USING SMALL ASTEROIDS TO DEFLECT LARGER DANGEROUS ASTEROIDS

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Abstract: *The main idea of the proposed approach consists of targeting a very small asteroid to impact a larger dangerous one. The minimum size of this small asteroid is determined by the ability to detect it and to determine its orbit. The small object may have a diameter of about 10 - 15 meters. Asteroids are selected from the near-Earth class with the fly-by distance from Earth of the order of hundreds of thousands of kilometers. According to current estimates, the number of near Earth asteroids with such sizes is high enough. So there is a possibility to find the required small asteroid. Further, the possibility is evaluated of changing the small asteroid's orbit so that by application of a very limited delta-V impulse to the asteroid, the latter is transferred to a gravity assist maneuver (Earth swingby) that puts it on a collision course with a dangerous asteroid. It is obvious that in order to apply the required ΔV pulse it is necessary to install on the small asteroid an appropriate propulsion system with required propellant mass. A control system similar to that used on a spacecraft is also necessary. Of course, any real test of this (or of any other deflection) strategy should first be performed on a benign asteroid whose orbital parameters give the asteroid no significant chance of a natural impact with the Earth during at least the next million years, even in the case of small changes in its orbit.*

Keywords: *Near-Earth asteroids, planetary protection*

1. Introduction

A concept based on the use of small asteroids for deflecting hazardous near-Earth objects from the trajectory of their probable collision with the Earth was proposed at a Symposium held on Malta in October 2009. The Symposium was devoted to the solution of problems connected with the prediction and possibilities of prevention of such collisions. After the Symposium the paper describing this concept was published in Cosmic Research Journal #5 in 2010 [1]. The essence of the concept is applying to a small enough asteroid, with the size about 10-15 meters, a comparatively small velocity pulse (about 10-15 m/s) in order to transfer it on the trajectory to the Earth where with the use of a gravity assist maneuver this asteroid will be put on trajectory of collision with a hazardous near Earth object, for example Apophis [2]. The last one (with estimated size equal 270m) is considered to be an object with one of the higher (but small) probabilities to hit the Earth. The event is expected to happen in 2036. In [1] it was shown that the proposed method is by two orders more effective as compared with the method of direct targeting of the spacecraft to the hazardous asteroid with consequent collision of the bodies. This estimation of effectiveness is calculated in terms of the ratio of the hazardous asteroid velocity change and the mass of the projectile hitting the dangerous object.

But in the mentioned paper the question was not answered: is it possible in the list of reachable asteroids to find the one which is possible using gravity assist maneuver to put it on the trajectory of collision with Apophis. It was supposed that because there are enough of asteroids having the acceptable size it is possible to choose the asteroid acceptable for our task solving. But one needs to understand that to detect so small asteroids and to determine their orbital parameters is difficult enough problem to be solved. But we observe now very fast process in solving this problem: in USA and Europe very ambitious programs in this area are in the phase of realization. During last decade of these programs running it was discovered more near Earth asteroids than during whole period of preceding observations. European Space Agency plans to launch in 2013 astrometry spacecraft GAIA into vicinity of Solar-terrestrial libration point L2 with the goal to discover and catalogue several thousands new near Earth objects [3].
So it is very important to answer the question does such asteroids exist in the contemporary catalogue which is possible to transfer on the trajectory of collision with Apophis before it close approach to the Earth in April 2036 supposing that we can use only available now launch vehicles and technologies of spacecraft motion control. We mean such scenario realization when spacecraft with propellant of enough mass is sent to chosen small asteroid-projectile, then lands on this asteroid and after its fixation on the surface spacecraft engine unit gives to asteroid velocity pulse, which transfer the asteroid to the trajectory of its collision with Apophis using gravity assist maneuver near Earth.
Given below results of studies are goaled to give some preliminary confirmation that the described method of Apophis deflection from its trajectory of possible collision with the Earth is doable.
Taking into account the last successes in discovering new near Earth asteroids one may state that the situation with the chances to find required asteroid-projectile to implement the proposed method of Earth defense will be improving.

2. Gravity assist maneuver as a tool to target asteroid-projectile to hazardous near Earth object.

Asteroids which are considered as small have mass about 1500 tons so to control their motion in classical use of this word i.e. by applying pulse of rocket engine in order to change their velocity by several kilometers per second is hardly possible. But if gravity assist maneuver is used as the tool to amplify orbital parameters change when small pulse of velocity (delta-V) allows changing the pericenter height of controlled body (asteroid-projectile) when it fly-bys the planet (Earth) by value sufficient to turn the relative velocity vector by dozens of degrees, then we receive extremely efficient tool of orbital control for the sky body having so huge mass as compared with the usual spacecraft.
By choose of relative velocity position at infinity with respect to the fly-by planet (with fixed pericenter radius) we receive any required plane of relative orbit with appropriate direction of relative velocity vector after fly-by. This is illustrated by Fig.1 where cylinder of possible vectors of relative velocities (at infinity) at arrival and resulted after fly-by cone of velocity vectors of departure are imaged.

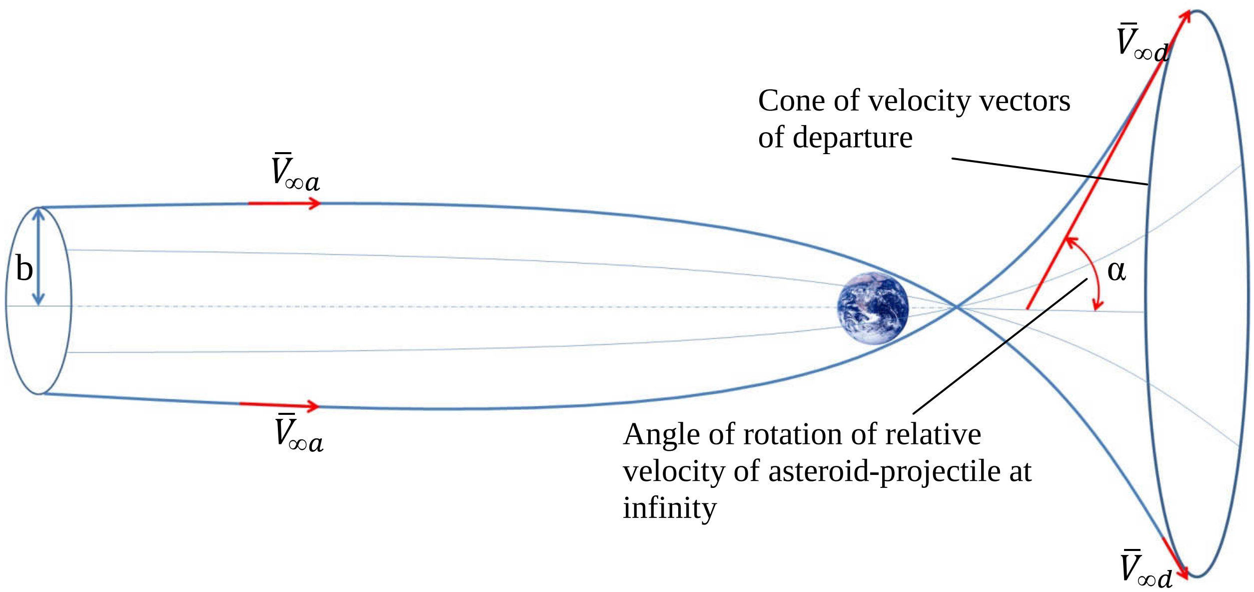


Figure 1. Cylinder of possible vectors of relative velocities at arrival and resulted after fly-by cone of velocity vectors of departure are imaged

Geometry of gravity assist maneuver is illustrated by Figure 2.

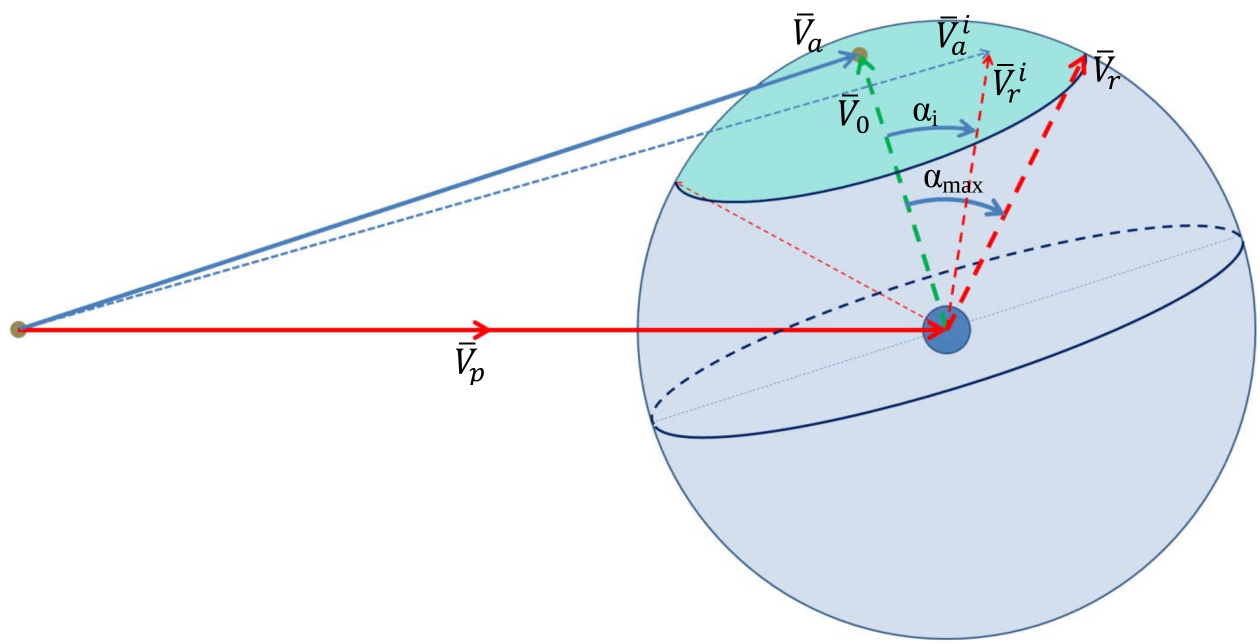


Figure 2. Geometry of gravity assist maneuver in coordinate system connected with Sun

In Fig. 2 $\mathbf{V}_p$ 2 $\mathbf{V}_p$ is velocity vector of the planet with a fly-by during gravity assist maneuver (the Earth in our case) given in coordinate system connected with the Sun, $\mathbf{V}_a$ is arrival velocity vector in the same reference (coordinate) system and $\mathbf{V} = V_0$ is relative with respect to planet velocity vector of asteroid at the moment of its arrival to the Earth referred to infinity (if gravity of the Earth would be equal zero this vector would be equal the one we consider). After planet fly-by this vector is turned by the α angle which can be calculated using the following formulae [4]:

$$\sin\frac{\alpha}{2} = \frac{1}{1 + rV^2 / \mu} \quad (1)$$

where

r – pericenter radius, V – relative velocity of asteroid-projectile at infinity, μ – gravitational constant of the planet.

As one can see from the formulae with decreasing the pericenter radius the α angle of relative velocity vector turn due to fly-by increases reaching 180 degrees if pericenter radius reaches zero. Thus if there are no limits constraining pericenter radius lowest value, then vector of relative asteroid velocity will change its initial direction on opposite one. The whole family of the departure velocity vectors forms in this case the sphere with V radius. Asteroid velocity vector in reference system connected with the Sun is the sum of planet velocity vector and relative vector of asteroid with respect to planet (Earth) **V.** Thus asteroid velocity vector in coordinate system connected with Sun may be any vector with initial point in the same point as planet vector and any final point on the mentioned sphere. But this is valid in case if radius of planet is zero. If one will take into account the radius of the planet, then only part of the sphere may present the possible surface for the reachable asteroid vectors after gravity assist maneuver. This part is constrained by the cone with the axis coinciding with arriving vector of asteroid relative velocity $\mathbf{V}_0$. Semiangle of this cone is equal α_{max}, calculated by use of given above formula where pericenter radius is equal minimum allowed one. With increasing allowed radius of pericenter of fly-by trajectory from zero to infinity the area of reachable departure velocity vectors shrinks from the whole sphere to the one point which is the end point of arriving velocity vector. Accordingly decreases the area of achievable velocity vectors of planet fly-by body (asteroid-projectile).

3. Lambert problem as the principal constituent of trajectories design with the use of gravity assist maneuvers.

Scenario of mission for deflecting hazardous near Earth object from trajectory with Earth collision consists from the following phases:

- start of spacecraft from the Earth to trajectory of transfer to asteroid intended to be use as controlled spacecraft of large size for its targeting to hazardous object;
- landing and fixing the spacecraft on asteroid surface by use of its engine unit to decrease the relative velocity of spacecraft with respect to asteroid to zero level and fulfilling the procedure of docking with asteroid;
- flight of controllable asteroid in passive mode until reaching the moment of its start to trajectory to the Earth;
- applying to the asteroid-projectile the velocity pulse in order to transfer it to the trajectory of the Earth fly-by in regime of gravity assist maneuver by use engine unit of the spacecraft landed and fixed on asteroid surface;
- flight along trajectory to the Earth with executing necessary correction maneuvers to reach required parameters of fly-by trajectory with necessary accuracy;

- executing gravity assist maneuver transferring asteroid-projectile to the trajectory of collision with hazardous near Earth object, in our case Apophis is chosen;
- flight along Apophis hitting trajectory with executing necessary correction maneuvers with final collision with this object which is to deflect the asteroid-target from initially dangerous orbit with possible hit of the Earth.

It is obvious that during the whole mission all available tracking instruments and facilities are to be involved in the solving the navigation tasks with maximum achievable accuracy. The importance of this part of the mission operations is much higher than in case of the usual spacecraft motion control because the values of correction maneuver to great extent determined by accuracy of orbital parameters determination as in standard space missions, but in terms of propellant consumption it means that the mass of propellant for correction may exceed its mass for nominal maneuvers.

The task of the mission design at large consists in the choose of all available free parameters in such a way which with maximum payload reachable by contemporary available launch vehicles allows to receive maximum deviation of velocity vector of target asteroid (Apophis) after hitting it by controllable asteroid-projectile.

This task is multi parametrical so it is to be solved by several phases.

First step is to choose optimal transfer of the spacecraft from low near Earth satellite orbit to the asteroid chosen as candidate asteroid-projectile. For this the Lambert problem is to be solved with goal to minimize required delta-V to execute this mission. As it is well known the Lambert problem [5, 6] consists in choice of such initial orbital parameters which during given time allows to transfer along Keplerian orbit the zero mass object from one given point to another. Our goal is to choose from possible dates of launch and arrival the optimal one in terms of total delta-V. During this phase acceptable candidate asteroids are to be chosen as the dates of their departure and arrival. It should be mentioned that on this phase of calculations of asteroid motion the gravity field is supposed to be central with the center in Sun. The coordinates of Earth and coordinates of asteroids are taken from appropriate catalogue contained in SPICE system [7].

Next step is to find transfer orbit from the Earth to Apophis with date and time of start the same as the ones of arrival to the Earth of asteroid-projectile. Besides the module of relative with respect to Earth velocity vector is chosen the same as the module of arriving velocity vector. As a result of this second (modified) Lambert problem solution we receive the departure relative velocity vector and transfer trajectory for asteroid-projectile at large. Practically it was confirmed that standard algorithm of Lambert problem solution may be used for both part of the trajectory: for approaching to the Earth part and for the part from the Earth to Apophis. For this it is enough to minimize the sum of the delta-V for transfer asteroid-projectile from its initial orbit to the orbit reaching Earth and difference of modules of arriving and departing relative velocities in Earth point of the arriving and departing trajectories. As calculations showed the result of such minimization is zero difference of these modules what exactly correspond to the demand of mutual match of these trajectories, so we receive one uninterrupted trajectory of asteroid-projectile from the first correction velocity pulse to the collision with asteroid Apophis.

With the use of described method and the catalogue of the solar system bodies orbital parameters contained in SPICE calculations have been fulfilled aimed to choose the candidate asteroids for targeting them to Apophis with the use of gravity assist maneuver near Earth. The criterion of selection of these asteroids was the required delta-V to transfer asteroid from its original orbit to the trajectory of collision with Apophis after gravity assist maneuver near Earth. The sizes of the asteroids were also taken into account as the necessary delta-V to deliver the spacecraft onto surface of asteroid-projectile. Five the best asteroids, satisfying described demands were chosen by our studies and their key characteristics are given in Table 1.

Table 1. Results of candidate asteroids choose and orbits design

Asteroid	2006 XV4	2006 SU49	1997 XF11	2011 UK10	1994 GV
Delta-V value, m/s	2.38	7.89	10.05	15.94	17.72
Perigee radius, km	16473.19	15873.40	42851.84	31912.94	7427.54
Velocity in perigee with respect to Earth, km/s	9.61	5.03	14.08	8.98	13.37
Angle of the relative to the Earth velocity turn, deg.	23.98	59.78	5.14	21.14	50.85
Date of maneuver execution	2029/03/17	2027/06/11	2027/04/27	2025/09/13	2028/09/12
Date of perigee reaching	2031/12/11	2029/01/23	2028/10/26	2026/10/10	2031/04/13
Date of collision of asteroid-projectile with Apophis	2034/04/08	2029/10/06	2030/08/06	2027/08/06	2031/12/24
Impact velocity with Apophis, km/s	15.3	4.9	11.0	2.3	14.1
Magnitude	24.87	19.54	16.9	24.91	27.46
Size of asteroid-projectile	25 ≈ 60 m	330 ≈ 750 m	1 ≈ 2 km	25 ≈ 60 m	8 ≈ 19 m
V^2 at infinity after s/c launch from near Earth orbit, km^2/s^2	3.7	0.36	6.447	47.182 (1.488*)	30.128 (2.427*)
Delta-V of braking for landing S/C on an asteroid, km/s	9.6	4.67	7.89	0.543 (5.571*)	0.543 (6.860*)

*for departure delta-V optimization

4. Optimization of spacecraft transfer trajectory with start from low near Earth orbit and landing on asteroid-projectile surface.

Natural criterion of optimization for choice of the trajectory of spacecraft delivery to the asteroid-projectile surface is the maximum mass of the spacecraft after landing. For our studies it is enough to use instead of this criterion very close to it one: the total ΔV_t required to start from low near Earth orbit and to execute the maneuver to land spacecraft on asteroid surface. For calculating first constituent (ΔV_s) of the delta-V we assume that the spacecraft starts from the circular near Earth orbit with 200 km height. The second constituent we estimate as the relative velocity of spacecraft with respect to asteroid when reaching it.

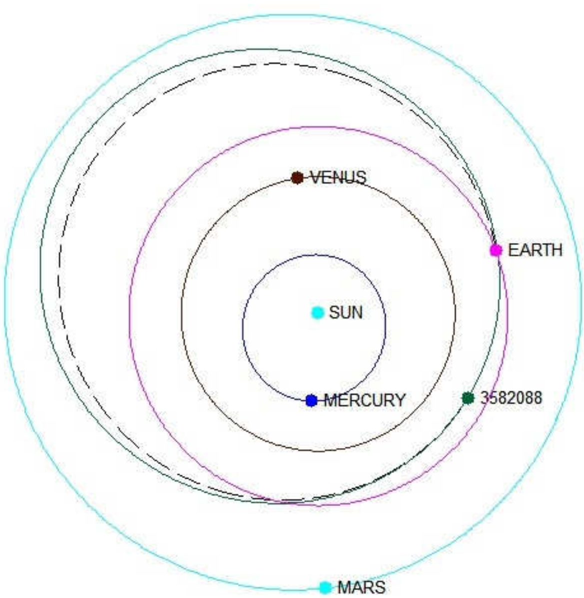
In order to minimize available software [8, 9, 10] modification, the other approach for solving the problem was used: we minimized the function $F = W_1*C_3+W_2*\Delta V_a$, where C_3 is the square of relative asymptotic velocity with respect to Earth of departing to asteroid trajectory, ΔV_a is relative velocity of the spacecraft with respect to asteroid at the moment of arrival to asteroid, W_1, W_2 are the weight factors for the used procedure of F function minimization. Several values of W_2 were tested keeping $W_1 =1$ for all cases. It is obvious that with increasing the W_2 value the influence of arriving velocity on the solution is raising. In the presented table of received solutions which includes the data of start, data of arrival, square of asymptotic velocity at departure, delta-V required for departure, relative velocity (equal delta-V, necessary for landing) at arrival to asteroid. Table 2 presents key orbital parameters for transfer mission of the spacecraft to be landed on 2011 UK10 asteroid calculated by the method of optimization for different weight factors W_1, W_2. The search of optimal trajectory was done for the spacecraft departure inside interval of dates beginning with departure not earlier than 2020-01-01 and ending by arrival not later than 2025-08-15.

Table 2. Key orbital parameters for transfer mission of the spacecraft to asteroid-projectile

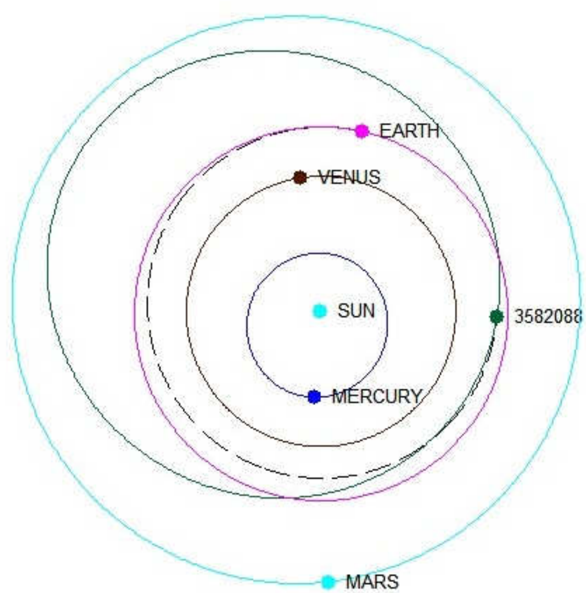
W_2	Optimal time of departure from Earth	Optimal time of arrival to 2011 UK10	C_3, km^2/s^2	ΔV_a, km/s	Duration of transfer days	ΔV_s, km/s	ΔV_t, km/s
1	2021/12/10	2022/08/25	1.4879	5.5709	257.9243	3.302	8.873
2	2021/12/08	2022/08/21	2.023	5.2113	255.1062	3.326	8.537
6	2022/08/20	2023/08/02	6.99	3.4856	346.7858	3.537	7.033
10	2022/08/24	2023/08/06	7.443	3.4306	346.9625	3.567	6.970
12	2022/08/28	2023/08/08	8.1783	3.3644	345.7742	3.599	6.963
13.5	2022/09/18	2023/08/17	16.7777	2.6997	332.2901	3.971	6.670
14	2022/09/21	2023/08/18	18.5637	2.5698	330.8983	4.047	6.617
14	2022/10/13	2023/12/10	46.9763	0.55489	422.8127	5.193	5.747
15	2022/10/13	2023/12/09	47.0002	0.55327	422.5287	5.194	5.747
20	2022/10/13	2023/12/09	47.1824	0.54275	422.5234	5.201	5.744
W_2	**Optimal time of departure from Earth**	**Optimal time of arrival to 1994 GV**	**C_3, km^2/s^2**	**ΔV_a, km/s**	**Duration of transfer days**	**ΔV_s, km/s**	**ΔV_t, km/s**
1	2027/04/17	2028/06/07	1.9758	7.3286	416.8176	3.324	10.653
2	2027/03/17	2028/04/22	2.4656	6.86	401.9748	3.346	10.206
6	2027/05/04	2028/05/17	8.3888	5.2059	378.4848	3.609	8.815
7	2027/05/04	2028/05/16	8.39	5.2057	378.4707	3.609	8.815
7.5	2027/05/04	2028/05/16	8.3904	5.2057	378.4551	3.609	8.815
7.8	2027/05/04	2028/05/16	8.3904	5.2057	378.4952	3.609	8.815
7.9	2026/03/17	2028/03/21	30.1269	2.4273	734.737	4.526	6.954
8	2026/03/17	2028/03/21	30.127	2.4273	734.7517	4.526	6.954
10	2026/03/17	2028/03/21	30.1283	2.4271	734.6328	4.526	6.954
20	2025/12/03	2028/01/02	50.6314	0.5913	1056.6681	5.333	5.924

As one can see from the table the last string presents the optimal trajectory with total delta-V equal 5.744 km/s. The impulse required to land spacecraft on asteroid surface is equal only 0.5427 m/s.
This trajectory is imaged on Fig.3a in ecliptic projection together with orbits of inner planets and orbit of 2011 UK asteroid. As one can see from Table 2 the first string variant corresponds practically the case when only departure delta-V is optimized with ignored arrival value of delta-V. And it can be easy to see that in this case the departure delta-V is practically minimal which allows leaving Earth gravity giving to the spacecraft the velocity almost equal parabolic one. This trajectory is imaged on Fig.3b and repeated on Fig.5 together with other trajectories illustrating the mission at large.
Rough estimations show that standard Proton launch vehicle with Breeze upper stage using optimal transfer trajectory can deliver to the surface of 2011 UK10 asteroid (after consumption total delta-V equal 5.744 km/s) the payload with 2300 kg mass.
Analogous calculations have been fulfilled for 1994 GV asteroid. The studies were done for the spacecraft start options beginning not earlier than 2025-01-01 and arrival to asteroid not later than 2012. For this asteroid-projectile option and for mentioned above allowed dates of transfer mission to asteroid, the minimum total delta-V is 5.904 km/s (departure delta-V is 5.3131 km/s, arrival delta-V is 0.5913 km/s), departure date is 2025-3-12, arrival date is 2028-2-1, duration of transfer from Earth to 1994 GV is 1056.67 days. The trajectory of transfer is presented by Fig.3c. The same launch vehicle and upper stage can deliver onto this asteroid surface payload with 2270 kg mass.
Fig.3d illustrates the case when for 1994 GV reaching the departure delta-V is minimized.

a)



b)



c) d)

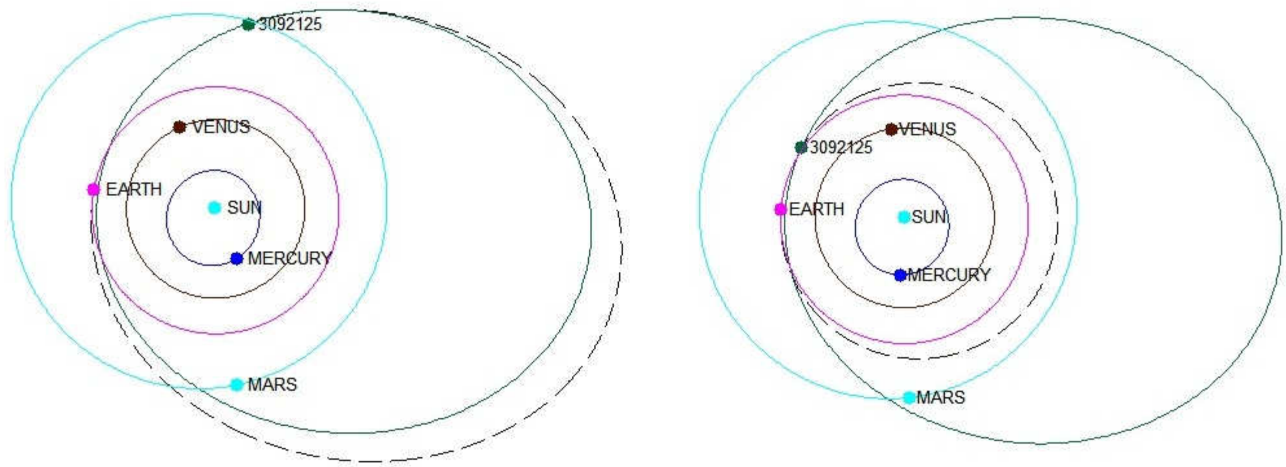


Figure 3. a - Optimal transfer trajectory to 2011 UK asteroid with minimum total delta-V, b - Transfer trajectory to 2011 UK asteroid with minimum departure delta-V, c - Optimal transfer trajectory to 1999 GV asteroid with minimum total delta-V, d - Transfer trajectory to 1999 GV asteroid with minimum departure delta-V.

5. Comparison of Lambert problem solution and results of numerical integration of differential equations of Solar System bodies motion

For 2011 UK10 (#3582088 in SPICE system) the studies have been fulfilled in order to compare the results of numerical integration of differential equations system, which describes the asteroid motion taking into account solar system gravity field generated by Sun and planets, with the ones calculated by Lambert problem solution.

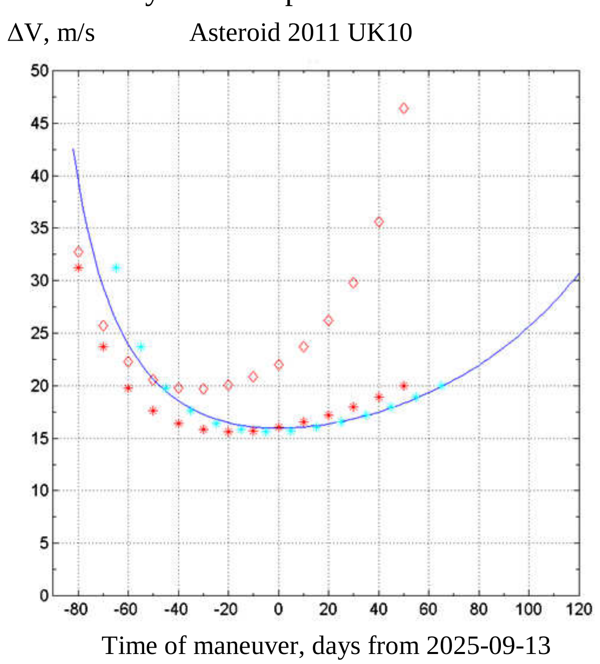


Figure 4. Delta-V value to transfer asteroid to gravity assist maneuver trajectory for Apophis impact.

On the Fig. 4 zero value of time corresponds to the optimal time point of maneuver sending asteroid projectile to the Earth, calculated in framework of Lambert problem solution, which gives minimum delta-V of such maneuver. Accordingly the solid blue line is function of delta-V of time for this approach. The red diamonds correspond to the case when the orbit was calculated by numerical integration for the same times of maneuver and arrival to the perigee of the fly-by orbit (in Lambert case it is arrival in the Earth center). The red stars correspond to the case where the maneuver dates are as shown but arrival to the Earth perigee dates are optimized for the case of numerical integration. Blue stars curve is received by shift of red stars curve to the right side by 15 days in order to show that by such shift we receive the initial solid blue line. In other words this fact confirm that solution of Lambert problem gives the same results (in terms of delta-V for maneuver) as more precise solution with use of numerical integration. The only difference is the mentioned shift by 15 days in time in the whole window of possible initial maneuver.

Table 3. trajectory parameters corresponding minimum delta-V

	Use Lambert problem	Numerically integrated trajectory	Numerically integrated trajectory with optimization by choose T_1 and T_3
Date and time of maneuver execution - T_1	2025-09-13 10:37	2025-09-13 10:37	2025-08-24 10:37
Delta-V value, m/s	15.9	22.0	15.6
Date and time of perigee reaching - T_2	2026-10-10 13:07	2026-10-10 07:06	2026-10-10 13:02
Perigee radius, km	31912.7	31578.7	31683.7
Velocity in perigee with respect to Earth, km/s	8.98	8.977	8.995
Date and time of collision of asteroid-projectile with Apophis - T_3	2027-08-06 07:13	2027-08-06 08:54	2027-08-07 10:27
Apophis impact velocity, km/s	2.25	2.26	2.28

Thus the data given in the above list of key trajectory parameters confirm that for our studies related to the possibilities of the use small asteroids to deflect the dangerous ones from their original trajectory hazardous by some nonzero probability of collision with the Earth, can be done using Lambert problem solution methods.

6. Two mission's examples intended for Apophis deflection using small asteroids as projectiles.

Full mission to hit Apophis by 2011 UK asteroid is illustrated by Fig.4 where ecliptic projection of the trajectories of bodies, involved in the mission to deflect Apophis from its original trajectory is presented. The key points and dates of the mission are marked on the Fig.4, beginning from the start of the spacecraft from the Earth to be landed on 2011 UK asteroid and ending Apophis impact by this asteroid. These points are the following:

I. 2021-12-10: Launch of spacecraft from the Earth
II. 2022-08-25: Spacecraft arrive to 2011 UK10 and landing
III. 2025-09-13: Applying the velocity impulse to the 2011 UK10 asteroid
IV. 2026-10-10: Perigee passing by asteroid during gravity assist maneuver
V. 2027-08-07: 2011 UK10 asteroid impact to Apophis

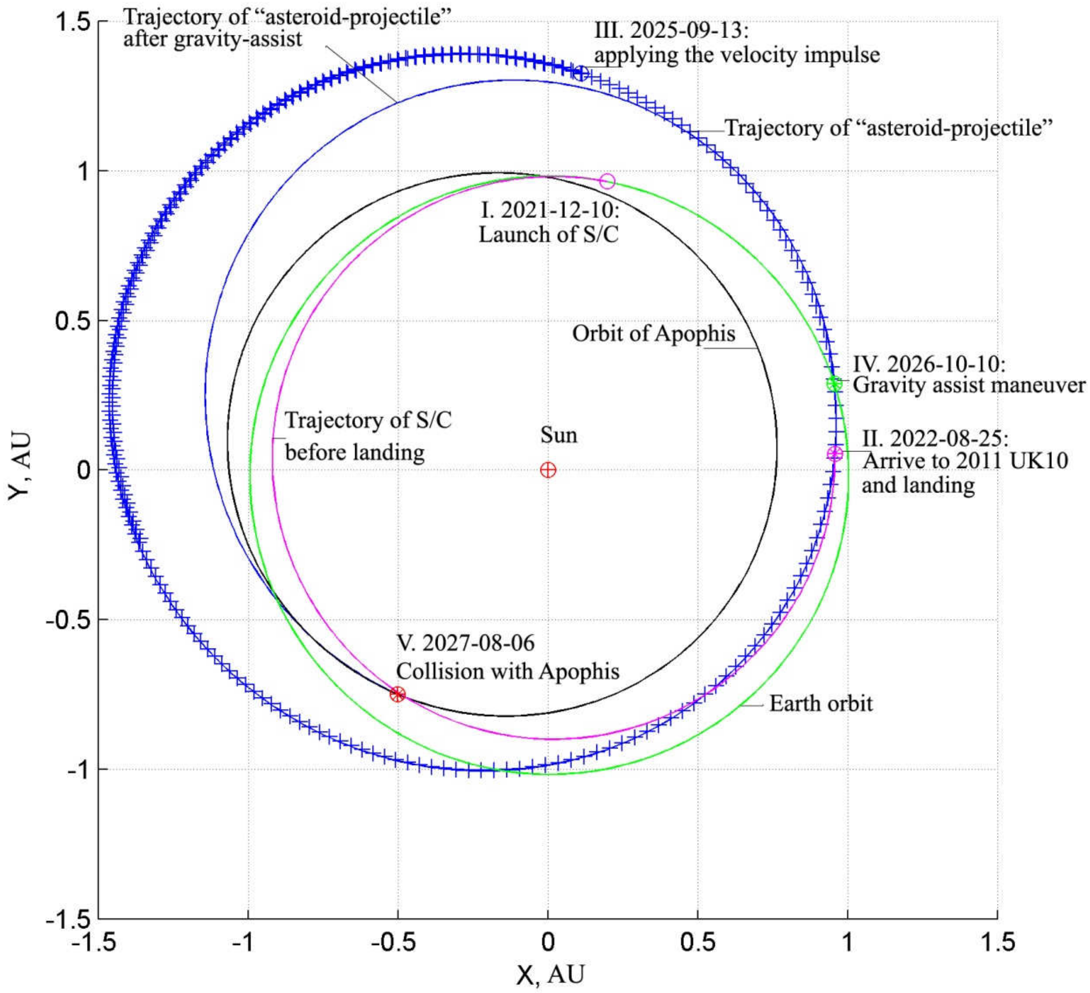


Figure 5. Trajectories of Earth, Apophis, asteroid-projectile 2011 UK and spacecraft transfer trajectory from Earth to 2011 UK

The Apophis asteroid trajectory is presented by black, Earth by green, 2011 UK asteroid by blue crosses before maneuver pulse applying and solid blue after maneuver, spacecraft to be landed on 2011 UK by crimson.

Fig. 6-8 given below show as a function of time the distance from the center of Earth to 2011 UK10 and Earth – asteroid - Sun angle during the mission beginning from the asteroid start to the Earth for gravity assist maneuver and ending by collision with Apophis.

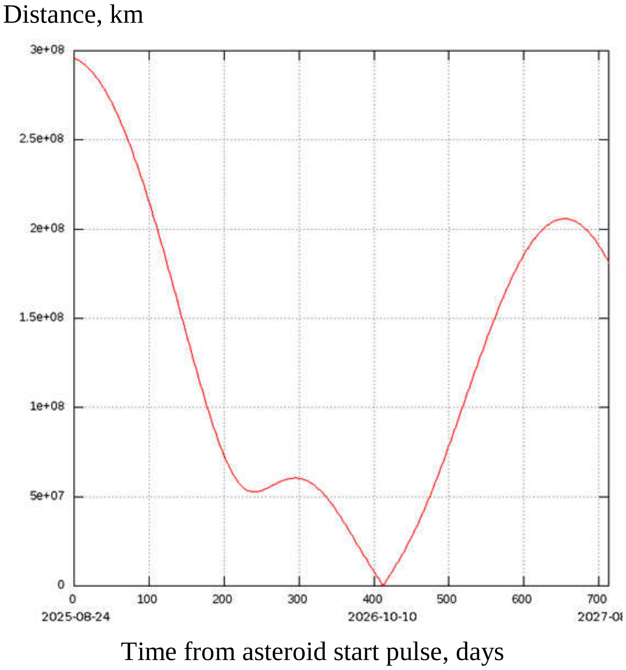


Figure 6. Distance from the 2011 UK10 asteroid to the Earth center for the time interval from asteroid start pulse to the impact with Apophis.

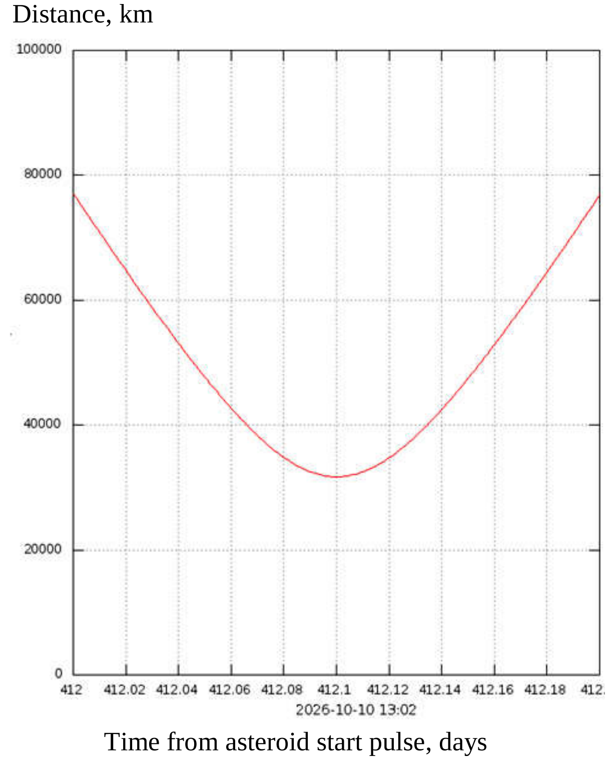


Figure 7. Distance from the 2011 UK10 asteroid to the Earth center in the region of perigee during gravity assist maneuver.

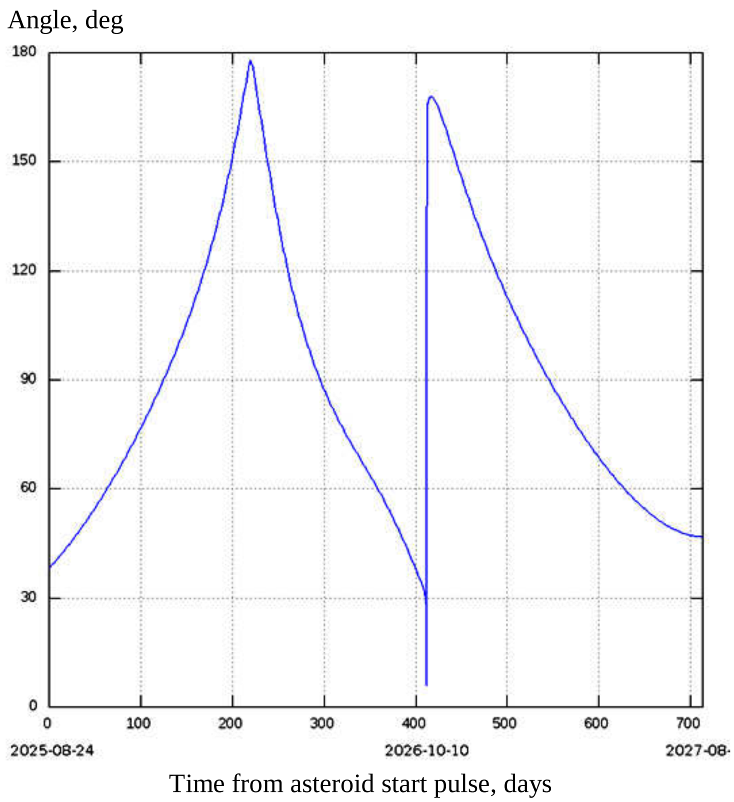


Figure 8. Earth-asteroid-Sun angle during the mission after 2011 UK10 start to the Earth vicinity for gravity assist maneuver.

The second example of trajectories of the natural and artificial bodies participating in Apophis deflection mission is presented by Fig.9 for 1994 GV as asteroid-projectile.
The key points of the mission are the following:

I. 2027-030-17 start of the spacecraft from the Earth to asteroid-projectile
II. 2028-04-22 landing spacecraft onto asteroid 1994 GV surface
III. 2028-09-12 start maneuver of asteroid targeting it to the vicinity of the Earth for gravity assist maneuver
IV. 2031-04-13 perigee passing during gravity assist maneuver
V. 2031-12-24 collision of 1994 GV with Apophis.

The colors are the same as the ones used in Fig. 4. The only difference is that asteroid-projectile is 1994 GV instead of given in Fig.4 asteroid 2011 UK10

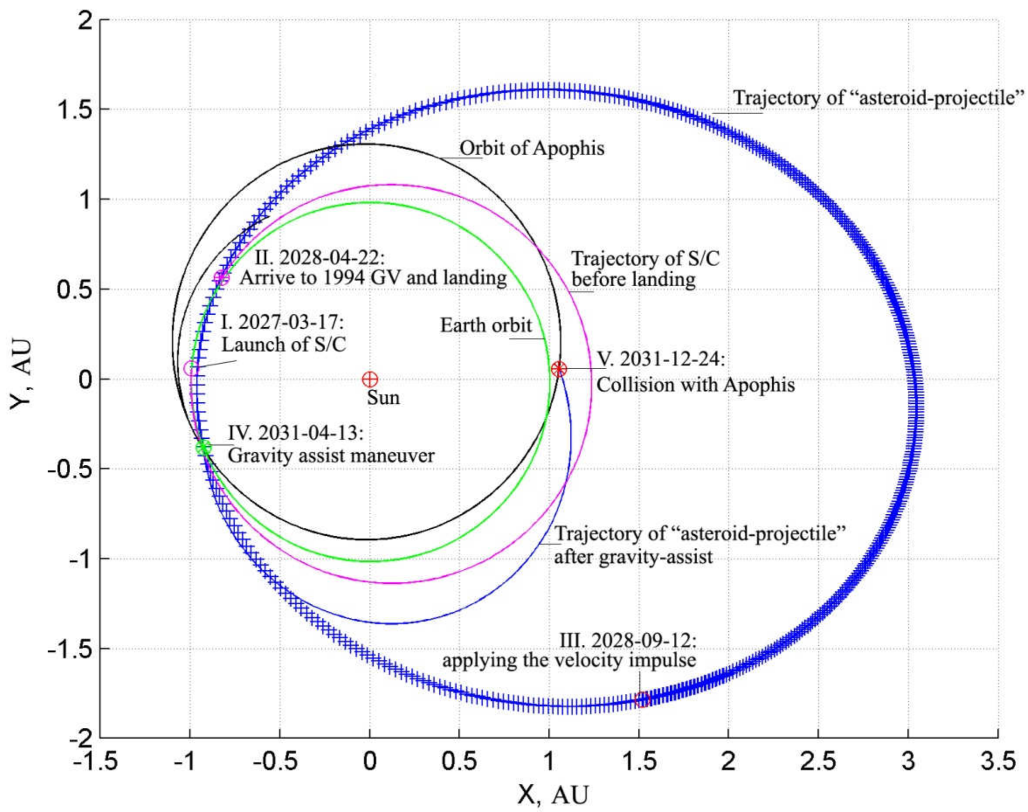


Figure 9. Trajectories of the Earth, Apophis, asteroid-projectile 1994 GV and spacecraft transfer trajectory from the Earth to 1994 GV.

7. Analysis of results for ideal cases

The figures given in Table 1 and presented by key parameters of trajectories found by our studies are received for ideal cases, i.e. for the ones when trajectory is nominal when deviation real trajectory from calculated trajectory is zero. But for the considered missions the part of propellant to be consumed during mission for correction maneuvers may be comparable with those for nominal ones.

If one consider the mission effectiveness only in terms of nominal trajectory then from presented by Table 1. 5 cases, the most promising case is the mission with the use asteroid 1994 GV. The main argument for such chose is the estimated mass of asteroids presented in the Table 1. The closest in size asteroids are 2011 UK10 and 2006 XV4 are supposed (if their densities are the same) to have by factor 30 higher. Assuming that mass of 1994 GV asteroid is 1350 tons (radius 6 meters, density 1.5 t/m^3) in order to change its velocity by required 17.72 m/s using engine unit with specific impulse 3300 m/s one needs to consume 7.23 tons of propellant. It means that 4 Proton launch vehicles are to be used in order to deliver the required amount of propellant to the asteroid surface. It is doable but looks as not simple enough. In addition some consumption of propellant is necessary in order to fulfill trajectory correction maneuvers. If one refers to the Deep Impact mission to estimate of required delta-V for these maneuvers than the expected figures may reach a few dozen meters per second [11]. It means that for implementation of the proposed technology of dangerous asteroid deflection we need to search more asteroids satisfying our demands in terms of their size and required delta-V to target them to dangerous object like Apophis. For example if 2006 XV4 would have the same size as 1994 GV and similar required delta-V to reach it then the required mass of propellant to target it to Apophis would be only 0.98 tons if one would not take into account the propellant needed for correction maneuvers.

But one needs to mention that even with the use of 10 Proton launches intended to realize the proposed technology for the case chosen is by factor of 27 is more effective than the direct targeting of spacecraft to Apophis.

As to correction maneuvers there is the possibilities to decrease them significantly by the use of responder to be delivered to the surface of the Apophis before execution of mission for its deflection.

8. Conclusions

The described method of dangerous asteroids deflection from the trajectory of collision with the Earth as it was shown on the example of Apophis may be considered as doable. It was found that very small delta-V (2.38 m/s) may be required to transfer small asteroid to the trajectory, what includes gravity assist maneuver near Earth, followed by collision of this asteroid with the hazardous object like Apophis. Proposed method allows to change velocity of dangerous object by the value unachievable by any other contemporary technologies. For practical implementation of the proposed approach some further progress in broadening the catalogue of candidate asteroid-projectile in needed especially as it is related to small asteroids. Also additional studies are required for reaching lower demands for correction maneuver delta-Vs.

9. References

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