Eur. Phys. J. Plus (2017) **132**: 367
DOI 10.1140/epjp/i2017-11644-0

THE EUROPEAN
PHYSICAL JOURNAL PLUS

Regular Article

Distant retrograde orbits and the asteroid hazard*

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Received: 7 July 2017
Published online: 29 August 2017 –

Abstract. Distant Retrograde Orbits (DROs) gained a novel wave of fame in space mission design because of their numerous advantages within the framework of the US plans for bringing a large asteroid sample in the vicinity of the Earth as the next target for human exploration. DROs are stable solutions of the three-body problem that can be used whenever an object, whether of natural or artificial nature, is required to remain in the neighborhood of a celestial body without being gravitationally captured by it. As such, they represent an alternative option to Halo orbits around the collinear Lagrangian points L_1 and L_2. Also known under other names (*e.g.*, quasi-satellite orbits, cis-lunar orbits, family-f orbits) these orbital configurations found interesting applications in several mission profiles, like that of a spacecraft orbiting around the small irregularly shaped satellite of Mars Phobos or the large Jovian moon Europa. In this paper a basic explanation of the DRO dynamics is presented in order to clarify some geometrical properties that characterize them. Their accessibility is then discussed from the point of view of mission analysis under different assumptions. Finally, their relevance within the framework of the present asteroid hazard protection programs is shown, stressing the significant increase in warning time they would provide in the prediction of impactors coming from the direction of the Sun.

1 Introduction

As it often happens when celestial mechanics and flight dynamics are involved, peculiar orbital configurations are discovered (or re-discovered) independently, according to the specific problem addressed. This is the case for the so-called Distant Retrograde Orbits (DRO) for their variety of applications in the space mission design. Originally studied by Michel Hénon in his extensive exploration of periodic orbits in Hill's problem [1], they represent a useful alternative to gravitationally bounded orbits when a space probe needs to remain for a long time in the neighborhood of a celestial body. Their very existence springs from the combination of the motion of two bodies revolving around a common primary which under certain assumptions and in an *ad hoc* rotating frame, results in one of the two bodies apparently orbiting the other on a large quasi-elliptical retrograde orbit.

In order to describe the underlying dynamics we take into consideration a simplified model where no mutual attraction between the two bodies is accounted for and the system is thus composed of two independent two-body problems. This is the case for example of the heliocentric Keplerian orbits of the massless Earth and of a spacecraft as shown in fig. 1 (left). The two orbits are chosen so that the Earth moves on a 1 AU circular orbit, while the spacecraft's orbit has the same semimajor axis (and therefore the same period of revolution by Kepler's third law) as that of the Earth but a non-zero eccentricity. Moreover, the two bodies are initially placed along their orbital path (*i.e.*, phased) so that conjunctions occur at the spacecraft's perihelion and aphelion passages. As seen from the Earth the spacecraft appears then to move faster nearing perihelion and slower toward aphelion (fig. 1 (left)), while at the same time it crosses the orbit of our planet from inside to outside and vice versa. In a rotating frame centered on the Earth and rotating with its angular speed such that the Sun and the Earth are fixed, the motion of the spacecraft resembles that of a distant retrograde satellite circling the Earth with a one-year period, although no direct interaction between the

* Contribution to the Focus Point on "Highlights of Planetary Science in Italy" edited by P. Cerroni, E. Dotto, P. Paolicchi.
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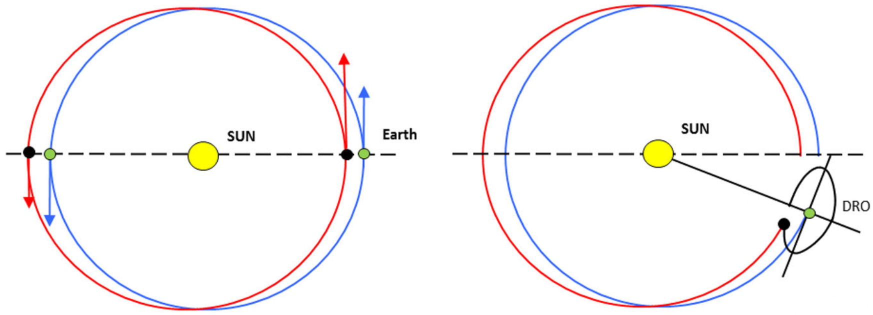

Fig. 1. The dynamical configuration of a DRO can be described, as a first approximation, using a double two-body system. The heliocentric orbit of the Earth (green dot) is drawn in blue and that of the spacecraft (black dot) in red. In the right plot the trajectory of the spacecraft with respect to the Earth in a reference frame rotating with our planet is shown.

two bodies has been included in the model (fig. 1 (right)). The one-year periodicity is ensured by the assumption that both bodies have the same period of revolution, while the size of the orbit, *i.e.* the maximum elongation along the track and with respect to the Earth-Sun line is linked to the orbital eccentricity.

Of course the "double two-body" model just described is no longer valid when introducing third-body perturbations, *i.e.* when adding the gravitational attraction of the Earth on the spacecraft. One must then resort to the investigations on the stability of the restricted three-body problem (in Hill's approximation [2]) carried out by Michel Hénon back in the late '60s [1]. DROs can in fact be identified as one of the five families of simple-periodic symmetrical orbits found, and in particular as the so-called family f of periodic orbits. It is worthwhile quoting Hénon's considerations on the subject: "Thus a remarkable result emerges: *the retrograde periodic orbits of family f are all stable*. This result is of some practical interest, since it means that retrograde satellites, either natural or artificial, can exist at very large distance from the second body, much farther than the Lagrange points. For example retrograde satellites of the Earth could exist at a distance of several millions of kilometers".

In what follows a review of some of the applications that have been translating Hénon's insight into actual mission opportunities is carried out and the Earth DRO family is characterized in terms of size and accessibility. Finally, an application for realizing an efficient NEO early warning system in space is described in some detail.

2 Space mission applications

DROs are not new to flight dynamics. Under the name of "fly-around" trajectories they are used in human spaceflight as an intermediate step during near-Earth rendezvous and docking phases for keeping two manned spacecraft permanently close but always at a safe distance [3]. A known example of this kind is the Soyuz trajectory circling the International Space Station for external inspection either in the approach phase or before returning to Earth. In this case, due to the negligible gravitational attraction between the two spacecrafts, the resulting dynamics follows the "double two-body" model described in the previous Section. Solar system exploration missions take full advantage of the DRO (or "quasi-satellite orbit") dynamics when aiming at rendezvous with a small, irregularly shaped body because of the stability properties. By changing the eccentricity of the orbit one can either approach closely the body or remain for long time in a relatively distant parking orbit. This is why within the framework of the NASA Phobos exploration, DROs are considered as the best compromise for accomplishing the mission objectives with minimum fuel consumption and station keeping operational requirements [4].

When more massive bodies are involved, as in the case of the Moon, DROs have the additional advantage of avoiding the need for expensive orbit insertion manoeuvres. This is the case for the US Asteroid Redirect Mission (ARM) concept for collecting a large boulder from a Near-Earth Asteroid (NEA) and deliver it in the vicinity of the Earth to be visited by a subsequent crewed mission. A Lunar DRO [5] would then represent an ideal solution for ensuring the necessary accessibility (thus maximizing the mass of the returned object) and stability (no escape or collisions with the Moon or the Earth) since the Sun perturbations can be safely neglected. This would allow the astronauts to explore an asteroidal body in cis-lunar space as an intermediate step toward the more complex interplanetary crewed missions needed to reach Mars and beyond. A further application has been proposed in the case of an Europa orbiter as quarantine orbit as well as representing stable paths for escape and capture around the Jovian moon [6].

When facing the challenges posed by protecting our civilization and its technological assets from space-born natural hazards, DROs can be used as sentinels in space for early warning. A NEA approaching from the direction of the Sun, where ground-based telescopes are blinded, and the occurrence of strong magnetic storms triggered by unusually high

Table 1. Summary of advantages and drawbacks of various orbit types for a space mission aimed at NEO detection. MEO stands for Medium Earth Orbit, GEO for Geostationary orbit and IEO for Inner Earth Objects, *i.e.* asteroids whose orbit is completely inside that of the Earth and that for this reason are extremely difficult to observe having always small solar elongations; orbit-type Hénon refers to the DROs described in this paper.

Approximate geocentric distance (AU)	Orbit type	Target NEOs	Main advantage	Main drawback
10^{-5}	Sun synchronous	IEOs, impactors	Small launcher	Short warning
10^{-4}	MEO GEO	Fireballs	Piggyback payload	No warning
10^{-3}	Moon	All	Large telescope	Challenging technology
10^{-2}	L_1	IEOs, impactors	Continuous observations	Large phase angle
10^{-1}	Hénon	IEOs, impactors	Always near Earth	Small constellation
10^{0}	Venus	IEOs, impactors	Long warning	Large distances

solar activity need to maximize the time span between detection and the arrival of a potential strike on our planet. In this respect monitoring the solar activity and detecting imminent Earth impactors bear the common need of placing a spacecraft as far as possible away from our planet on the sunward side. The difference relies in the telescope pointing, which for NEO translates into observing at favorable phase angles thus implying a generic anti-solar direction. In table 1 some orbital options available to this end are listed, including the low-altitude Sun-synchronous orbits typical of Earth observation missions, the halo orbits around L_1 extensively used for solar missions and the "Venus-like" orbit that according to the results of the ESA NEO Space Mission Initiative [7] would provide the best geometry for maximizing the NEO detection efficiency and warning time. The latter case has obvious drawbacks in terms of complexity and cost associated to building, launching and operating an interplanetary mission. Within this scenario, it has been pointed out [8] that a small constellation of satellites moving around the Earth in a moderately eccentric DRO would provide an acceptable compromise between detection efficiency, warning time and cost as will be discussed more extensively in sects. 3 and 4.

An additional advantage in using DROs for setting up a space-based early warning system is the possibility of using the well assessed low-energy transfer trajectories which exploit space manifold dynamics [9] for achieving a low-velocity escape in interplanetary space.

3 DRO accessibility

The accessibility of a DRO from the Earth can be evaluated using different dynamical models, *e.g.*, in [10] the departure orbit is a LEO. In what follows it has been assumed instead that the spacecraft reaches interplanetary space through L_1 with a relative velocity low enough to inherit the heliocentric velocity of the Earth. In this way no reference is due to the launch scenario and a simple Hohmann transfer can be used for computing the total velocity change (Δv) needed to reach the heliocentric orbit corresponding to a given stable DRO configuration.

In order to carry out a systematic exploration of the DRO accessibility, a method for generating them has been developed and discussed from the point of view of mission analysis. As introduced in sect. 1, a DRO can be considered a perturbed periodic orbit about a primary having the same period of revolution but a different (average) eccentricity as the perturbing body (second primary). When the two bodies are properly phased the resulting trajectory is an apparent retrograde orbit about the second primary, although no gravitational capture occurs. According to [11], initial conditions for DROs can be found in the Circular Restricted Three-Body Problem (CR3BP) in the limit of the mass of the second primary tending to zero (Hill's approximation). Note that this approximation falls, at its limit, in the "double 2-body" model described in sect. 2 and that the stability of the resulting DROs depends on how much the real system can be described by Hill's problem [1]. Therefore, the initial conditions found for Hill's approximation provide a good initial guess for finding a "general" DRO, *i.e.* a DRO in the full CR3BP. These latter DROs are still remarkably stable (about a hundred years in the planar case [6]) and therefore are of great interest for mission design. We will then focus on the derivation of general DROs in the planar CR3BP, and on the estimation of their accessibility in terms of Δv using a Hohmann transfer strategy.

In order to do so we set the Planar CR3BP in an inertial reference frame centered in the Earth-Sun barycenter, normalizing the units so that the Sun-Earth distance and the sum of the masses are unity and the period of revolution is 2 π. The equations of motion are given by

$$\begin{cases} \hat{q} = p, \\ \hat{q} = \dfrac{-\partial \hat{U}(q)}{\partial q}, \end{cases}$$

where

$$\hat{U}(q) = \frac{-(1-\mu)}{\sqrt{(q_x - \mu\cos(t+\pi))^2 + (q_y - \mu\sin(t+\pi))^2}} - \frac{\mu}{\sqrt{(q_x - (1-\mu)\cos(t))^2 + (q_y - (1-\mu)\sin(t))^2}}$$

is the gravitational potential due to the primaries, $q = [q_x, q_y]$ and $p = [p_x, p_y] = [\hat{q}_x, \hat{q}_y]$ are the position and momenta vectors, respectively, and $\mu = \frac{M_E}{M_E + M_S}$ is the scaled mass of the Earth.

An Earth-centered rotating frame is then adopted by properly shifting the center of the reference frame, scaling the distances by a factor $\gamma = \mu^{\frac{1}{3}}$ and setting it in constant velocity rotation such that the Sun is fixed at $[-1, 0]$. The transformation of coordinates to the new position and momenta vectors $x = [x, y]$ and $X = [X, Y]$ is given by

$$\begin{pmatrix} x \\ y \end{pmatrix} = \frac{1}{\gamma}\begin{pmatrix} \cos(t) & \sin(t) \\ -\sin(t) & \cos(t) \end{pmatrix}\begin{pmatrix} q_x \\ q_y \end{pmatrix} - \frac{1}{\gamma}\begin{pmatrix} 1-\mu \\ 0 \end{pmatrix}$$

and

$$\begin{pmatrix} X \\ Y \end{pmatrix} = \frac{1}{\gamma}\begin{pmatrix} \cos(t) & \sin(t) \\ -\sin(t) & \cos(t) \end{pmatrix}\begin{pmatrix} \hat{q}_x \\ \hat{q}_y \end{pmatrix} - \frac{1}{\gamma}\begin{pmatrix} 0 \\ 1-\mu \end{pmatrix}$$

yielding as equations of motion

$$\begin{cases} \hat{x} = X + y, \\ \hat{y} = Y - x, \\ \hat{x} = x + 2\hat{y} - \dfrac{(-1+\mu)}{\gamma} - \dfrac{\partial U(x)}{\partial x}, \\ \hat{y} = y - 2\hat{x} - \dfrac{\partial U(x)}{\partial y}, \end{cases}$$

where

$$U(x) = \frac{-(1-\mu)}{\gamma^2\sqrt{(\gamma x + 1)^2 + (y)^2}} - \frac{\mu}{\gamma^2\sqrt{x^2 + y^2}}.$$

Approximated initial conditions for distant retrograde orbits are then found with the aid of a Poincarè map, which is the projection of the orbit on the $y = 0$, $X = 0$ plane (*e.g.*, see [12]), for every value of a fixed parameter. Note that in order to obtain Hill's approximation the equations of motion would have to be expanded in Taylor series retaining only the 0th order. Figure 2 shows an orbit obtained by integrating the full planar CR3BP with initial conditions

$$\begin{cases} x_0 = [x_0, y_0] = [-6.9546,\ 0], \\ X_0 = [X_0, Y_0] = [0,\ 7.3519], \end{cases}$$

found with the method just described.

In the barycentric inertial system of reference, scaled back to the real units of measure, the initial conditions would be

$$\begin{cases} q_0 = [q_{x0}, q_{y0}] \cong [0.8997\,\mathrm{AU},\ 0], \\ p_0 = [p_{x0}, p_{y0}] \cong \left[0,\ 32.9450\dfrac{\mathrm{km}}{\mathrm{s}}\right]. \end{cases}$$

As shown in fig. 3, the first impulse for the Hohmann maneuver Δv_1 is provided when the spacecraft is in L_1 (*i.e.* on the Sun-Earth line) to inject the spacecraft into the transfer orbit. This is an elliptical orbit with perihelion at L_1 and aphelion at aphelion of the DRO, hereafter r_{DRO_a}. The initial velocity of the spacecraft at L_1 is

$$v_{L_1} = \sqrt{\frac{GM_S}{d_{Sun/L_1}}} 29.936\frac{\mathrm{km}}{\mathrm{s}}.$$

The velocity the spacecraft has to reach for the elliptical transfer, at its perihelion, is

$$v_{T_p} = \sqrt{GM_S\left(\frac{2}{d_{Sun/L_1}} - \frac{2}{d_{Sun/L_1} + r_{DRO_a}}\right)},$$

where G is the universal gravitational constant $6.674 \times 10^{-11}\ \frac{\mathrm{m}^3}{\mathrm{kgs}^2}$.

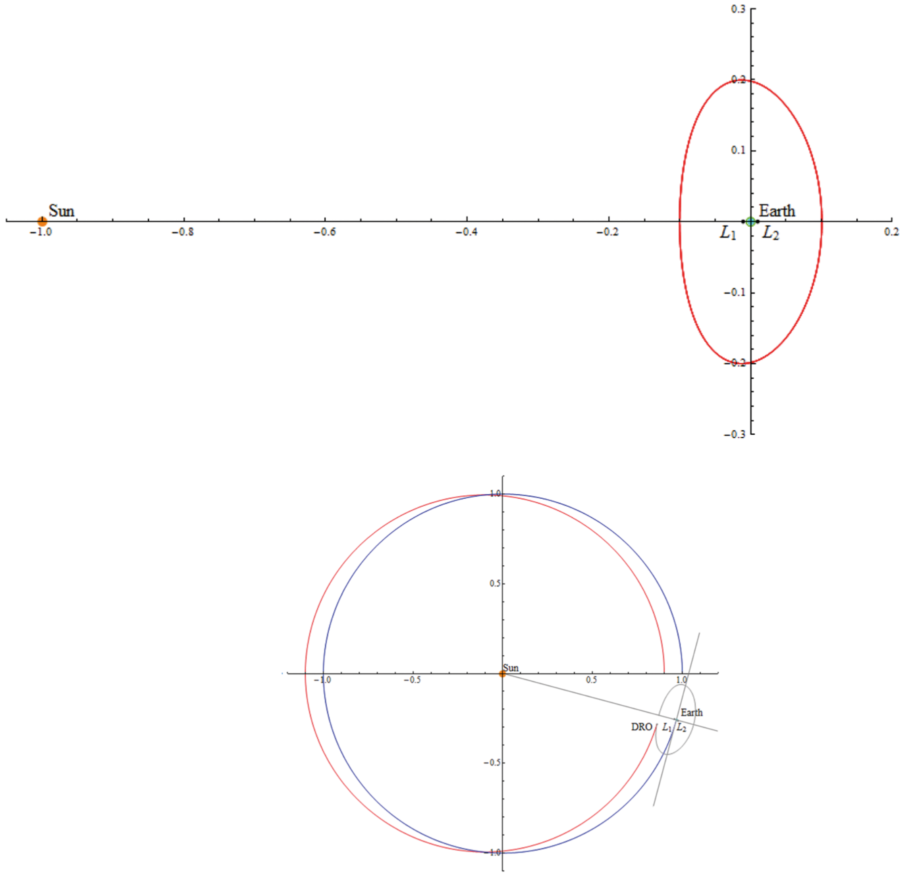

Fig. 2. Graphical representation of a DRO having minimum distance 0.1 AU from the Earth. The orbit is shown both in the rotating (anticlockwise), Earth centered system of reference (top) and in the barycentric, inertial system of reference (bottom). The position of the Sun and of the collinear Lagrangian points L_1 and L_2 on the X-axis are remarked.

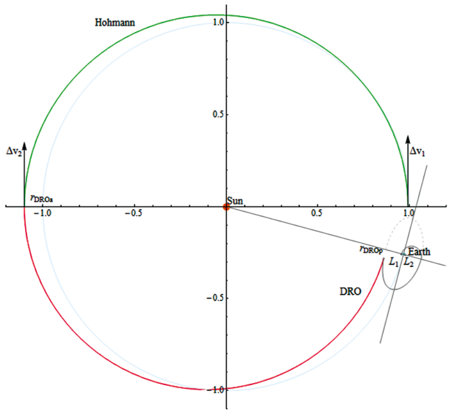

Fig. 3. Graphical representation of the Hohmann transfer (in green) to the DRO (in red) having minimum distance 0.1 AU from the Earth in the barycentric, inertial system of reference. The transfer orbit and the DRO are also shown in the rotating (anticlockwise), Earth centered system of reference (in grey). The two shoots Δv_1 and Δv_2 are displayed, notice that Δv_2 is against the direction of motion. The positions of the Sun and of the collinear Lagrangian points L_1 and L_2 are remarked on the X-axis.

The required shooting amount Δv_1 will thus be given by the difference between the desired velocity and the initial one, namely:

$$\Delta v_1 = v_{T_p} - v_{L_1}.$$

The second shoot Δv_2 instead is needed to brake at arrival at the aphelion of the transfer ellipse in order to match the instantaneous velocity of the chosen DRO at aphelion r_{DROp}. The velocity of the spacecraft at the aphelion of the

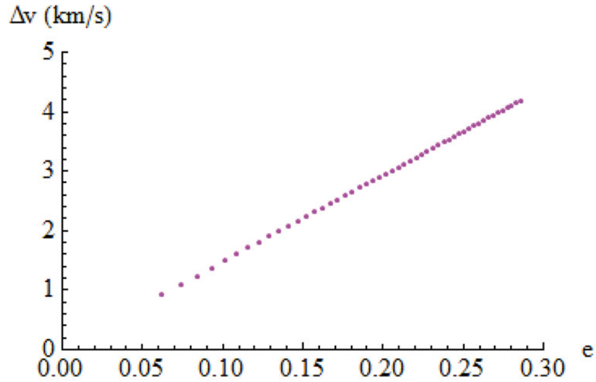

Fig. 4. The accessibility (Δv) of DROs around the Earth as a function of their average eccentricity, which in turn is linked to the size of the orbit.

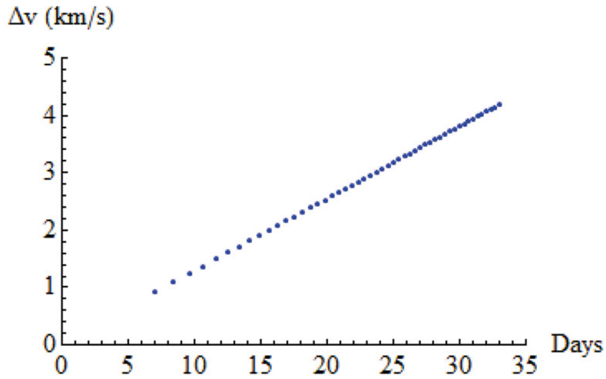

Fig. 5. Warning time with respect to accessibility (Δv) of the DROs reported in fig. 4.

transfer orbit is

$$v_{T_a} = \sqrt{GM_S \left(\frac{2}{r_{DRO_a}} - \frac{2}{d_{Sun/L_1} + r_{DRO_a}} \right)},$$

while the velocity in the DRO at its maximum distance from the Sun v_{DRO_a} (corresponding to the osculating eccentricity and semimajor axes of the heliocentric orbit) has been obtained by numerically integrating the DRO pattern.

The amount of braking required is $\Delta v_2 = v_{DRO_a} - v_{T_a}$ which is negative as the shooting is directed against the direction of motion. The total amount of Δv required is then given by the sum of the two contributions:

$$\Delta v = \Delta v_1 \vee -\Delta v_2 \vee .$$

By repeating iteratively this procedure a general picture of the accessibility of DRO configurations at increasing distance from the Earth is obtained (fig. 4). If one wants to evaluate the overall Δv to the figures computed in this way one should add the contribution from launch to the Earth escape branch of the trajectory.

These results are relevant when addressing the hazard detection system introduced in sect. 2, which foresees the use of early warning satellites placed in an Earth DRO configuration. In this case the need of reaching out as far as possible from the Earth is limited by cost effectiveness considerations, such as the capability of the launcher and of the on-board propulsion and the dimensioning of the telecommunication subsystems. Assuming a typical Near-Earth Asteroid geocentric speed at approach of about 15 km/s in fig. 5 is displayed the relation between the DRO accessibility and the warning time.

4 The NEO imminent impactors problem

On 15 February 2013 two remarkable events occurred almost simultaneously: the close passage of asteroid (367943) Duende (then known with its provisional designation 2012 DA_{14}) and the Chelyabinsk superbolide, *i.e.* a small asteroid (17–20 m in size) entering the atmosphere at high speed (18.5 km/s) and exploding at about 20 km altitude.

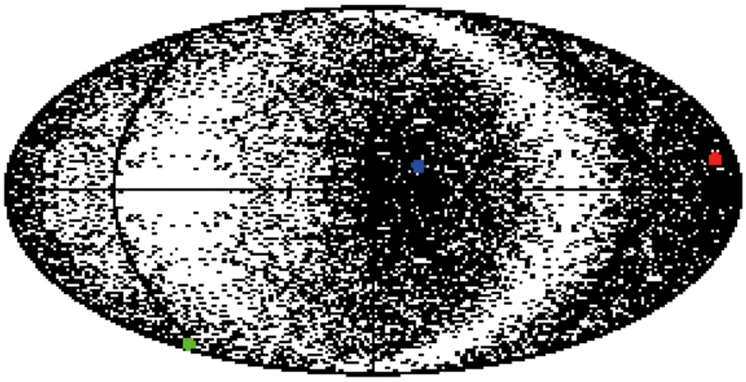

Fig. 6. Radiants can be plotted using an equal area projection centered on the opposition and using the ecliptic as reference plane; the angular coordinates are ecliptic longitude minus the longitude of the Sun, and ecliptic latitude. Therefore, in this representation the center of diagram corresponds to the antisolar direction, opposite to the right and left contours. The black dots represent the impactors population model developed in [13].

The lessons that can be learnt for NEO hazard monitoring are twofold:

- the validity of the design choice of the NEO Segment of the ESA Space Situational Awareness programme, *i.e.* to focus on the detection of small objects approaching our planet, has been confirmed;
- a major limitation of present and near-future ground- and space-based NEO surveys has been highlighted.

These conclusions can be drawn by analyzing the details which characterized the two events. Asteroid 2012 DA_{14} had been discovered in February 2012, so its approach to the Earth could be computed in advance even if the 2013 flyby geometry was not favorable. The Chelyabinsk event was not predicted, although in October 2008 a much smaller asteroid, 2008 TC_3 had been discovered and tracked almost a day before its fall. The reason is that the Chelyabinsk impactor approached the Earth from the direction of the Sun, thus it could not be discovered telescopically, as it would have been in the daytime sky. The situation is summarized in fig. 6, where the radiants of 2012 DA_{14} (green), of Chelyabinsk (red) and of 2008 TC_3 (blue) are highlighted.

Present- and near-future ground-based surveys are effective in discovering small impactors only in the long run [14] —*e.g.* it is possible to show that the survey envisaged for the NEO Segment of the ESA Space Situational Awareness [15] would almost reach a 90% probability of discovering the next Tunguska-class impactor (50 m in size) before impact in about one century from its start. The contribution of space missions either devoted to NEO detection (NEOSSAT) or where NEOs are not primary targets (*e.g.*, WISE, Gaia) have intrinsic limitations, not only because of operational constraints, but also because for impactors coming from very nearly the direction of the Sun, as in the Chelyabinsk event, the extremely low phase angle cuts sharply the brightness of the object.

In principle this limitation can be overcome by placing a spacecraft on a heliocentric orbit allowing to observe objects approaching the Earth from the direction of the Sun with a favorable phase angle, *i.e.* from a position closer to the Sun than the Earth. The mission scenario fulfilling this observational requirement makes use of the already mentioned "Venus-type orbit" (an eccentric heliocentric orbit located between Venus and Earth) as originally adopted for the *Euneos* proposal [7] and envisaged for the US *Sentinel* mission. Yet the realization of an interplanetary mission may rise cost-effectiveness issues and a number of further considerations come into play.

It is then worthwhile comparing the mission options summarized in table 1 and the DRO accessibility figures discussed in sect. 3. The first two entries of the table refer to Earth-bound orbits; these are easily accessible, and allow to observe close to the Sun, but the advantage over ground telescopes that they would have in terms of warning time for an imminent impactor is, in all cases, negligible. In particular, satellites in the MEO region could be useful to detect fireballs, when they occur, but would be useless for early warning of imminent impactors. Telescopes on the far side of the Moon would enjoy a quiet environment, but their advantage in terms of warning time would, again, be negligible. Satellites at L_1 would be more efficient than those in LEO, but the warning time would still be too short, of the order of the day. If the spacecraft is in an eccentric heliocentric orbit interior to that of the Earth, its sensor can point in the anti-solar direction, to spot objects when their brightness is near maximum due to the small phase angle. However, such an orbit suffers from the drawback that the difference of heliocentric longitude between the spacecraft and the Earth goes from 0 to 180 degrees making the communications difficult and impairing, at times, the ability to spot imminent impactors. Within this framework DROs are the only option which offer several choices in terms of accessibility, warning time and mission complexity thus representing an affordable alternative strategy for protecting the Earth from small impactors coming from the direction of the Sun.

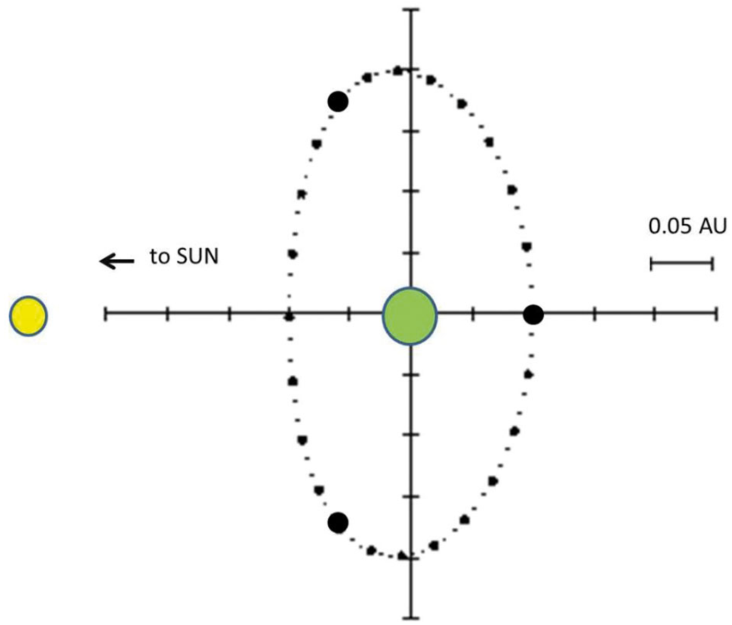

Fig. 7. The Hénon mission trajectory in a reference frame rotating with the Earth (the Sun is on the negative X-axis); tic marks are spaced by 0.05 AU.

Table 2. First column: time; second and third column: geocentric distance and angle between the Earth-spacecraft direction and the Earth-antiSun direction for the first spacecraft; following columns: the same, for the second and the third spacecraft.

t (d)	r (AU)	α (deg)	r (AU)	α (deg)	r (AU)	α (deg)
0	0.100	0.00	0.184	−108.36	0.184	108.36
15	0.108	27.44	0.163	−117.78	0.197	100.50
30	0.129	−48.73	0.137	−130.72	0.200	93.07
45	0.153	−63.96	0.112	−150.32	0.193	85.30
60	0.175	−75.28	0.100	−178.23	0.177	76.43
75	0.191	−84.34	0.109	153.16	0.156	65.45
90	0.199	92.21	0.133	132.55	0.131	50.80
105	0.197	−99.63	0.160	118.99	0.110	30.32
120	0.186	−107.39	0.182	109.26	0.100	3.40

5 The Hénon mission

In what follows we draw a consistent scenario for a space system profiting at best of the advantages of the DRO configurations. In particular we focus on heliocentric orbits in which the semimajor axis is almost exactly 1 AU, the inclination is very small, so that excursions above and below the ecliptic plane are small, and the eccentricity is small to moderate, *e.g.* 0.1. A spacecraft moving on such an orbit (fig. 7) would appear circling the Earth in retrograde motion with a period of one year, undergoing a radial maximum excursion of 0.1 AU in both the solar and antisolar direction, and an excursion in longitude such that it would, at the extrema, precede/follow the Earth by about 0.2 AU [16,8].

A spacecraft equipped with a telescope even of modest size (70 cm aperture diameter) pointing toward the Earth, at its largest sunward radial excursion would spot impactors while they are within the last 0.1 AU of their trajectory towards our planet at extremely favorable illumination conditions allowing about 10 days warning time (fig. 5). As demonstrated by the 2008 TC3 and the Chelyabinsk event, this is an acceptable time span for reliably computing the impact location and undertake proper mitigation actions (either informing the population on how to behave or evacuate the interested areas).

The only drawback is that the favorable geometry would last only while the spacecraft is on the sunward branch of its trajectory. Foreseeing a small constellation of three spacecrafts, appropriately spaced, into the same orbit would guarantee complete coverage of the region immediately interior to the Earth, since there would be at any time at least one spacecraft optimally placed for impactor monitoring. In fig. 7 the motion of the three spacecrafs in a geocentric rotating frame is shown; the tic marks are spaced by 0.05 AU, and the three largest dots (at $x = 0.1$, $y = 0$, at $x = -0.06$, $y = -0.17$, at $x = -0.06$, $y = 0.17$) show the spacecrafts at a certain epoch. After that epoch, the spacecrafts move in the clockwise direction; small dots show the positions every 5 days, and intermediate size dots show every 15 days.

Table 2 gives some additional quantitative data on the proposed constellation, namely the geocentric distance of each spacecraft and the angle, seen from the Earth, between each spacecraft and the antiSun direction, at 15 d intervals. Note that after slightly more than 120 d the geometry repeats.

6 Conclusions

The DRO peculiar dynamical configurations have been shown to bring several advantages when designing a mission devoted to timely detect imminent impactors of the Earth. When focussing on the contribution that the proposed 3-spacecraft constellation would bring to the ESA SSA Programme, the following considerations are meaningful:

- *accessibility*: the estimated Δv for the chosen DRO is on the lower end with respect to those characterizing interplanetary missions and the possibility of remaining always relatively close to our planet is less demanding in terms of telecommunications and mission operations.
- *efficiency*: focusing on relatively small energy events (*i.e.*, bolides, fireballs, meteorite falls) it fulfills the most frequent SSA NEO segment customer needs (*e.g.*, insurance companies) allowing the onset of suitable mitigation measures (*e.g.*, alert, avoidance, evacuation);
- *completeness*: the availability of a space based asset would complement the SSA NEO segment design, covering a critical region of the sky not observable by ground-based assets, *e.g.* [17].

Finally, DRO trajectories cross regions of space of the highest interest also for Space Weather, *i.e.* interplanetary space and the extreme limb of the magneto-tail. Therefore, the possibility of hosting in the same spacecraft space weather payload would significantly increase the scientific return of such a mission.

Part of this work has been done under ESA Contract No. 4000117371/16/D/MRP Maintenance of COTS, application SW and HW components for NEO Segment. The authors would like to thank Pasquale Lafiosca for his contribution to the numerical algorithm implementation.

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