

1           **ELECTRIC SAIL FOR NEAR-EARTH ASTEROID SAMPLE**  
2                           **RETURN MISSION: CASE 1998 KY26**

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4           **Abstract**

5           The electric solar wind sail (E-sail) is an innovative propellantless concept for interplan-  
6           etary space propulsion that uses the natural solar wind as a thrust source with the help of  
7           long, artificially charged tethers. The characteristic property of an E-sail based spacecraft is  
8           that the propulsive acceleration scales as the inverse Sun-spacecraft distance, and the thrust  
9           vector can be varied within about 30 degrees away from radial direction.

10          The aim of this paper is to estimate the transfer times required to fulfill a mission toward  
11          the near-Earth asteroid 1998 KY26. In doing so the propulsive acceleration of the E-sail, at  
12          a reference distance from the Sun, is used as a performance parameter so that the numerical  
13          results are applicable to E-sails of different sizes and different payload masses. The paper  
14          shows that the flight time scales nearly linearly with the inverse of the spacecraft maximum  
15          propulsive acceleration at 1 Astronomical Unit from the Sun, when the acceleration is greater  
16          than  $0.3 \text{ mm/s}^2$ . For smaller propulsive accelerations the relationship for the flight time is  
17          more involved, because the transfer trajectory is complex and more than one revolution  
18          around the Sun is necessary to accomplish the mission. The numerical analysis involves a  
19          sample return mission in which the total flight time is parametrically correlated with the

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20 starting date for a given E-sail propulsion system.

21 **Keywords:** Electric sail, near-Earth asteroid exploration, mission analysis.

## 22 NOMENCLATURE

$\mathbb{A}$	=	matrix $\in \mathbb{R}^{6 \times 3}$ , see Eq. (7)
$\mathbf{a}$	=	spacecraft propulsive acceleration
$a_{\oplus}$	=	spacecraft characteristic acceleration
$\mathbf{d}$	=	vector $\in \mathbb{R}^{6 \times 1}$ , see Eq. (8)
$e$	=	orbital eccentricity
$f, g, h, k$	=	modified equinoctial elements
$H$	=	Hamiltonian function
$i$	=	orbital inclination
$J$	=	performance index
$L$	=	true longitude
$p$	=	semilatus rectum
$r$	=	Sun-spacecraft distance, with $r_{\oplus} \triangleq 1$ AU
$\mathbf{r}$	=	spacecraft position vector
$t$	=	time
$\mathbf{x}$	=	state vector
$\alpha$	=	sail cone angle
$\Delta t$	=	flight time
$\boldsymbol{\lambda}$	=	adjoint vector
$\lambda$	=	adjoint variable
$\mu_{\odot}$	=	Sun's gravitational parameter
$\nu$	=	true anomaly
$\tau$	=	switching parameter
$\Omega$	=	right ascension of the ascending node

$\omega$  = argument of perihelion

23 *Subscripts*

AE = asteroid-Earth phase

EA = Earth-asteroid phase

$f$  = final

$i$  = initial

max = maximum

$w$  = waiting phase

$\odot$  = Sun

24 *Superscripts*

$\wedge$  = unit vector

( $ss$ ) = photonic solar sail

25 **INTRODUCTION**

26 The successful conclusion of Japanese Hayabusa mission, which, on 13 June 2010, re-  
27 turned to Earth with a material sample from asteroid 25143 Itokawa (Baker 2006), has  
28 renewed the scientific community's interest in studying those minor celestial bodies that  
29 populate the interplanetary space surrounding Earth. Unlike previous missions, such as  
30 Galileo, which first obtained close images of asteroids Ida and Gaspra during its flight to-  
31 wards Jupiter, or NASA's probe Near Earth Asteroid Rendezvous (NEAR) Shoemaker, which  
32 on 2001 touched down on asteroid 433 Eros, the Hayabusa mission has first demonstrated  
33 the technical feasibility of retrieving material samples from a near-Earth asteroid (Barucci

34 et al. 2011). This new frontier will be further advanced by Origins-Spectral Interpretation-  
35 Resource Identification-Security-Regolith Explorer (OSIRIS-REx) mission, whose launch is  
36 scheduled for 2016, and whose aim is to reach the potentially hazardous asteroid 1999 RQ36  
37 in 2019.

38 From a commercial viewpoint, returning an asteroid sample to the Earth is only the  
39 first step toward a future exploitation of resources from celestial bodies (Lewis 1996). In  
40 fact, asteroids contain materials that could be used for different purposes both in space  
41 and on Earth (Metzger et al. 2012). In scientific terms, an in-depth analysis of asteroid  
42 samples would guarantee a substantial knowledge improvement about the Solar System evo-  
43 lution (Barucci et al. 2011). In particular, a collection of samples with chemically unbound  
44 water from near Earth asteroids would represent a concrete step forward in understanding  
45 water distribution within interplanetary space, including the question from where Earth re-  
46 ceived its water. A main objective of OSIRIS-REx mission is, in fact, to return an asteroid  
47 sample to Earth, in order to reveal the presence of volatiles and organics that could represent  
48 the starting material for chemical evolution.

49 Within the set of asteroids that are accessible to spacecraft rendezvous, asteroid 1998  
50 KY26 has interesting characteristics. It was discovered in 1998 when it passed at 2.1 lunar  
51 distances from Earth (Ostro et al. 1999). It is a small asteroid of only 30 m diameter and,  
52 according to a spectral analysis, it is a carbonaceous body (C-type asteroid). Therefore, it is  
53 likely to contain water. Unfortunately, its fast rotation speed (about 0.11 rpm) poses severe  
54 constraints against the possibility of mining it to collect material samples. Nevertheless,  
55 the 1998 KY26 is chosen in this paper as a representative near-Earth asteroid candidate  
56 target for the preliminary design of a sample return mission for a spacecraft, whose primary  
57 propulsion system is an electric solar wind sail (E-sail).

58 The E-sail is an innovative deep space propulsion concept that uses the solar wind dy-  
59 namic pressure for generating thrust without the need of reaction mass (Janhunen 2010;  
60 Janhunen 2009; Janhunen et al. 2010). A spacecraft with an E-sail propulsion system, is

61 spun around its symmetry axis and uses the centrifugal force to deploy and stretch out a  
62 number of thin, long and conducting tethers, which are kept in a high positive potential  
63 by an onboard electron gun (Janhunen et al. 2010). The latter compensates the electron  
64 current gathered by the conducting tethers from the surrounding solar wind plasma.

65 Between the spacecraft and each of the tethers there is a potentiometer that allows each  
66 tether to be in a slightly different potential from the others. Because the thrust magnitude  
67 depends on the tether potential, this gives a way to control the thrust experienced by each  
68 tether individually and guarantees the possibility of attitude control of the tethers spin-  
69 plane. In fact the spin-plane can be turned by modulating the potentiometer settings by  
70 a sinusoidal signal synchronized to the rotation. The phase of the signal determines the  
71 direction in which the spin-plane turns and its amplitude regulates how fast such a turning  
72 occurs.

73 Under mild assumptions (Janhunen et al. 2010), a characteristic feature of an E-sail  
74 propulsion system is that the thrust produced is proportional to  $1/r$ , where  $r$  is the Sun-  
75 spacecraft distance. More precisely, such a thrust variation with the solar distance is valid  
76 provided that the potential sheath overlapping between different tethers is negligible, that the  
77 available electric power varies as  $1/r^2$ , and that the employed tether voltage is independent  
78 of  $r$ . However, the thrust vector control capability of an E-sail based spacecraft is moderate,  
79 because the thrust direction can be changed by inclining the spin-plane with respect to the  
80 solar wind flux (which is nearly coincident with the radial direction) within a cone whose  
81 half-width is around 30 deg. This peculiarity poses a challenge in mission analysis from the  
82 viewpoint of trajectory design, especially when a rendezvous-mission is considered.

## 83 **MISSION ANALYSIS**

84 The asteroid sample return mission is analyzed in an optimal framework (from the view-  
85 point of the flight time) as a function of the spacecraft characteristic acceleration  $a_{\oplus}$ , that  
86 is, the maximum propulsive acceleration at a reference distance  $r_{\oplus} \triangleq 1$  AU from the Sun.  
87 The E-sail based spacecraft is modeled as a point-mass vehicle with a constant mass and a

88 propulsive acceleration  $\mathbf{a}$  given by

$$89 \quad \mathbf{a} = a_{\oplus} \tau \left( \frac{r_{\oplus}}{r} \right) \hat{\mathbf{a}} \quad \text{with} \quad \arccos(\hat{\mathbf{a}} \cdot \hat{\mathbf{r}}) \triangleq \alpha \leq \alpha_{\max} \quad (1)$$

90 where the hat symbol denotes a unit vector, while the switching parameter  $\tau = (0, 1)$  is a  
 91 dimensionless coefficient that models the E-sail on/off condition and is introduced to account  
 92 for coasting arcs in the interplanetary trajectory. In particular, the spacecraft propulsive  
 93 thrust can be turned off ( $\tau = 0$ ) at any time by simply switching off the onboard electron  
 94 gun. The sail cone angle  $\alpha$ , that is, the angle between  $\hat{\mathbf{a}}$  and  $\hat{\mathbf{r}}$ , is assumed to have an  
 95 upper bound of  $\alpha_{\max} \triangleq 30$  deg. As a result, the propulsive thrust vector lies within a conical  
 96 region whose axis coincides with the Sun-spacecraft direction, see Fig. 1. Indeed (Janhunen  
 97 et al. 2007), only the component of the solar wind perpendicular to the tethers produces a  
 98 propulsive thrust, while the flow parallel to the tethers has no effect. As a result, a simple  
 99 geometrical consideration shows that for a set of spinning tethers inclined at an angle  $\theta$  with  
 100 respect to the solar wind flow, the net thrust is directed at an angle  $\alpha \simeq \theta/2$ . Cone angles  
 101 larger than about 30 deg are likely to be impractical because of the thrust reduction at high  
 102 values of  $\alpha$  and, possibly, due to mechanical instabilities.

103 The heliocentric orbital parameters of Earth and asteroid 1998 KY26 are taken from JPL  
 104 ephemerides (Standish 1998; Standish 1990) and are summarized in Table 1. All simulations  
 105 have been performed assuming a direct transfer (that is, without gravity assist maneuvers)  
 106 between the two celestial bodies. Also, the optimal transfer trajectory was found under the  
 107 assumption of a spacecraft deployment on a parabolic Earth escape trajectory, that is, with  
 108 zero hyperbolic excess with respect to the starting planet. This is a conservative hypothesis  
 109 in terms of mission transfer time. Also note that the effect of a hyperbolic excess energy  
 110 different from zero can be taken into account in the optimization process only provided that  
 111 the characteristics of the launch system are given.

112 The whole space mission can be ideally divided into three phases, as is schematically

113 shown in Fig. 2. In a first transfer phase, whose time length is  $\Delta t_{\text{EA}}$ , the spacecraft is  
 114 transferred from the Earth’s heliocentric orbit to the asteroid’s orbit. At the end of this phase  
 115 the spacecraft concludes its rendezvous maneuver with the target asteroid and maintains a  
 116 prescribed orbit relative to it. In the second phase, referred to as scientific phase, the  
 117 spacecraft completes the prescribed scientific measurements. The time interval of this second  
 118 phase is  $\Delta t_w$  (waiting time). In this phase a lander can be used to possibly reach the asteroid’s  
 119 surface and collect material samples. At the end of the second phase the lander (or part of  
 120 it) performs a docking maneuver with the E-sail based spacecraft. The third phase starts at  
 121 the docking instant and ends with an Earth’s rendezvous (with no hyperbolic excess), within  
 122 a time interval of  $\Delta t_{\text{AE}}$ . According to this simplified model the total mission time  $\Delta t$  is

$$123 \qquad \qquad \qquad \Delta t = \Delta t_{\text{EA}} + \Delta t_w + \Delta t_{\text{AE}}. \qquad (2)$$

124 For a given value of spacecraft characteristic acceleration  $a_{\oplus}$ , the flight times for the  
 125 first and last mission phases have been calculated by minimizing the time interval required  
 126 for the rendezvous-maneuvers (Earth-asteroid and asteroid-Earth phases). Accordingly, the  
 127 two values  $\Delta t_{\text{EA}}(a_{\oplus})$  and  $\Delta t_{\text{AE}}(a_{\oplus})$  have been calculated numerically as a function of the  
 128 starting date of each phase in terms of Modified Julian Date (MJD). The mathematical  
 129 model used in the trajectory optimization has been summarized in the appendix. Note that  
 130 the optimization algorithm is fully general, and can be applied, with minor changes, to a  
 131 wide range of mission scenarios. Due to numerical challenges and the number of required  
 132 iterations involved in solving this kind of problem, the solution has been obtained through  
 133 a two-step procedure, which is now described in detail.

### 134 **Orbit-to-orbit optimal trajectories**

135 Firstly (step one), minimum time, three-dimensional transfer trajectories have been cal-  
 136 culated, for phases one and three, by neglecting any ephemeris constraint on the two celestial  
 137 bodies. In other terms, within an orbit-to-orbit optimal trajectory, the spacecraft matches

138 the (Keplerian) heliocentric orbit of the two celestial bodies at both departure and arrival  
139 time instants. This implies that only the shape and orientation of the two orbits (corre-  
140 sponding to the data of Table 1) were taken into account in the optimization process, see  
141 the appendix. The numerical procedures used here have been validated in previous stud-  
142 ies (Mengali et al. 2008; Quarta and Mengali 2010). The main information that can be  
143 obtained from this analysis is the optimal performance (that is, the minimum flight time)  
144 during a single phase as a function of the spacecraft characteristic acceleration only and a  
145 rough estimation of the launch window within which the optimal transfer can take place.

146 To obtain a cost estimate of phases one and three, observe that each phase, in a two-  
147 impulse mission scenario, requires a total minimum velocity variation of about 3.4 km/s, that  
148 is, a velocity variation of 2.895 km/s at Earth's orbit, and a velocity variation of 0.505 km/s  
149 at asteroid's orbit. Therefore, a four-impulse sample return mission with a three phases  
150 scenario, see Fig. 2, requires at least a total velocity variation of about  $6.8 \text{ km/s} = 2 \times$   
151  $3.4 \text{ km/s}$ . Note that, according to the rocket equation, a velocity variation of 6.8 km/s  
152 corresponds to a propellant mass fraction of about 86.2% when a (conservative) specific  
153 impulse of 350 s is assumed. Within an optimal two-impulses scenario (for phases one and  
154 three), the Keplerian transfer trajectory, see Fig. 3, is characterized by a semimajor axis of  
155 1.248 AU, an eccentricity of 0.1858, an inclination of 0.6523 deg, a longitude of ascending  
156 node of 113.36 deg, and an argument of periapsis of 182.92 deg.

157 Of course, a velocity variation less than 3.4 km/s (for phases one and three) could be  
158 potentially obtained using a multiple-impulse transfer trajectory. In that case the total  
159 mission's cost should be evaluated, in an optimal framework, by minimizing the sum of the  
160 velocity variations for each impulse. However, such an analysis is beyond the aim of this  
161 paper.

162 The main results concerning the first mission phase have been summarized in Table 2,  
163 while Table 3 shows the corresponding results for the third (return) phase. The variable  
164  $\nu_i$  (or  $\nu_f$ ) represents the spacecraft heliocentric true anomaly along the starting (arrival)

165 orbit at the initial (final) time instant. Recall that the spacecraft's starting and arrival orbit  
 166 coincides with the heliocentric (Keplerian) orbit of one of the two celestial bodies of the  
 167 problem. The fifth column of Tables 2 and 3 shows the number of complete revolutions  
 168 around the Sun during the mission phase.

169 Tables 2 and 3 clearly show a rapid increase in the flight time by decreasing the spacecraft  
 170 characteristic acceleration  $a_{\oplus}$ . The simulation results can be fitted by closed-form expressions  
 171 that are useful for preliminary mission analysis. For example, flight time and spacecraft  
 172 characteristic acceleration are related by the following best-fit curve

$$\Delta t_{EA} \simeq 83.82 a_{\oplus}^{-1.201} \quad (3)$$

$$\Delta t_{AE} \simeq 83.39 a_{\oplus}^{-1.214} \quad (4)$$

173 where  $a_{\oplus} \in [0.1, 1] \text{ mm/s}^2$  and the flight time is expressed in days.

174 From Table 2 a rapid transfer to the asteroid 1998 KY26, that is, a mission whose transfer  
 175 time is less than one terrestrial year (Mengali and Quarta 2009), requires a characteristic  
 176 acceleration  $a_{\oplus} \geq 0.3 \text{ mm/s}^2$ . In this case, the mission is completed within less than a full  
 177 revolution about the Sun, see Figs. 4(a)–4(c), as is indicated by the fifth column of Table 2.  
 178 Instead, if  $a_{\oplus} < 0.3 \text{ mm/s}^2$ , the heliocentric trajectory is more involved, with multiple rev-  
 179 olutions around the Sun, see Fig. 4(c), and the flight time increases up to exceeding four  
 180 years if  $a_{\oplus} < 0.09 \text{ mm/s}^2$ , see Table 2.

181 A similar result holds for the return phase, and Figs. 5(a)–5(c) show a spacecraft trajec-  
 182 tory projection on the ecliptic plane.

183 To better emphasize the capability of performing such a mission type, the performance  
 184 of an E-sail are now compared to those of an ideal (photonic) solar sail. The latter may be  
 185 thought of as being equivalent to a perfectly reflecting flat surface. The propulsive accelera-

186 tion  $\mathbf{a}^{(ss)}$  provided by an ideal solar sail may be expressed as

$$187 \quad \mathbf{a}^{(ss)} = a_{\oplus}^{(ss)} \left( \frac{r_{\oplus}}{r} \right)^2 \cos^2 \alpha^{(ss)} \hat{\mathbf{a}}^{(ss)} \quad (5)$$

188 where  $\hat{\mathbf{a}}^{(ss)} \triangleq \mathbf{a}^{(ss)} / \|\mathbf{a}^{(ss)}\|$ , and  $\alpha^{(ss)} \in [0, \pi/2]$  is the solar sail cone angle, that is, the angle  
 189 between the direction of  $\hat{\mathbf{a}}^{(ss)}$  and that of the incoming photons. In an ideal solar sail the  
 190 propulsive acceleration unit vector  $\hat{\mathbf{a}}^{(ss)}$  coincides with the unit vector normal to the sail  
 191 nominal plane in the thrust direction.

192 In Eq. (5),  $a_{\oplus}^{(ss)}$  is the spacecraft characteristic acceleration, that is, the maximum propul-  
 193 sive acceleration at a distance equal to 1 AU from the Sun. Note that, unlike an E-sail, the  
 194 maximum modulus of the solar sail propulsive acceleration  $\hat{\mathbf{a}}$  varies as the inverse square dis-  
 195 tance from the Sun. Also, the solar sail cone angle  $\alpha^{(ss)}$  is only constrained by the condition  
 196 that the propulsive acceleration cannot be oriented toward the Sun.

197 Even though an E-sail and a solar sail are both propellantless propulsion systems, these  
 198 two systems are much different in terms of dimensions, required mass and also from the  
 199 viewpoint of the physical mechanism through which the thrust is produced. Therefore, a  
 200 comparison between the two propulsion systems must be performed with care and taking  
 201 into account not only the flight time, but also other quantities such as the characteristic  
 202 dimensions of the two systems and the allowable payload mass fraction.

203 It will be now emphasized that, within the mission scenario discussed in this paper, an  
 204 E-sail offers better performance than an ideal solar sail. To prove this claim, a compari-  
 205 son between the two propellantless propulsion systems is made under suitable assumptions.  
 206 The mission consists of an orbit-to-orbit transfer in which Earth-asteroid and asteroid-Earth  
 207 transfer phases are both analyzed. Assuming a flight time corresponding to the value neces-  
 208 sary for an E-sail to complete the mission, it is possible to calculate the minimum character-  
 209 istic acceleration  $a_{\oplus}^{(ss)}$  required by a solar sail to fulfill the same mission. The optimization  
 210 model necessary to solve such a problem has been adapted from Mengali and Quarta (2009),

211 to which the interested reader is referred for an in depth discussion. The numerical simula-  
212 tions results are summarized in the last column of Tables 2 and 3.

213 Note that, the flight time being the same, the characteristic acceleration required by a  
214 solar sail is greater than that corresponding to an E-sail, that is,  $a_{\oplus}^{(ss)} > a_{\oplus}$ . This implies  
215 that, from the viewpoint of the combination flight time-characteristic acceleration and for  
216 this particular mission scenario, an E-sail offers better performance than a solar sail. The  
217 reason of this result is closely related to the type of optimal trajectory obtained to fulfill the  
218 mission. Indeed, as is illustrated in Fig. 3, the heliocentric orbit of asteroid 1998 KY26 has a  
219 perihelion radius close to 1 AU (exactly equal to 0.9838 AU). Because the heliocentric orbit  
220 of Earth is nearly circular with a radius equal to 1 AU, it can be verified that the transfer  
221 trajectories for both an E-sail and a solar sail take place at a solar distance greater than  
222 1 AU. Under these conditions, as stated, an E-sail behaves better than a solar sail from the  
223 point of view of the propulsive acceleration's maximum modulus.

## 224 **Rendezvous constrained transfers**

225 The actual position of the two celestial bodies along their orbits must now be taken  
226 into account (second step of the procedure). The angular position of both Earth and target  
227 asteroid is obtained, in a simplified way, using a two-body dynamical model and the data of  
228 Table 1.

229 For a given value of the spacecraft characteristic acceleration and a time interval within  
230 which the mission must be fulfilled, it is possible to look for the best available launch window  
231 using the orbit-to-orbit optimal results from the previous section. In this study, a spacecraft  
232 characteristic acceleration  $a_{\oplus} = 0.1 \text{ mm/s}^2$  and a time interval of ten years from January 1,  
233 2020 will be assumed. The optimal launch window opens on August 2026 (MJD = 61254),  
234 when the transfer time is  $\Delta t_{EA} \simeq 1349$  days (this length is close to the orbit-to-orbit reference  
235 value of Table 2).

236 Note that a characteristic acceleration of  $0.1 \text{ mm/s}^2$  is expected to be a rather conservative  
237 value, which should be guaranteed by a full scale E-sail of the first generation. Also, current

238 estimates induce to think that such a value could be improved of an order of magnitude for  
239 an E-sail of the second generation. As such, a characteristic acceleration of about  $1 \text{ mm/s}^2$   
240 should be a reasonable value for a near term E-sail.

241 Having found an optimal solution with a planetary ephemerides constraint, the launch  
242 date was modified to get parametric relationships between mission starting date and flight  
243 time  $\Delta t_{EA}$ . The results of this analysis are summarized in Fig. 6. The figure shows a marked  
244 dependence of the flight time on the start date and the existence of a second (sub-optimal)  
245 launch window on June 2023 (MJD = 60098), with a corresponding flight time of about  
246 100 days longer than the optimal value.

247 The asteroid-Earth return phase was then investigated assuming a tentative waiting time  
248  $\Delta t_w$  of one year. A parametric study concerning the return phase involves the time  $\Delta t_{AE}$  as  
249 a function of the reentry date within the interval January 2028 – January 2033. The results  
250 of this numerical analysis are summarized in Fig. 7. In particular, Fig. 7 shows a sub-optimal  
251 launch date in October 2029 (MJD = 62420) with a flight time of about 1390 days. Such a  
252 result is not consistent with the optimal transfer of the first phase unless the starting launch  
253 window is moved earlier (with a consequent increase in the Earth-asteroid transfer time).  
254 This suggests calculating the end mission date as a function of the starting date and the  
255 waiting time by suitably combining the information from Figs. 6 and 7.

256 The results of this analysis have been summarized in Fig. 8, which shows the mission  
257 end date as a function of the launch window for different values of waiting time ( $\Delta t_w =$   
258  $\{0, 0.5, 1, 1.5, 2\}$  years). Note that  $\Delta t_w = 0$  corresponds to the limiting case in which the  
259 spacecraft performs an asteroid rendezvous and immediately starts the returning flight to  
260 Earth. A more interesting mission scenario is obtained when the spacecraft waits around the  
261 asteroid for a time interval sufficient to mine the asteroid's surface and collect the material  
262 samples. In this case, assuming  $a_{\oplus} = 0.1 \text{ mm/s}^2$  and a waiting time of about one year, Fig. 8  
263 shows that the whole mission can be fulfilled in slightly less than 11 years.

## 264 CONCLUSIONS

265 Even though the E-sail propulsion system is expected to enable high characteristic ac-  
266 celerations with small and moderate payloads, in an asteroid sample return scenario or in a  
267 mission application that involves a large scientific payload, a small spacecraft characteristic  
268 acceleration is also of practical interest. The asteroid 1998 KY26 has been chosen in this  
269 study as a significant candidate for a possible E-sail based mission. The main results of this  
270 paper are contained in Tables 2 and 3 where the E-sail flight time to and from the near-Earth  
271 asteroid 1998 KY26 are given, parameterized by the characteristic acceleration  $a_{\oplus}$ , which is  
272 determined by the payload mass relative to the E-sail size. A mission scenario that con-  
273 sideres the ephemeris constraints, shows that an E-sail with a characteristic acceleration of  
274  $0.1 \text{ mm/s}^2$  (a rather low value for a sail of the next generation) is able to complete a sample  
275 return mission, with a waiting time of one year, in about 11 years. Moreover the E-sail offers  
276 an interesting flexibility in the launch window. Indeed, the propulsion system's continuous  
277 thrust could be used to maintain the same Earth-return date by changing both the transfer  
278 time and the waiting time.

279 A final remark concerns the feasibility of further reducing the mission time. A lunar  
280 gravity assist maneuver can be used to obtain (or damp) a moderate hyperbolic excess speed  
281 when leaving or entering the Earth's trajectory. That happens, of course, at the cost of  
282 introducing further restrictions on the launch window. In particular, when returning to  
283 Earth, if the target is to bring the payload to ground, it is often acceptable to leave it on  
284 collision course with Earth with a nonzero hyperbolic excess speed. This last option will  
285 imply a reduction of travel time. In this sense the mission times calculated in this paper are  
286 conservative estimates.

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## APPENDIX: E-SAIL TRAJECTORY OPTIMIZATION

The equations of motion of an E-sail based spacecraft are written in terms of non-singular parameters as the Modified Equinoctial Orbital Elements (Walker et al. 1985; Walker 1986) (MEOE)  $p, f, g, h, k,$  and  $L$ . Accordingly, within a heliocentric inertial reference frame, the E-sail dynamics is described by the following first-order vectorial differential equation (Betts 2000):

$$\dot{\mathbf{x}} = a_{\oplus} \tau \left( \frac{r_{\oplus}}{r} \right) \mathbb{A} \hat{\mathbf{a}} + \mathbf{d} \quad (6)$$

where  $\mathbf{x} \triangleq [p, f, g, h, k, L]^T$  is the state vector of the problem, and  $\hat{\mathbf{a}}$  is the propulsive acceleration unit vector whose components should be expressed in a local-vertical/local-horizontal orbital reference frame, see also Eq. (1). In Eq. (6),  $\mathbb{A} \in \mathbb{R}^{6 \times 3}$  is a matrix in the form:

$$\mathbb{A} \triangleq \sqrt{\frac{p}{\mu_{\odot}}} \begin{bmatrix} 0 & \left[ \frac{2p}{1 + f \cos L + g \sin L} \right] & 0 \\ [\sin L] & \left[ \frac{(2 + f \cos L + g \sin L) \cos L + f}{1 + f \cos L + g \sin L} \right] & \left[ \frac{-g (h \sin L - k \cos L)}{1 + f \cos L + g \sin L} \right] \\ [-\cos L] & \left[ \frac{(2 + f \cos L + g \sin L) \sin L + g}{1 + f \cos L + g \sin L} \right] & \left[ \frac{f (h \sin L - k \cos L)}{1 + f \cos L + g \sin L} \right] \\ 0 & 0 & \left[ \frac{(1 + h^2 + k^2) \cos L}{2 (1 + f \cos L + g \sin L)} \right] \\ 0 & 0 & \left[ \frac{(1 + h^2 + k^2) \sin L}{2 (1 + f \cos L + g \sin L)} \right] \\ 0 & 0 & \left[ \frac{h \sin L - k \cos L}{1 + f \cos L + g \sin L} \right] \end{bmatrix} \quad (7)$$

where  $\mu_{\odot} \triangleq 132712439935.5 \text{ km}^3/\text{s}^2$  is the Sun's gravitational parameter, and the vector  $\mathbf{d} \in \mathbb{R}^{6 \times 1}$  is defined as

$$\mathbf{d} \triangleq \left[ 0, 0, 0, 0, 0, \sqrt{\mu_{\odot} p} \left( \frac{1 + f \cos L + g \sin L}{p} \right)^2 \right]^T \quad (8)$$

Note that  $p$  is the semilatus rectum of the spacecraft osculating orbit, whereas the transfor-

mations from MEOE to the classical orbital elements are given by

$$a = \frac{p}{1 - f^2 - g^2} \quad (9)$$

$$e = \sqrt{f^2 + g^2} \quad (10)$$

$$i = 2 \arctan \sqrt{h^2 + k^2} \quad (11)$$

$$\sin \omega = g h - f k \quad , \quad \cos \omega = f h + g k \quad (12)$$

$$\sin \Omega = k \quad , \quad \cos \Omega = h \quad (13)$$

$$\nu = L - \Omega - \omega \quad (14)$$

308 where  $a$  is the semimajor axis,  $e$  is the eccentricity,  $i$  the orbital inclination,  $\omega$  is the argument  
 309 of perihelion,  $\Omega$  is the longitude of the ascending node, and  $\nu$  is the true anomaly of the  
 310 spacecraft's osculating orbit. On the other hand, the transformations from classical orbital  
 311 elements to MEOE are

$$p = a (1 - e^2) \quad (15)$$

$$f = e \cos(\omega + \Omega) \quad (16)$$

$$g = e \sin(\omega + \Omega) \quad (17)$$

$$h = \tan(i/2) \cos \Omega \quad (18)$$

$$k = \tan(i/2) \sin \Omega \quad (19)$$

$$L = \Omega + \omega + \nu \quad (20)$$

312 In Eq. (6), the Sun-spacecraft distance  $r$  may be expressed as a function of MEOE as (Betts  
313 2000)

$$314 \quad r = \frac{p}{1 + f \cos L + g \sin L} \quad (21)$$

315 Consider the first phase of the sample return mission, that is, the Earth-asteroid phase  
316 (the same method can be easily extended to the last phase, or the asteroid-Earth phase).  
317 Assuming, as previously stated, a zero hyperbolic excess with respect to the starting planet,  
318 the initial spacecraft osculating orbit coincides with the Earth's (Keplerian) heliocentric  
319 orbit. Table 1 summarizes the planet's classical orbital elements.

320 The optimization problem consists of finding the minimum time trajectory that transfers  
321 the E-sail from the initial orbit to the asteroid's heliocentric orbit, for a given value of the  
322 sail characteristic acceleration  $a_{\oplus}$ . This amounts to maximizing the objective function  $J \triangleq$   
323  $-\Delta t_{\text{EA}}$ , where  $\Delta t_{\text{EA}}$  is the flight time of the first phase. Using an indirect approach (Betts  
324 1998), the optimal thrust direction  $\hat{\mathbf{a}}$  and the switching parameter  $\tau$  are obtained by means  
325 of Pontryagin's maximum principle, that is, by maximizing at any time the Hamiltonian of  
326 the system and taking into account the constraints  $\|\hat{\mathbf{a}}\| = 1$  and  $\alpha \leq \alpha_{\text{max}}$ . In particular,  
327 the Hamiltonian function of our problem is

$$328 \quad H \triangleq a_{\oplus} \tau \left( \frac{r_{\oplus}}{r} \right) \mathbb{A} \hat{\mathbf{a}} \cdot \boldsymbol{\lambda} + \mathbf{d} \cdot \boldsymbol{\lambda} \quad (22)$$

329 where  $\boldsymbol{\lambda} \in \mathbb{R}^{6 \times 1}$  is the adjoint vector

$$330 \quad \boldsymbol{\lambda} \triangleq [\lambda_p, \lambda_f, \lambda_g, \lambda_h, \lambda_k, \lambda_L]^T \quad (23)$$

331 whose time derivative is given by the Euler-Lagrange equations (Bryson and Ho 1975)

$$332 \quad \dot{\boldsymbol{\lambda}} = -\frac{\partial H}{\partial \mathbf{x}} \quad (24)$$

333 The explicit expressions of the Euler-Lagrange equations, together with the optimal values

334 of the controls  $\hat{\mathbf{a}}$  and  $\tau$  as a function of both the state vector and the adjoint vector, have  
 335 been evaluated using a symbolic math toolbox and are omitted here for the sake of brevity.  
 336 However, an in-depth discussion of the optimal control law can be found in Quarta and  
 337 Mengali (2010).

338 The optimal control problem is mathematically described by the six equations of motion  
 339 (6) and the six Euler-Lagrange equations (24). This differential system must be completed  
 340 with 12 suitable boundary conditions. For example, in an optimal orbit-to-orbit transfer, in  
 341 which both the initial and the final true longitude  $L$  are outputs of the optimization process,  
 342 the first 10 boundary conditions are the values of  $p$ ,  $f$ ,  $g$ ,  $h$ , and  $k$  on the (Keplerian)  
 343 heliocentric orbit of both the Earth (at the initial time  $t = 0$ ) and the asteroid 1998 KY26  
 344 (at the final time  $t = \Delta t_{\text{EA}}$ ), see Table 1 and Eqs. (15)–(19). The remaining two boundary  
 345 conditions, together with the constraint necessary to calculate the minimum flight time  $\Delta t_{\text{EA}}$ ,  
 346 are obtained by enforcing the transversality conditions (Bryson and Ho 1975; Casalino et al.  
 347 1998; Casalino et al. 1999), viz.

$$348 \quad \lambda_L(t = 0) = 0 \quad , \quad \lambda_L(t = \Delta t_{\text{EA}}) = 0 \quad , \quad H(t = \Delta t_{\text{EA}}) = 1 \quad (25)$$

349 where the Hamiltonian function is given by Eq. (22). Note that conditions (25) are necessary  
 350 but not sufficient for a global optimality of the transfer.

351 In a rendezvous constrained optimal transfer, twelve boundary conditions are the values  
 352 of the six MEOE on the (Keplerian) heliocentric orbit of both the Earth (at the initial  
 353 time) and the asteroid 1998 KY26 (at the final time). In this case, the actual value of  
 354 the true longitude  $L$  on the (Keplerian) heliocentric orbit of the two celestial bodies has  
 355 been evaluated using a two-body dynamics (that is, without orbital perturbations). In other  
 356 terms, in this simplified mathematical model, the MEOE  $p$ ,  $f$ ,  $g$ ,  $h$ , and  $k$  are constants of  
 357 motion. The minimum flight time is obtained by enforcing the transversality condition that  
 358 coincides, again, with the last of Eqs. (25).

359 The simulation results have been obtained by integrating the equations of motion (6),  
360 and the Euler-Lagrange equations (24) in double precision using a variable order Adams-  
361 Bashforth-Moulton solver scheme (Shampine and Reichelt 1997), with absolute and relative  
362 errors of  $10^{-12}$ . The two-point boundary-value problem associated to the variational problem  
363 has been solved through a hybrid numerical technique that combines genetic algorithms (to  
364 obtain a first estimate of adjoint variables), with gradient-based and direct methods to refine  
365 the solution (Mengali and Quarta 2005).

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orbital element	Earth+Moon barycenter	asteroid 1998 KY26
semimajor axis [AU]	$9.99996 \times 10^{-1}$	1.23199
eccentricity	$1.66928 \times 10^{-2}$	$2.01378 \times 10^{-1}$
inclination [deg]	$1.66709 \times 10^{-3}$	1.48113
long. of asc. node [deg]	176.223	84.4464
arg. of per. [deg]	286.744	209.182
mean anomaly [deg]	356.926	306.675

**Table 1. Reference orbital elements at MJD = 55927 (January 1, 2012).**

$a_{\oplus}$ [mm/s <sup>2</sup> ]	$\Delta t_{\text{AE}}$ [days]	$\nu_i$ [deg]	$\nu_f$ [deg]	revs.	$a_{\oplus}^{(ss)}$ [mm/s <sup>2</sup> ]
1	94.36	189.45	84.11	0	1.0883
0.9	97.85	188.24	85.87	0	1.0430
0.8	102.12	186.76	87.99	0	0.9937
0.7	107.44	184.92	90.59	0	0.94
0.6	118.92	181.49	96.39	0	0.8471
0.5	146.75	172.96	108.76	0	0.7009
0.4	200.91	156.65	129.55	0	0.5604
0.3	371.33	105.75	177.94	0	0.4578
0.2	568.89	170.42	110.35	1	0.2616
0.15	892.26	231.16	105.71	2	0.2161
0.14	929	205.72	105.32	2	0.2163
0.13	973.67	179.71	110.33	2	0.2159
0.12	1040.4	146.62	124.23	2	0.21
0.11	1211.3	106.24	176.75	2	0.1471
0.1	1347.9	204.23	106.98	3	0.1444
0.09	1449.8	152.32	122.22	3	0.1425

**Table 2. Orbit-to-orbit optimal performance for Earth-asteroid transfer phase.**

$a_{\oplus}$ [mm/s <sup>2</sup> ]	$\Delta t_{\text{EA}}$ [days]	$\nu_i$ [deg]	$\nu_f$ [deg]	revs.	$a_{\oplus}^{(ss)}$ [mm/s <sup>2</sup> ]
1	80.46	280.12	184.46	0	1.1003
0.9	83.27	278.91	185.74	0	1.0671
0.8	86.67	277.46	187.27	0	1.03
0.7	92.32	274.79	189.41	0	0.9751
0.6	112.42	265.34	196.05	0	0.8288
0.5	144.76	251.18	206.14	0	0.6879
0.4	205.56	227.74	223.86	0	0.5581
0.3	389.42	176.22	278.05	0	0.4335
0.2	573.67	247.43	211.1	1	0.2597
0.15	897.04	253.89	153.15	2	0.1862
0.14	934.24	253.31	177.76	2	0.1754
0.13	980.46	247.57	203.78	2	0.1649
0.12	1052.49	232.71	238.94	2	0.1541
0.11	1257.63	170.37	283.47	2	0.1395
0.1	1355	251.63	180.87	3	0.1245
0.09	1463.94	234.42	234.16	3	0.1143

**Table 3. Orbit-to-orbit optimal performance for asteroid-Earth transfer phase.**

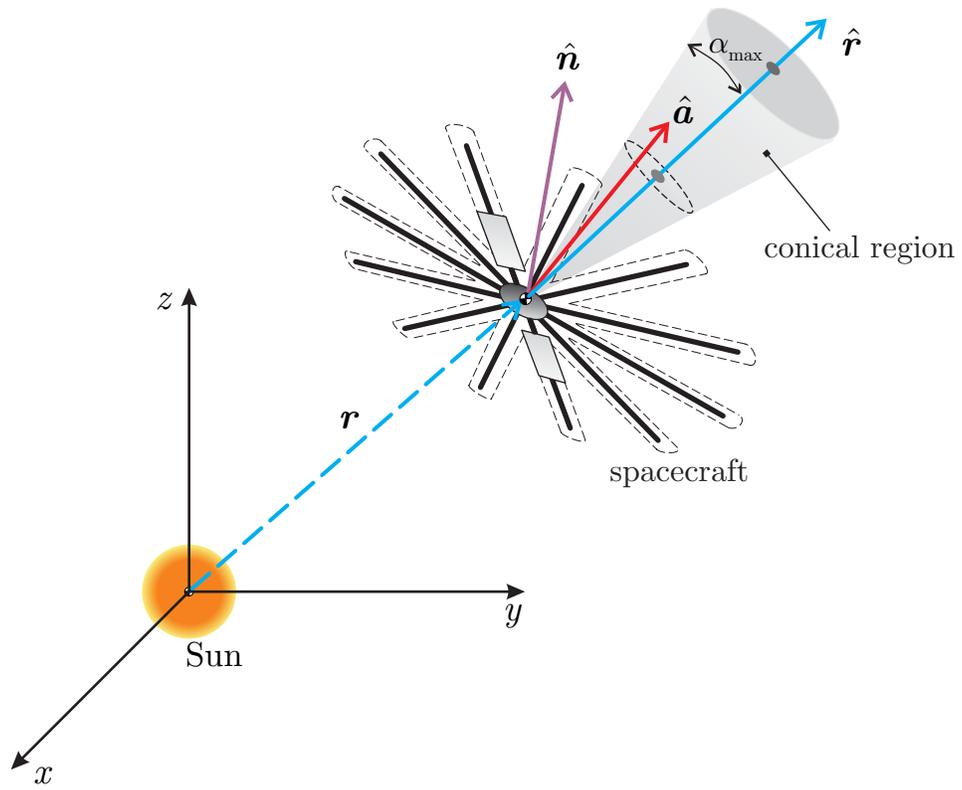
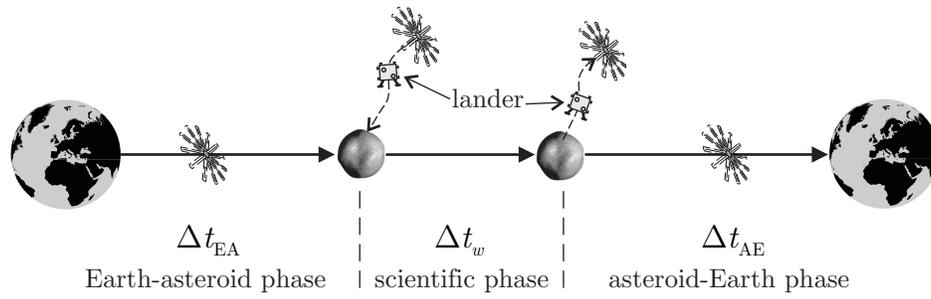
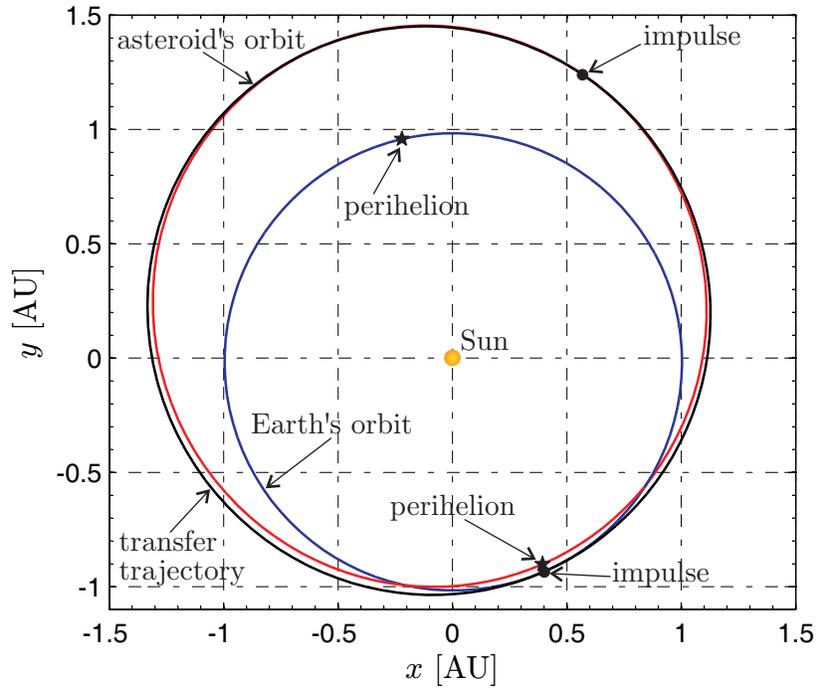


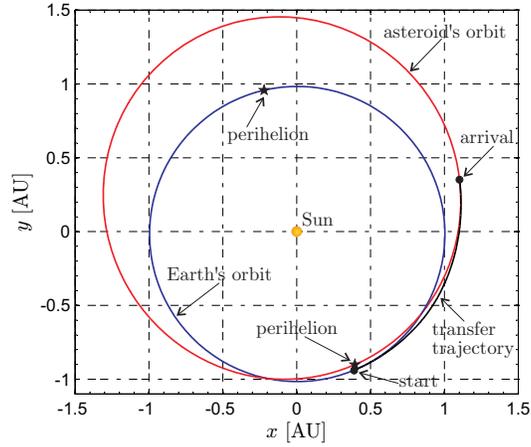
Figure 1. Spacecraft propulsive thrust vector direction.



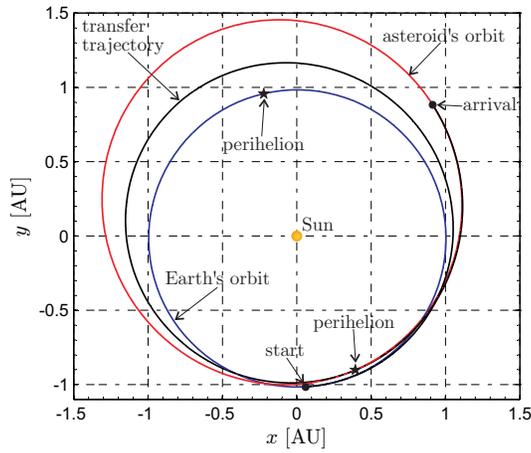
**Figure 2. Schematic view of mission phases.**



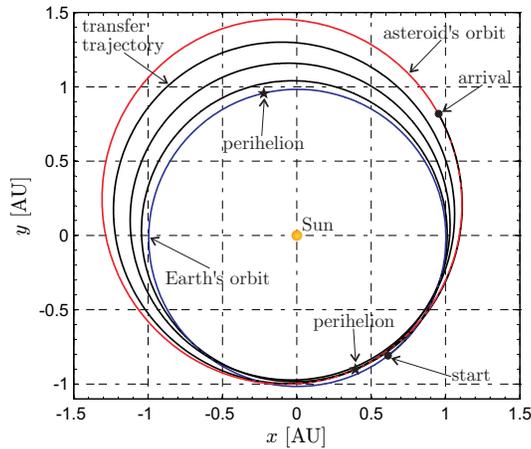
**Figure 3. Two-impulse transfer trajectory with minimum velocity variation (orbit-to-orbit transfer).**



(a)  $a_{\oplus} = 1 \text{ mm/s}^2$

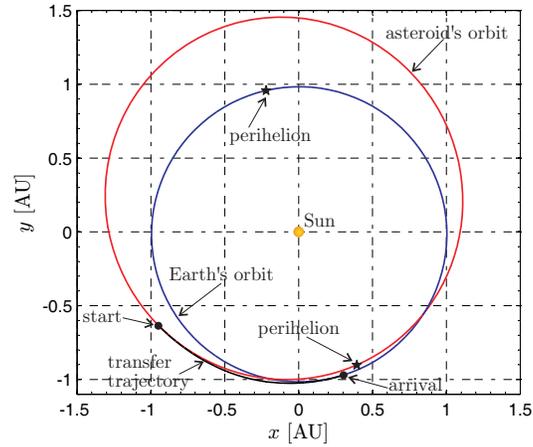


(b)  $a_{\oplus} = 0.2 \text{ mm/s}^2$  (1 rev.)

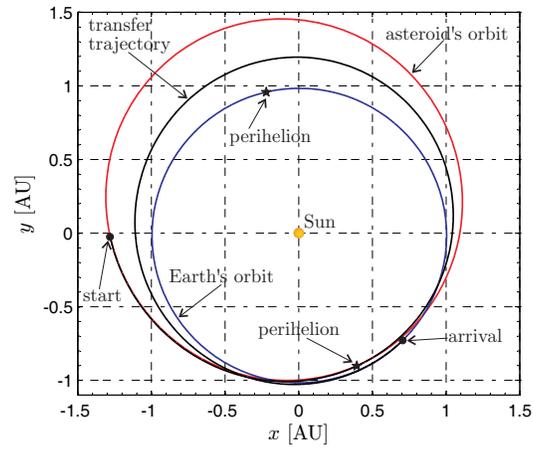


(c)  $a_{\oplus} = 0.1 \text{ mm/s}^2$  (3 revs.)

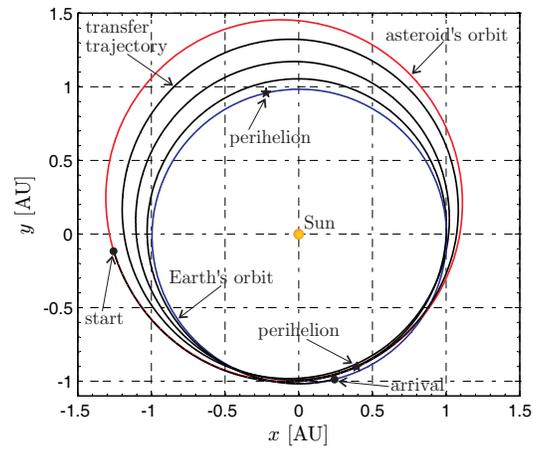
**Figure 4. Ecliptic projection of the (orbit-to-orbit) optimal Earth-asteroid transfer trajectory.**



(a)  $a_{\oplus} = 1 \text{ mm/s}^2$

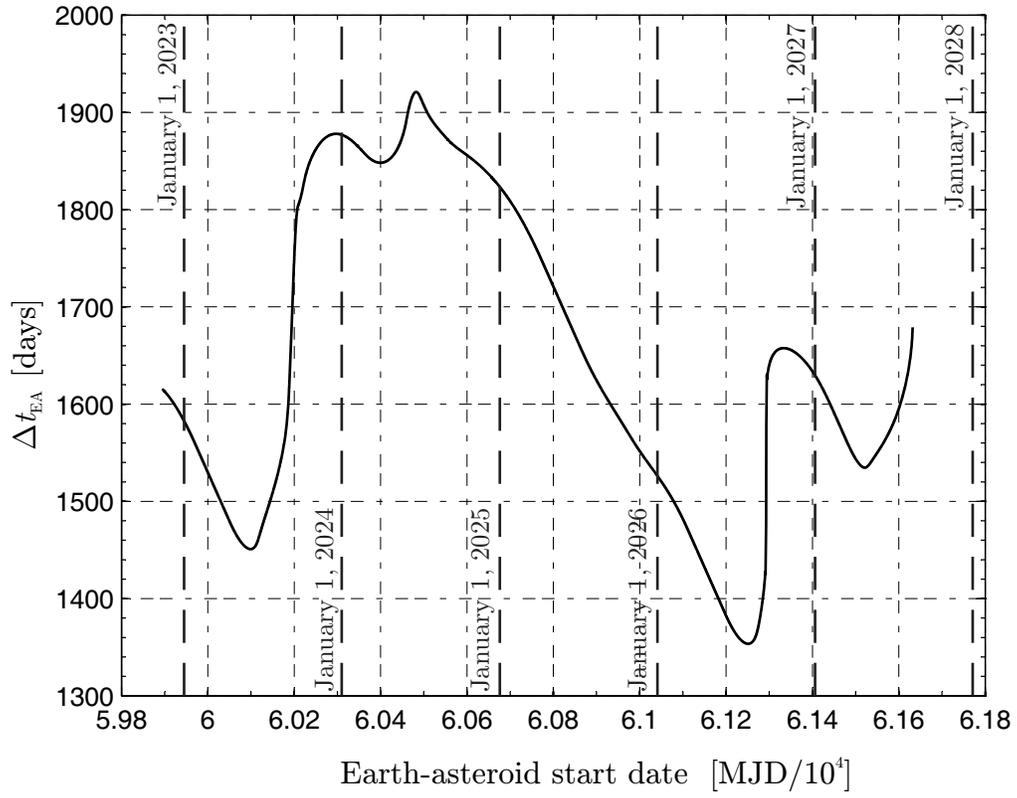


(b)  $a_{\oplus} = 0.2 \text{ mm/s}^2$  (1 rev.)

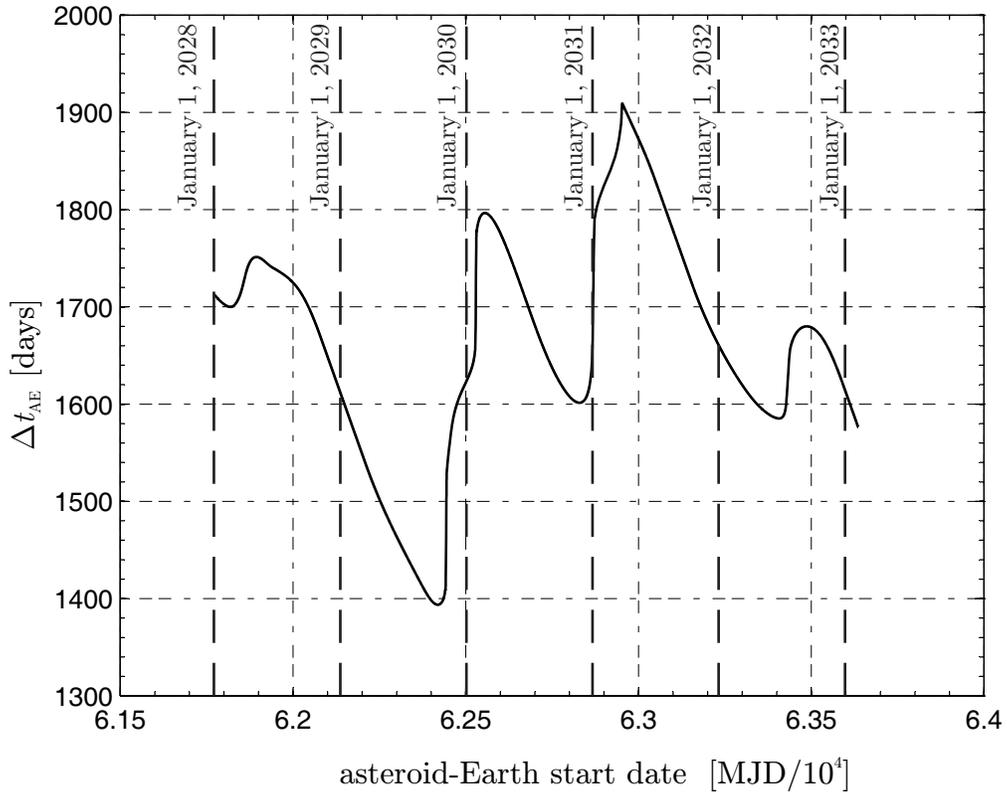


(c)  $a_{\oplus} = 0.1 \text{ mm/s}^2$  (3 revs.)

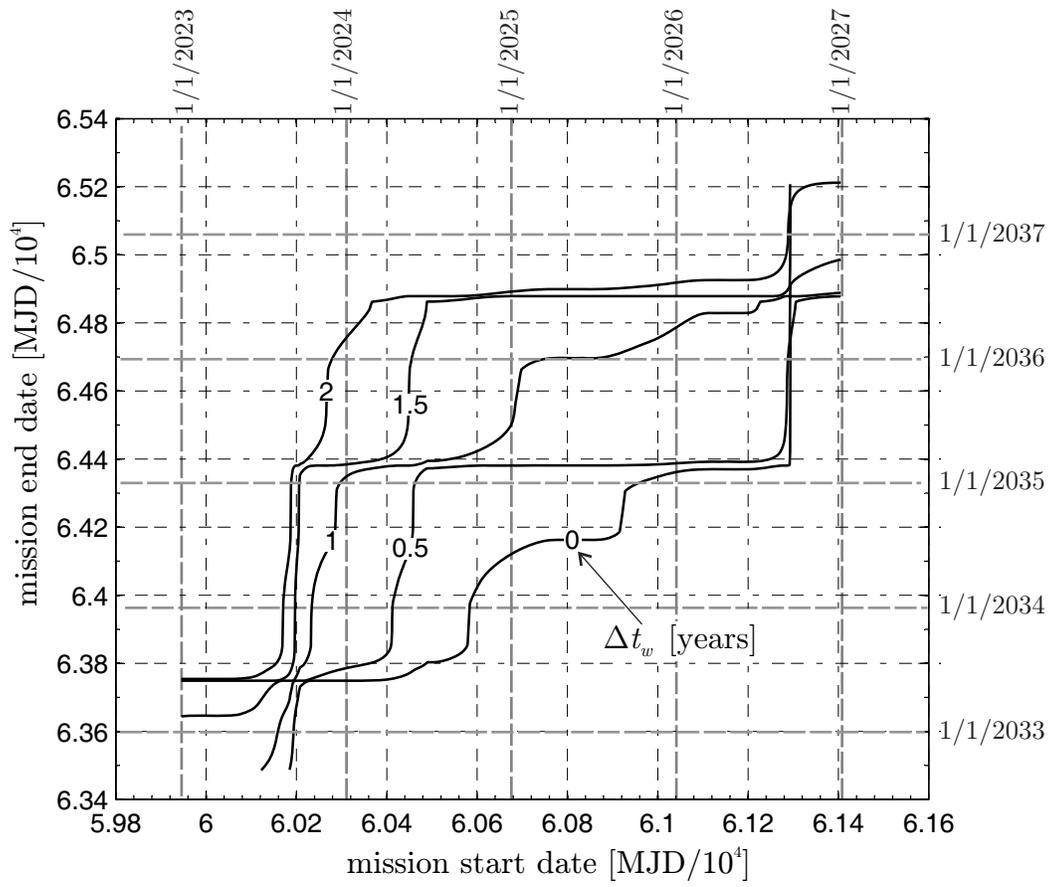
**Figure 5. Ecliptic projection of (orbit-to-orbit) optimal asteroid-Earth transfer trajectory.**



**Figure 6. Earth-asteroid minimum transfer time as a function of the Modified Julian Date for  $a_{\oplus} = 0.1 \text{ mm/s}^2$ .**



**Figure 7. Asteroid-Earth minimum transfer time as a function of the Modified Julian Date for  $a_{\oplus} = 0.1 \text{ mm/s}^2$ .**



**Figure 8.** Mission end date as a function of waiting time  $\Delta t_w$  and start date ( $a_{\oplus} = 0.1$  mm/s<sup>2</sup>).