the asteroid, causing a momentum exchange between the two and modifying the trajectories of both. The change in the asteroid orbit is neglected in the study conducted in Ref. 1 because the mission goal is to modify the trajectory of the spacecraft, whose mass is assumed to be much smaller than that of the asteroid.

Ref. 3 estimates that the minimum required delta velocity that must be imparted to a NEO to result in an acceptable Earth miss distances on the order of several cm/sec. Naturally, the total required impulse to be imparted to the asteroid would depend on its size. Moreover, Ref. 3 estimates the minimum required delta V in the presence of inherent uncertainty in estimating the NEO trajectory, which translates, in turn, to a miss distance uncertainty. It is shown there that a typical increase by about a factor of 2 of a deterministic estimate of delta V would produce acceptable levels of collision risk reduction for three of the asteroids that were studied.

The results presented in Ref's. 1 through 3 provide the motivation for the current study. By reversing the goal of Ref. 1 from that of spacecraft trajectory modification to that of changing the asteroid's orbit sufficiently to provide an adequate Earth miss distance with acceptable risk of collision. The advantage of the proposed method is that the entire asteroid can be diverted rather than fractured by a direct impact. The fracture could result in a shotgun type impact with unpredictable consequences. Moreover, this technique would impart twice the momentum as a direct inelastic impact since it is equivalent to an elastic collision.

Ref. 1 points out one of the main limiting factors of this approach-- the finite material strength of the tether cable. This constraint limits the relative approach velocity between the payload and the asteroid, which in turn, limits the amount of momentum exchange that can occur between the two thus severely restricting the NEO size that can be handled by this approach.

The interaction time between the spacecraft and the NEO is on the order of minutes, so another problem that must be overcome is getting the tether anchored in the asteroid very quickly and keeping it there under the enormous tension. One method may be a net type capture device, which could apply to smaller asteroids such as those considered in the paper.

The severity of the constraints described above may be alleviated if the maneuver could be performed sequentially as a series of tethered payload encounters the asteroid. An alternative mitigation approach may be considered if the asteroid is fractured by a direct impact into smaller rigid fragments moving away from each other either deliberately or inadvertently. It is assumed here that the asteroid is not a loose aggregate of rubble.

Ref's. 4 and 5 bring to notice an assessment that asteroid fragmentation rather than deflection may still have serious implications due to terrestrial biosphere vulnerability to severe perturbation from impacts. This may be evident in terms of an increased cratering efficiency of several smaller fragments rather than by a single large object, or that an atmospheric impact may manifest as an increased surface temperature with catastrophic consequences.

If smaller fragments, however, could be handled separately to assist in their dispersion before reaching Earth by using the approach described in this paper, these concerns may be somewhat alleviated. Ref. 6 points out that due to the asteroid structure in some cases, such a break-up may not be an especially energetic phenomenon, suggesting that developing an appropriate asteroid impacting strategy may be achievable. In the event of large fragments that remain gravitationally bound and re-aggregate following a collision impact, each piece may still be pulled apart and handled separately by the proposed method.

2. ASTEROID TRAJECTORY MODIFICATION

2.1 Gravity Assist

Fly-bys, or gravity assist maneuvers, are a standard technique in spaceflight and are used by many interplanetary missions in order to send space payloads to the far solar system using as little fuel as possible. The maneuver is characterized by an interaction between a spacecraft and a space object like a planet or an asteroid. That interaction, whether hard (eg., crash or land) or soft (eg., flyby), obeys Newton's Third Law and the total momentum of the system is conserved. In the context of space exploration missions, trajectory modification techniques such as fly-bys can provide the crucial benefit needed to accomplish mission goals. The momentum exchange that occurs with the planet is utilized as a propellant-less transportation technique to boost the payload to higher or lower energy orbits.

As a spacecraft approaches a planet, the forces between them become appreciably strong in the context where the two objects are considered, to a close approximation, as isolated from all other objects in space (e.g., Sphere of Influence). Since the spacecraft comes from a great distance, it can be assumed to be following a hyperbolic orbit with respect to the center of gravity of the planet. The gravitational force attracts them equally, but in opposite directions. From the planet's point of view, its accelerating gravitational force makes a significant change in the direction of the spacecraft's velocity, but not in its magnitude (See Figure 1).

The planet, however, has a very large angular momentum as it revolves around the Sun. During its interaction with the planet the spacecraft acquires a significant portion of the planet's velocity vector relative to the Sun. Figure 2 shows how the spacecraft trajectory is bent by the planet's gravity, which also helps to increase its final velocity. The spacecraft tugs on the planet and decreases its orbital momentum by a tiny amount. In exchange, the spacecraft acquires a significant amount of momentum from the planet, compared to the momentum it already had.

The main shortcoming of the gravity assist maneuver is its limited availability. The planets are few and must be properly aligned to provide the required gravity assist advantage for interplanetary missions.

Asteroids, however, are more abundant in the solar system. Ref. 1 and later studies proposed a method to exploit that resource to modify spacecraft trajectories. Although these space objects are too small to provide any useful gravity assist capability, an alternative approach of inducing momentum exchange between the spacecraft and the asteroid is for the spacecraft to hook itself to the asteroid using a tether and anchoring device. The spacecraft would then swing around the asteroid for a short duration of time relative to the orbital period and then be released to fly to its new destination.

Figure 3 shows the three phases of the maneuver: spacecraft attachment to the asteroid, swinging by the asteroid and detachment. Due to the very large mass ratio between the asteroid and the spacecraft, the impact on the latter would be much more pronounced than on the former. On the other hand, from the point of view of planetary, could that small change of the NEO's orbit be sufficient to eliminate a threatening impact?

2.2 Tether Assist Asteroid Deflection

As noted earlier, the main limitation of a tether assist spacecraft deflection maneuver is tether material strength. The same constraint places a bound on the effectiveness of a tether assisted asteroid deflection maneuver and essentially restricts the size of an asteroid that can be handled by this approach.

As the spacecraft approaches the asteroid, it would hook itself to the asteroid using the tether and an anchoring device and would swing in a circular arc about it. To prevent excessive tension immediately after attachment, the tether line must be nearly perpendicular to the surface of the asteroid at the instant when anchoring occurs.

During the ensuing swing about the asteroid, the relative velocity between the spacecraft and the asteroid is constrained by current tether material (assumed to be of the Kevlar or carbon nanotube type) to an order of 1-3 km/sec. Recall that in Ref's. 1 and 2 the goal was to send the spacecraft to its next target; therefore, the assist maneuver continued until its trajectory received the necessary boost.

In the current study, however, the goal is shifted to that of modifying the asteroid orbit to maximize its miss distance with the Earth, or more accurately, to minimize the likelihood of such a threat. A 180^{o} swing-by of the spacecraft around the asteroid would maximize the amount of momentum exchange between the two.

Figure 4 shows the three phases of the maneuver from the asteroid's perspective. At the start of the maneuver the asteroid and the spacecraft move along parallel paths and in the same direction with a relative velocity of 1-3 km/sec to satisfy the tether strength limitation. The magnitudes of the inbound and outbound velocities are assumed to be the same as in the case of a traditional gravity assist maneuver. From a heliocentric perspective, however, the spacecraft has gained in velocity at the expense of the asteroid. This scenario is depicted in Figure 5.

Note that from the asteroid's perspective, the incoming spacecraft seems to be moving to the left whereas after having swung around it, it is now seen to be moving to the right (Figure 4). From a solar perspective, on the other hand, both the incoming and the outgoing spacecraft velocities are seen to be pointing to the right except that the latter is greater in magnitude than the former. That gain in the spacecraft's speed is at the expense of the asteroid's, which ends up moving somewhat slower at the end of the maneuver (Figure 5). Conceptually, the relative velocities between the spacecraft and the asteroid could set be in the opposite directions, so that the NEO picks up a delta V at the expanse of the former. Which approach is more advantageous should be examined on a case-by-case basis.

2.3 An Example

Consider an asteroid in a collision orbit with the Earth with the following orbit characteristics.

$$R_{AP} = 1AU; \quad R_{AA} = 2AU \tag{1}$$

R_{AP} and R_{AA} are the asteroid radius of perigee and radius of apogee, respectively, and AU is the unit distance of the Earth from the Sun. Furthermore, assume that the timing is such that when the asteroid is at the perigee of its orbit, it finds itself collocated with the Earth, resulting in a collision between the two.

One strategy that could be used to deflect the asteroid from its collision course is depicted in Figure 3. A payload with a tether mechanism is sent from the Earth to come close to the asteroid when it is at the apogee of its orbit. The spacecraft flies on a path parallel to that of the asteroid but at a certain distance below it and at a slower speed. At that point the spacecraft deploys the tether and attaches itself to the asteroid. Note that conceptually, the spacecraft could be flying above the asteroid to achieve essentially the same effect.

The maximum achievable momentum exchange between the two would occur if the spacecraft were to swing around the asteroid by an arc of 180 degrees and then release. Assuming the asteroid to be much more massive than the spacecraft, the effect of that maneuver on the asteroid's trajectory would be much less pronounced than that of the spacecraft. Therefore, for the purpose of calculating the change in the trajectory of the spacecraft, the asteroid's position is considered fixed for the moment.

When the spacecraft detaches itself from the asteroid by severing the tether after having swung through 180 degrees, it is again flying in parallel to the asteroid path but at same tether length distance just above it and at a speed higher than that of the asteroid. The fate of the spacecraft is not important at that point, but the resulting minute change in the asteroid's orbit may make the difference between collision with the Earth near its perigee, or missing it altogether.

The duration of the spacecraft swinging around the asteroid is much shorter than its orbital period. If the relative velocity between the spacecraft and the asteroid is assumed to be 3 km/sec and the length of the tether 100 km, then swinging through an arc of 180 degrees lasts less than two minutes compared with an orbit period of 671 days. Therefore, the motion of both the spacecraft and the asteroid at the start and at the end of the swinging maneuver is assumed to be collinear, and the impact on the asteroid orbit is considered impulsive. This maneuver is assumed to be equivalent to an elastic collision between the two objects; therefore, the total linear momentum of the system is conserved. Equation 2 expresses that fact mathematically.

$$M_A V_{A1} + M_S V_{S1} = M_A V_{A2} + M_S V_{S2} \qquad (2)$$

M_A and M_S are the masses, and V_A and V_S are the velocities of the asteroid and the spacecraft, respectively. Subscript 1 denotes the start of the swing maneuver and subscript 2 denotes its end. The relative velocity between the spacecraft and the asteroid at the start of the swing maneuver is V_R, the magnitude of which is a design parameter depending on the strength of the tether material. Equation 3 relates that parameter to the known asteroid velocity at that point.

$$V_{A1} = V_{S1} + V_R \qquad (3)$$

The spacecraft exit relative velocity at the end of the swing maneuver is the same as the entry velocity at the start of that maneuver, except that it is in the forward direction. Equation 4 expresses that fact.

$$V_{S2} = V_{A2} + V_R \qquad (4)$$

Equations 3 and 4 can be used in Equation 2 to estimate the velocity of the asteroid at the end of the swing maneuver.

$$V_{A2} = V_{A1} - \frac{2M_S}{(M_A + M_S)} V_R \qquad (5)$$

Equation 5 establishes a link between the two design parameters M_S and V_R, and the attainable change in the asteroid velocity as a function of its mass. It indicates, as expected, that both design parameters should be maximized in order to yield the most effective tether assisted asteroid deflection maneuver.

Under the assumption that the asteroid is much more massive than the spacecraft, Equation 5 can be simplified.

$$\Delta V_A = \frac{2M_S}{M_A} V_R \qquad (6)$$

Equation 6 shows that the smaller the required delta V is, the more massive the asteroid that can be handled by the proposed approach.

2.4 Required Asteroid Delta V

The discussion above offers for consideration a tether-assisted method for asteroid deflection. Next, we need to establish the minimum required change in an asteroid's velocity to sufficiently reduce its conjunction threat with the Earth. Ref. 3 points out that this risk should be measured while accounting for uncertainties in the asteroid's trajectory, its size and the miss distance at the point of closest approach. This approach avoids an overly optimistic delta V that might result from simply basing a maneuver solution on increasing the miss distance without accounting for the uncertainty.

An attempt to reduce the probability of collision is shown in Ref. 3 to require a substantially larger delta V than by simply increasing the miss distance. Furthermore, it is shown that the amount of delta V required to reduce the probability to 10^{-6} is a factor of 2-3 larger as compared to the delta V for Earth miss distance of 1 Earth radius, and is on the order of 10 cm/sec or less for three of the asteroids studied. Moreover, the required delta V will be lower, sometimes to an order of 1 cm/sec or less, if the asteroid is further from the conjunction epoch. Use of these results in Equation 6 allows the establishment of the maximum size asteroid that can be deflected with an acceptable level of risk reduction in the presence of the uncertainties mentioned above.

2.5 Numerical Evaluation

The first step is to estimate the maximum size of an asteroid that can be handled by the tether-assisted asteroid deflection method. Equation 6 is rewritten as follows:

$$M_A = 2M_S \frac{V_R}{\Delta V_A} \quad (7)$$

To get a range of possible asteroid sizes the following assumptions are made:

1. V_R in the range of 1 to 3 [km/sec]
2. M_S in the range of 1000 to 10000 [kg]
3. ΔV_A in the range of 1 to 10 [cm/sec]
4. Asteroid density ρ_A in the range of 1 to 3 [g/cm^3]
5. The shape of the asteroid is a sphere of radius r_A [m]

Given the above data, Table 1 spans the range of possible asteroid sizes in terms of its mass M_A and its radius r_A.

Table 1. Range of asteroid sizes

Data	V_R [km/sec]	1	3
	M_S [kg]	1,000	10,000
	ΔV_A [cm/sec]	10	1
	ρ_A [g/cm^3]	3	1
Size	M_A [kg]	2e7	6e9
	r_A [m]	12	113

Table 1 clearly demonstrates that the suggested tether assisted asteroid deflection maneuver in this study can only handle relatively small asteroids. Larger objects would have to be fragmented and handled individually to possibly render them applicable to this approach.

For reference, significant damage, although unlikely to be globally catastrophic, can be inflicted by asteroids within the size range presented in Table 1. A well-known example is Meteor Crater in Arizona, USA, where an asteroid estimated to be 30 to 50 meters in diameter created nearly a mile size crater (Ref. 7).

The next step in the analysis is to relate the delta V that can be imparted to an asteroid to its miss distance from the Earth. Figure 6 depicts a hypothetical scenario where an elliptical asteroid orbit with Perigee radius of 1AU and Apogee radius of 2AU crosses with the Earth orbit at its Perigee. The approach taken here is deterministic; that is, uncertainties are not taken into account and the goal is to directly relate the miss distance to the impulse imparted on the asteroid. This expresses, to first order, a miss distance requirement in terms of asteroid delta V far ahead of the conjunction, and then further anchors it to the design parameters and the asteroid size.

It is assumed that, in general, the asteroid orbit and the Earth orbit are not coplanar. It is further assumed that the tether-assisted maneuver occurs at the Apogee and that its duration is much shorter relative to the orbital period so that the entire maneuver can be considered impulsive. Consequently, a small instantaneous change in the asteroid apogee velocity ΔV_A ends up slightly modifying the orbit period and its radius of Perigee without rotating its line of apsides. Table 2 shows the modified asteroid orbit Perigee radius and period stemming from a change in its Apogee velocity.

Table 2. Asteroid orbit VS. ΔV_A

Data	R_A [AU]	2	2
	R_P [AU]	1	1
	V_E [km/sec]	29.8	29.8
	ΔV_A [cm/sec]	10	1
Mod. orbit	ΔR_P [km]	2,610	261
	ΔT [sec]	506	51
	Miss dist. [km]	15,063	1,506

Table 2 establishes that for the scenario depicted in Figure 4 ΔV_A cannot go too low or it will not support a miss distance with sufficient margin to account for the inherent uncertainties. For example, with ΔV_A = 10 cm/sec the miss distance comes out to be roughly 2.5 Earth radii which may or may not suffice to account for these uncertainties. Ref. 3 does indeed indicate that with ample warning time the range of required ΔV_A may very well be inside the extent shown in Table 2. Note that this result does not depend on the size of the asteroid and therefore is independent of the method used for its deflection, using a tether-assisted maneuver or any other approach.

The DT shown in Table 1 will change depending upon whether the impulsive maneuver is performed one-half of an orbital period before collision, or one and one-half of a period ahead, two and one-half, etc. For each full orbital period before final approach, the DT will increase by twice the value shown and improve the effectiveness of the technique.

3. SUMMARY AND CONCLUSION

This study suggests a tether-assisted asteroid deflection method to mitigate its collision threat with the Earth. The minimum deflection required to reduce this threat to an acceptable level in the presence of uncertainties is estimated based on results obtained in previous studies and is used to determine the largest asteroid that can be effectively deflected by the tether-assisted method. The results of a hypothetical case study show that due to tether material strength limitations this method can handle small-to-medium size asteroids (up to approximately 0.2 km in diameter), or possibly fragments of larger size objects.

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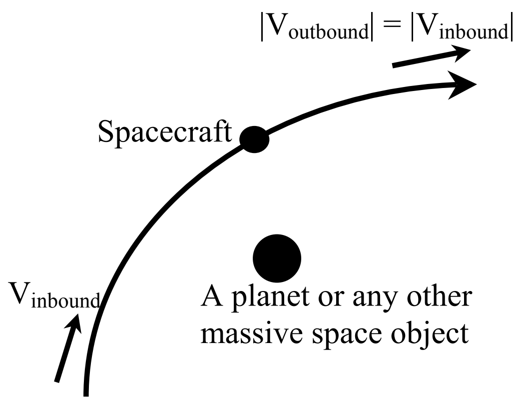

Figure 1. Gravity assist from a planet perspective

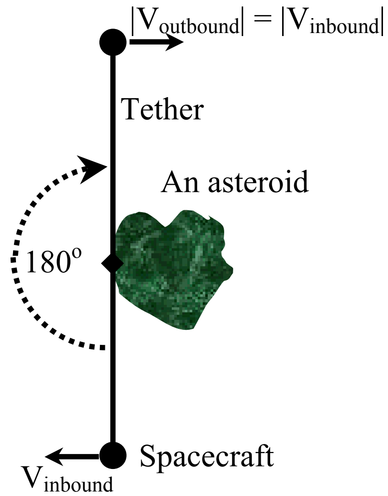

Figure 4. Maximized tether-assisted asteroid deflection from an asteroid perspective

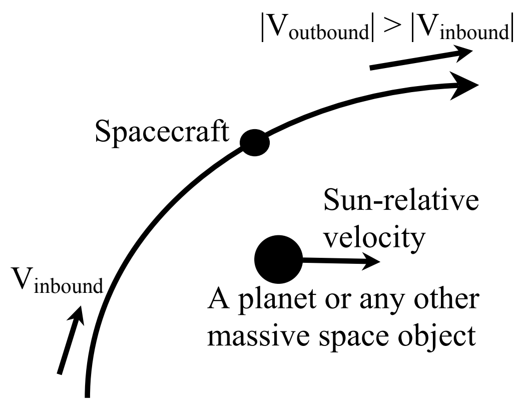

Figure 2. Gravity assist from a solar perspective

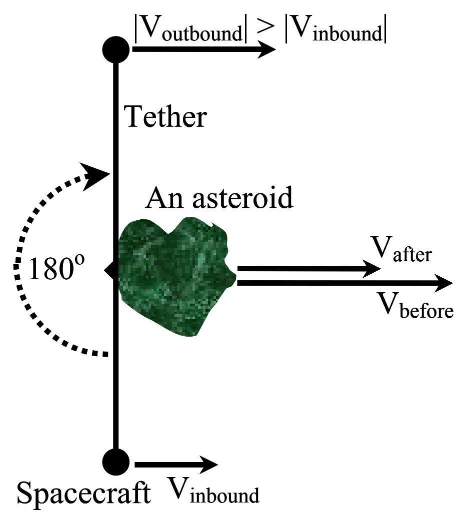

Figure 5. Maximized tether-assisted asteroid deflection from a solar perspective

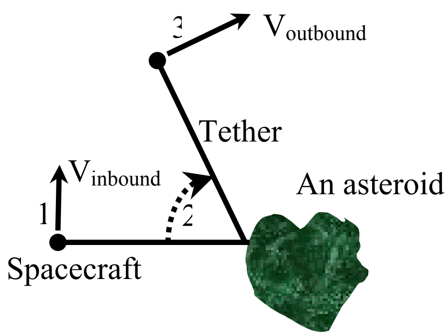

Figure 3. Three phases of tether-assisted spacecraft deflection

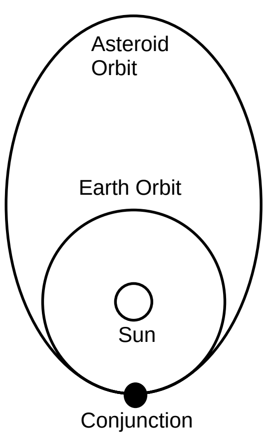

Figure 6. Hypothetical asteroid Earth conjunction