ON THE END-OF-LIFE DISPOSAL OF SPACECRAFT AND ORBITAL STAGES OPERATING IN INCLINED GEOSYNCHRONOUS ORBITS

Luciano Anselmo, Carmen Pardini

Space Flight Dynamics Laboratory, Institute of Information Science and Technologies (ISTI)
National Research Council (CNR), Via G. Moruzzi 1, 56124 Pisa, Italy
Email: Luciano.Anselmo@isti.cnr.it; Carmen.Pardini@isti.cnr.it

ABSTRACT

The current internationally accepted guideline for the appropriate end-of-life disposal of spacecraft and orbital stages, in order to guarantee no further long-term interference with the geosynchronous protected region, was developed when the nearly sole function of the geosynchronous region was to host geostationary satellites, maintained close to the Earth equator as much as possible. Significantly inclined geosynchronous orbits are nevertheless becoming increasingly attractive for various applications, so the original re-orbiting guideline was re-assessed to evaluate its application limits. It was found that an extension of the original formula would be possible to cover disposal inclinations up to ≈ 30°, with an additional velocity variation penalty ≤ 11.5 m/s. However, this approach will not be generally applicable for higher initial inclinations, were the disposal strategy should be addressed on a case by case basis, being strongly dependent on the initial orbital and celestial conditions.

1. INTRODUCTION

Having in mind the goal of preserving the utilization of the geostationary orbit, the Space Debris Mitigation Guidelines, compiled and issued by the Inter-Agency Space Debris Coordination Committee (IADC), defined a toroidal geosynchronous protected region, extending 200 km below and above the geosynchronous altitude of 35 786 km, and spanning in latitude ± 15° with respect to the Earth equatorial plane, and introduced a simple formula for the appropriate end-of-life disposal of geostationary spacecraft, in order to guarantee no further interference with the protected region over the long-term [1]. The requirement of remaining outside the protected region after mission completion was applicable not only to spacecraft, but also to any separable propulsion system (if any) or orbital stage used for reaching the geosynchronous region.

The "IADC formula" requests the fulfillment of the following two conditions for a successful disposal orbit above the geosynchronous protected region [1]:

1. A minimum increase of the perigee altitude h_p of:

$$\Delta h_p[\mathrm{km}] \geq h_0 + (1000 \cdot C_R \cdot A/M) \qquad (1)$$

where C_R is the solar radiation pressure coefficient, A is the average aspect area (in m^2) of the re-orbited object, and M its mass (in kg). h_0 is set equal to 235 km, representing the sum of the upper limit of the protected region (+ 200 km) and of the maximum descent of the re-orbited object due to luni-solar & geopotential perturbations (35 km).

2. An initial eccentricity:

$$e_0 \leq 0.003 \qquad (2)$$

Having the possibility of implementing an appropriate choice of the initial conditions of the disposal orbit, the IADC guideline, i.e. avoiding the interference with the protected region over the long-term, might be fulfilled even with different and slightly more convenient, from the propellant consumption point of view, re-orbiting strategies [2], but the purpose of the IADC formula was just to offer a simple and straightforward approach irrespective of initial conditions, aside from perigee altitude and eccentricity.

The geosynchronous protected region definition and the end-of-life disposal formula were however elaborated when the nearly exclusive utilization of the geosynchronous region consisted of geostationary satellites placed and maintained, during their operational lifetime, close to the Earth equatorial plane. When finally abandoned, due to the concurring action of geopotential and luni-solar perturbations, such objects display a characteristic periodic orbit plane evolution, with a period of about 53 years and a maximum inclination of about 15°.

More specifically, the angular momentum of the orbit, as seen from the North Pole, is characterized by a counterclockwise precession around an axis lying between the rotational axis of the Earth and the pole of the ecliptic, approximately 7.4° away from the polar axis of the Earth. Being one half of the aperture of the precession cone nearly 7.4° as well, the orbital inclination with respect to the equatorial plane of the Earth goes from zero to about 15° and then again to zero in approximately 53 years, completing a full precessional cycle. It is the superimposition of three precessions: around the rotational axis of the Earth, mainly due to the planet oblateness; around the ecliptic pole, due to the third body attraction of the Sun; and around the angular momentum of the Moon orbit, due to the third body attraction of the natural satellite [3]. The plane normal to the pole of the total precession is the Generalized Laplace Plane (GLP) of the system made of Sun, Moon and oblate Earth.

In 2017, the situation in and around the geosynchronous region, with semi-major axis 37 948 km $\leq a \leq$ 46 380 km and eccentricity $e \leq 0.25$, was still largely dominated by objects with orbital inclination with respect to the Earth equatorial plane $i \leq 18°$. On 1 January 2017, among a total tally of 1533 objects, only 19, i.e. 1.2%, had $i > 18°$, i.e. a Highly Inclined Geosynchronous Orbit (HIGSO) [4]. As of 31 July 2017, their number had grown by just one more, to 20, 19 spacecraft and 1 mission related object, i.e. the telescope dust cover of the International Ultraviolet Explorer (IUE).

However, significantly inclined geosynchronous orbits are now becoming increasingly popular for various applications, like satellite navigation systems, as the IGSO component of the Chinese Beidou ($i = 53°$-$58°$), the Indian IRNSS ($i = 28°$-$30°$), and the Japanese QZSS ($i = 40°$-$45°$), science (e.g. the NASA's Solar Dynamics Observatory, with $i = 29°$), telecommunications and intelligence, and this trend may rapidly increase in the future. In the light of these developments, the aim of this paper is to review the current definition of the IADC geosynchronous mitigation guideline, assessing if it would need an extension. Special attention is paid to the end-of-life disposal, in order to check the potential weaknesses of the current IADC formula and re-orbiting recommendations, focusing on the consequences of having operational orbits characterized by medium or high inclinations.

2. STABILITY VS. INSTABILITY OF NEARLY CIRCULAR GEOSYNCHRONOUS ORBITS

The orbital plane and eccentricity evolution of nearly circular geosynchronous orbits have been deeply investigated over many years, and a good recent review of some of the relevant literature can be found in [5] and [6]. The possibility of eccentricity instability for some inclinations was already pointed out back in the 1960's [7][8].

A mechanism, discovered for artificial satellites by Lidov in 1961 [9], and for asteroids by Kozai in 1962 [10], can strongly affect the eccentricity of objects with a high inclination with respect to the generalized Laplace plane [11]. Driven by the conservation of the orbit angular momentum component normal to the generalized Laplace plane, it consists in an exchange of eccentricity with inclination, leading to large-amplitude inclination and eccentricity oscillations accompanied by librations of the perigee around either 90° or 270°. Therefore, due to the Kozai-Lidov effect, the eccentricity of high inclination near geosynchronous orbits might grow significantly, irrespective of its initial value [12].

However, the picture is actually much more complex than that derived from the classical Kozai-Lidov mechanism, due to fact that luni-solar perturbations also give rise to resonances dependent only on inclination, and others dependent on semi-major axis, eccentricity and inclination [13][14][15]. The resonances depending only on inclination occur when there is commensurability between the secular rates of change of the right ascension of the ascending node $\dot{\Omega}$ and of the argument of perigee $\dot{\omega}$ [13][16][17]. The main inclination resonance conditions for prograde orbits (i.e. with $i \leq 90°$) are summarized in Tab. 1.

Table 1

Inclination resonance conditions for Earth's oblateness (J_2) and luni-solar perturbations ($\dot{\Omega}/\dot{\omega} \approx k$)

Value of *k*	**Resonant term**	**Inclination (°) prograde solution**
$-\infty$	$\dot{\omega} \approx 0$	63.44
-2	$2\dot{\omega} + \dot{\Omega} \approx 0$	56.06
-1	$\dot{\omega} + \dot{\Omega} \approx 0$	46.38
0	$\dot{\Omega} \approx 0$	90
1	$\dot{\omega} - \dot{\Omega} \approx 0$	73.15
2	$2\dot{\omega} - \dot{\Omega} \approx 0$	69.01
∞	$\dot{\omega} \approx 0$	63.44

If the initial inclination i_0 of nearly circular and geosynchronous orbits is around 30° or less, these resonances are not met during the long-term orbit plane evolution and the Kozai-Lidov mechanism does not affect the orbit behavior. Therefore, the eccentricity displays periodic variations with amplitude depending on the initial conditions, but remains in any case bounded for a very long time, with no significant systematic growth.

For inclinations $i_0 > 30°$, certain combinations of initial conditions lead, instead, to a significant eccentricity growth, often with a complex interplay between resonances depending only on inclination, resonances depending on semi-major axis, eccentricity and inclination [18], and, when applicable, the Kozai-Lidov mechanism. However, for the purposes of the present work, what must be remarked is that for nearly circular and geosynchronous orbits with $i_0 \leq 30°$ it is always possible to avoid a significant eccentricity growth over 200 years, irrespective of the other initial conditions (i.e. epoch, right ascension of the ascending node Ω and argument of perigee ω), while with $i_0 > 30°$ only specific initial conditions would be able to guarantee a relative stability of the eccentricity.

3. A POSSIBLE EXTENSION OF THE IADC FORMULA

In addition to the initial inclination, the eccentricity evolution of nearly geosynchronous orbits also depends on the relative positions of the Sun and the Moon. The Moon motion, in particular, is quite complex. With respect to the Earth equator and equinox reference coordinate system, the Moon argument of periapsis has a precession period of 8.85 years, while the right ascension of the ascending node oscillates about 0°, from approximately $-13°$ to $+13°$, with a period of 18.6 years. With the same period, the Moon inclination with respect to the Earth equator varies from 18° to 29° [19].

In order to investigate the eccentricity stability of nearly circular and geosynchronous orbits, the full range of initial conditions concerning the Sun, the Moon, the right ascension of the ascending node and the argument of perigee was explored using the STELA (version 3.1.1) and SATRAP (version 2.1) orbit propagators. The Semi-analytic Tool for End of Life Analysis (STELA) was designed by the Centre National d'Études Spatiales (CNES) to support the French Space Act and allows, among other things, efficient and accurate long-term propagations of nearly geosynchronous orbits [20]. The SATellite Reentry Analysis Program (SATRAP) is instead a numerical code for the trajectory propagation around the Earth, including all the major perturbations [21].

In addition to the full range of orbital and celestial initial conditions, the IADC formula (Eq. 1, with $h_0 = 235$ km) and any possible extension were tested over a time interval of 200 years assuming $e_0 = 0.003$, i.e. the maximum initial eccentricity recommended by the IADC, and $C_R A/M = 0.1$ m^2/kg, i.e. a quite conservative value, compared with those found for current spacecraft and upper stages, where $C_R A/M < 0.05$ m^2/kg. With such assumptions, it was found that the IADC guideline, i.e. Eq. 1 with $h_0 = 235$ km, would be able to guarantee no further interference with the geosynchronous protected region over 200 years, irrespective of initial Sun and Moon position, ascending node, and argument of perigee, only for initial disposal inclinations $i_0 \leq 2°$. In other words, with $e_0 = 0.003$, $C_R A/M = 0.1$ m^2/kg and $i_0 \leq 2°$, the IADC formula would be able to guarantee a perigee altitude ≥ 200 km above the geostationary height of the disposal orbit, over 200 years, for any initial right ascension of the ascending node, argument of perigee, day of the year and phase of the Moon orbit plane cycle.

With higher initial disposal inclinations, this would be no longer guaranteed, even though the applicability of the IADC formula might be extended up to $i_0 \leq 10°$, for any initial condition, if sporadic crossings of the geosynchronous protected region by ~ 10 km were deemed acceptable. Concerning still higher initial inclinations, up to $i_0 = 30°$, where the amplitude of the eccentricity oscillations progressively increases, but remains anyway upper bounded, a relationship with the form of Eq. 1 might be maintained by just appropriately increasing the value of h_0, as detailed in Tab. 2. With $i_0 > 30°$, for the reasons discussed in the previous section, the eccentricity may grow to quite large values, for certain initial conditions, due to resonances and/or the Kozai-Lidov mechanism. Consequently, it is not possible to define end-of-life disposal strategies being simple, universal and inexpensive at the same time, at least with current technologies.

Table 2
Modification of the parameter h_0 in the IADC formula, as a function of the initial disposal inclination i_0, to guarantee no further interference with the geo-synchronous protected region over 200 years, irrespective of initial Sun and Moon position, ascending node, and argument of perigee

h_0 (km)	i_0
235	$\leq 2°$
285	$\leq 18°$
300	$\leq 20°$
350	$\leq 26°$
400	$\leq 28°$
500	$\leq 29°$
550	$\leq 30°$

It should be remarked that the results obtained, summarized in Tab. 2, do not mean that much less expensive solutions, in terms of velocity variation (ΔV) or propellant consumption, do not exist. On the contrary, for many sets of initial conditions, quite smaller values of h_0 would be able to avoid the long-term crossing of the geosynchronous protected region for a given value of i_0. However, this should be checked with a case by case analysis, being not a general result valid for any possible combination of initial conditions. Moreover, when a geosynchronous mission ends, some initial disposal conditions may be adjusted and optimized, as the eccentricity and the argument of perigee, but changing the inclination and the right ascension of the ascending node would be too expensive and, of course, nothing could be made to change the Sun and Moon position.

In general, the worst initial conditions, i.e. those needing values of h_0 close to those listed in Tab. 2, occur around the spring equinox, with the perigee of the disposal orbit lying on the equatorial plane opposite to the Sun, i.e. with $\Omega_0 \approx 180°$ and $\omega_0 \approx 0°$. The additional Moon contribution depends on the initial inclination i_0. If i_0 is small, the worst case is found when the right ascension of the Moon ascending node is maximum (i.e. $\approx 13°$). When i_0 assumes intermediate values (10°-20°), the worst initial conditions correspond to a right ascension of the Moon ascending node around 0° and to the maximum inclination with respect to the Earth equator (i.e. $\approx 29°$). Finally, for $i_0 \approx 26°$-$30°$, the worst initial conditions correspond again to a right ascension of the Moon ascending node around 0°, but with the minimum inclination with respect to the Earth equator (i.e. $\approx 18°$).

The extreme values of Tab. 2, i.e. the first three and the last two ones, can be accurately fitted (Fig. 1) with the following equation, where h_0 is expressed in km and i_0 in degrees:

$$h_0 = 5.11 \times 10^{-5}\,(i_0 - 2°)^5 - 1.71 \times 10^{-3}\,(i_0 - 2°)^4 + \\ + 1.65 \times 10^{-2}\,(i_0 - 2°)^3 + 0.16\,(i_0 - 2°)^2 \qquad (3)$$

Eq. 3 might be therefore used to generalize Eq. 1 up to $i_0 = 30°$, even though for $i_0 \approx 26°$-$28°$ would lead to h_0 values significantly higher than those found and listed in Tab. 2.

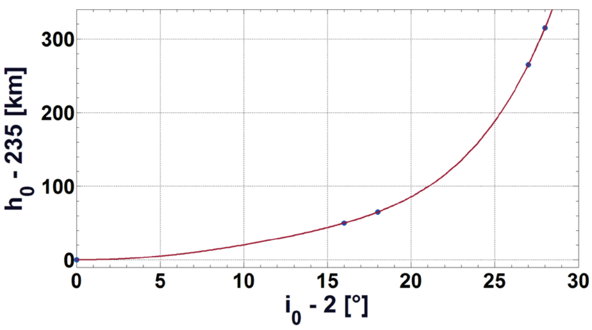

Figure 1. Plot of Eq. 3, fitting the first three and the last two values of Tab. 2

Regarding the additional ΔV needed to implement the extended IADC formula, it can be evaluated as follows. The end-of-life re-orbiting according to Eqs. 1 and 2, from a near circular geosynchronous orbit, leads to a semi-major axis increase Δa given by:

$$\Delta a = (1+e_0)[h_0 + (1000 \cdot C_R \cdot A/M)] + e_0 a_{\mathrm{GEO}} \qquad (4)$$

where a_{GEO} is the geostationary semi-major axis (i.e. 42 164.14 km). Compared with a re-orbiting carried out using the IADC formula, where $h_0 = 235$ km, the additional semi-major axis increase Δa_+ entailed by the application of the extended formula is given by:

$$\Delta a_+[\mathrm{km}] = (1+e_0)(h_0 - 235) \approx h_0 - 235 \qquad (5)$$

Being applicable, for $\Delta a \ll a$ and nearly circular orbits, the following expression, where n represents the mean motion:

$$\Delta a = \frac{2}{n}\Delta V \qquad (6)$$

the additional velocity variation ΔV_+ to be applied for implementing the extended IADC formula, compared with the original one, can be expressed as (Fig. 2):

$$\Delta V_+[\mathrm{km/s}] \approx 3.646 \times 10^{-5}(h_0[\mathrm{km}] - 235) \qquad (7)$$

For $h_0 = 550$ km, the maximum value given in Tab. 2 and applicable for $i_0 \leq 30°$, this would imply $\Delta V_+ = 11.5$ m/s, i.e. an additional velocity variation equivalent to that needed to re-orbit, with $e_0 = 0$, a spacecraft with $C_R\,A/M = 0.08$ m^2/kg, using the original IADC formula with $h_0 = 235$ km.

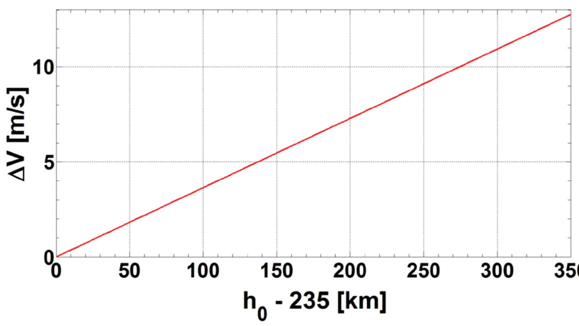

Figure 2. Additional ΔV, as a function of h_0, to be delivered compared with the original IADC formula

Therefore, even assuming the most unfavorable initial conditions, an extension of the IADC formula following either Tab. 2 or Eq. 3 might be practicable up to $i_0 = 30°$, with a maximum ΔV penalty of 11.5 m/s. This, of course, would have a not negligible cost, but could be economically affordable and technically feasible. Moreover, as already pointed out, much less expensive solutions, in terms of additional ΔV, would be often available for appropriate combinations of orbital and celestial initial conditions.

4. VERY HIGH INITIAL INCLINATIONS

As explained in Section 2, when $i_0 > 30°$ a significant growth of eccentricity cannot be avoided without a careful selection of the initial conditions. A simple and universal end-of-life re-orbit strategy, like that represented by Eqs. 1 and 2 with a varying value of h_0, is therefore no longer applicable.

Considering, for instance, the IGSO component of the Chinese Beidou navigation system ($i = 53°$-$58°$), the long-term eccentricity behavior is extremely sensitive to the initial conditions, in particular the right ascension of the ascending node Ω_0, for an assigned epoch [5]. Taking, as an example, the Beidou IGSO-1 spacecraft, assuming no further orbital control since 7 June 2013, with $\Omega_0 = 0°$ would have been possible to maintain a small eccentricity for nearly three centuries, for just one century with $\Omega_0 = 30°$, but for less than 25 years with $\Omega_0 = 120°$ or $\Omega_0 = 225°$ [5]. On the other hand, selecting the initial ascending node in the intervals $90° \leq \Omega_0 \leq 180°$ or $210° \leq \Omega_0 \leq 250°$ would have led to a growth of the eccentricity above 0.83, causing the reentry of the satellite in the atmosphere after intervals of time from several decades to a few centuries [5].

These examples clearly show that a case by case analysis would be needed to determine the outcome of the orbit long-term evolution when the initial inclinations are so large. Ignoring the possibility to reduce the initial disposal inclination below 30°, too expensive and unfeasible in practice, certain initial conditions might be compatible with the re-orbiting in a super-synchronous nearly circular orbit maintaining a small eccentricity over 200 years. In other cases, the initial ascending nodes might lead, after several decades or a few centuries, to large eccentricities and the consequent spacecraft decay. However, in general, the change of the right ascension of the ascending node would be too much expensive and unfeasible as an inclination reduction, and having a satellite reentering on the Earth after crossing the geosynchronous protected region for several decades or a few centuries would not be much different, in practice, from a non reentering object crossing the protected region as well. Even choosing an appropriate launch time, as proposed in [5] to obtain the desired disposal conditions, might conflict with mission constraints and operations, in exchange of uncertain potential benefits.

In principle, at least for very high inclination orbits, the Kozai-Lidov effect might offer a very elegant and effective disposal mechanism. In fact, as recalled in Section 2, the progressive amplitude growth of the eccentricity oscillations is accompanied by librations of the perigee around either 90° or 270°. This means that the crossings of the geosynchronous altitude would occur far from the nodes, i.e. considerably above or below the equatorial plane and the geostationary ring. However, this beautiful "protection mechanism" derives from the hierarchical and restricted three-body Kozai-Lidov formalism. In the real world, adding to the Sun the Earth's oblateness, the Moon, other relevant perturbations as direct solar radiation pressure, and the intricate web of luni-solar resonances, the simple pattern of the classical Kozai-Lidov mechanism is strongly affected and, during certain phases of the orbit evolution, the geosynchronous altitude can be crossed also near the equatorial plane, depending on the initial conditions.

Examples of this behavior are provided by a couple of long-term propagations, carried out with SATRAP, taking into account all relevant perturbations, including atmospheric drag for perigee heights below 1000 km. A fictitious passive satellite with $C_R\,A/M = 0.05$ m^2/kg, $a_0 = a_{\mathrm{GEO}}$ and $i_0 = 75°$ was propagated for more than 200 years, starting on 20 March 2019, with two sets of initial conditions:

1. Assuming $e_0 = 0.0001$, $\Omega_0 = 180°$, $\omega_0 = 0°$;
2. Assuming $e_0 = 0.001$, $\Omega_0 = 0°$, $\omega_0 = 0°$.

The results obtained in the first case are summarized in Figs. 3 and 4, those obtained in the second one in Figs. 5 and 6. In the first case, the inclination increases up to 84° and the eccentricity exceeds 0.7 (Fig. 3), but the satellite is still in orbit after 220 years. As in the classical Kozai-Lidov effect, the eccentricity growth and its successive large amplitude oscillation is accompanied by a libration of the perigee around a "stable" value (Fig. 4). However, such a value is neither 90° nor 270°, but $\sim 0°$. This means that when the eccentricity is large, the perigee is close to the node, i.e. to the equatorial plane, and the Kozai-Lidov mechanism is therefore not able to keep safe the geosynchronous protected region.

Looking at the second case, the eccentricity growth is so large to induce satellite decay after 128 years (Fig. 5). Concerning the coupled evolution of eccentricity and argument of perigee (Fig. 6), again the simple pattern of the classical Kozai-Lidov effect is disrupted, with no libration around a specific value of the argument of perigee. Nevertheless, during the first 100 years, when the perigee is close to the node, and to the equatorial plane, the eccentricity is small, while when the eccentricity is maximum (~ 0.8), the perigee is above the equator at the greatest distance as well, so a dynamical protection mechanism still plays a role (Fig. 6). Such protection mechanism is lost in the following orbit evolution, when the eccentricity goes down to 0.35

and then rebounds to 0.84 and the consequent decay, but this flight phase is too short (< 30 years) to involve significant interactions with the geosynchronous protected region.

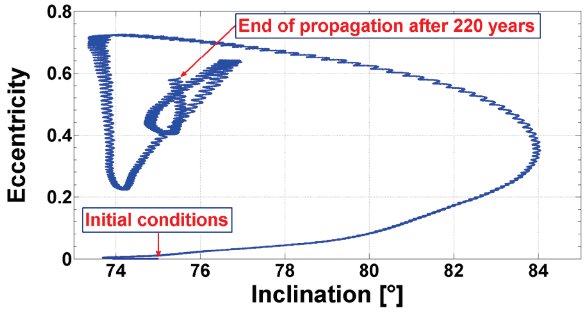

Figure 3. Eccentricity vs. inclination for the first case with $i_0 = 75°$

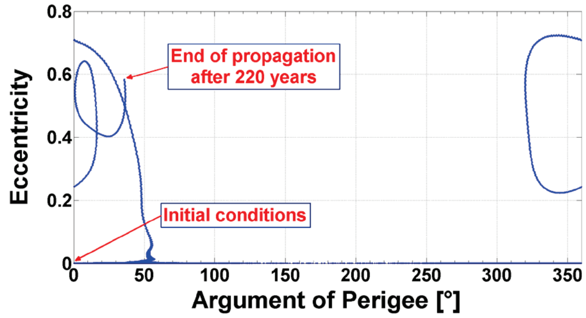

Figure 4. Eccentricity as a function of the argument of perigee for the first case with $i_0 = 75°$

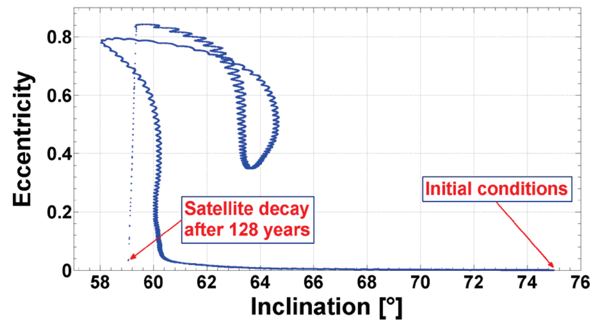

Figure 5. Eccentricity vs. inclination for the second case with $i_0 = 75°$

It is therefore quite evident that no simple, universal and effective end-of-life disposal guideline can be devised for the re-orbiting of near circular geosynchronous objects with $i_0 > 30°$. On the other hand, if the total number of these objects will remain constrained to a few thousand during the next century, even leaving them totally uncontrolled at the end-of-life would not increase significantly the collision probability in the geostationary ring and in the geosynchronous protected region, because they would be spread in a huge volume of space by the perturbations, spending only occasional and relatively short periods of time in the most populated geosynchronous belt. However, an end-of-life re-orbiting above the geosynchronous altitude of a few hundred km might be in any case beneficial, even with an unbounded eccentricity, for further reducing the average long-term collision probability with objects residing in the geosynchronous protected region.

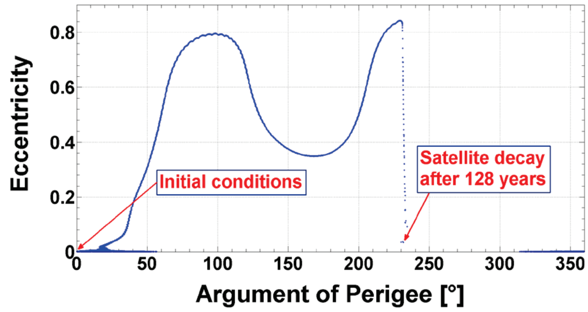

Figure 6. Eccentricity as a function of the argument of perigee for the second case with $i_0 = 75°$

5. CONCLUSIONS

The original IADC guideline for the end-of-life re-orbiting of spacecraft and orbital stages above the geosynchronous altitude, summarized by Eqs. 1 and 2, with $h_0 = 235$ km, was devised for objects close to the geostationary ring and resulted strictly valid, irrespective of the orbital and celestial initial conditions, for $i_0 \leq 2°$, and applicable in most cases for $i_0 \leq 10°$.

The potential growing use of geosynchronous orbits with higher inclinations has raised the problem of what strategy to adopt at the end-of-life for these new classes of objects. For $i_0 \leq 30°$, it was shown that an effective extension of the IADC formula would be possible by just increasing the value of h_0 as a function of i_0, with a maximum of 550 km for $i_0 = 30°$. This possibility arises from the fact that, despite the complexity of the perturbations acting on such orbits, the eccentricity would remain bounded to sufficiently low values for at least 200 years, irrespective of the orbital and celestial initial conditions. With $i_0 = 30°$ and $h_0 = 550$ km, no crossing of the geosynchronous protected region would occur, over 200 years, even in the less favorable combinations of orbital (Ω_0 and ω_0) and celestial (Sun and Moon position) initial conditions. It should be however pointed out that very often it would be possible to avoid any further long-term interference with the protected region adopting significantly lower values of h_0, being those listed in Tab. 2, or computed with Eq. 3, just the values able to guarantee the fulfillment of the guideline goal even with the worst initial conditions.

The cost of implementing the extended IADC formula, in terms of additional ΔV to be delivered by the propulsion system and mission impact, would not be negligible, but affordable. With $i_0 = 30°$, the maximum additional ΔV penalty would be 11.5 m/s, approximately doubling the cost currently incurred to apply the original IADC formula to nearly geostationary satellites.

Regarding the present geosynchronous satellite systems operational at high inclination, the Indian IRNSS ($i = 28°$-$30°$) and the NASA's Solar Dynamics Observatory ($i = 29°$) are the obvious candidates for the implementation of the extended IADC formula at the end-of-life. In other words, depending on the initial conditions, for each satellite to be disposed above the geosynchronous altitude, an appropriate value of $h_0 \leq 550$ km might be found to guarantee no further interference with the geosynchronous protected region for more than 200 years.

This approach will not be instead generally applicable for $i_0 > 30°$, i.e. for the Chinese Beidou ($i = 53°$-$58°$) and the Japanese QZSS ($i = 40°$-$45°$), just to mention the couple of systems currently operational. Reducing the initial inclination and/or changing the initial right ascension of the ascending node of the disposal orbit would be of course too much expensive and unfeasible, not to mention the Moon and Sun positions. Even choosing appropriate launch times, provided it were feasible from a mission point of view, might not be so beneficial, being quite tricky the accurate prediction of end-of-life epochs several years in advance.

The problem should be therefore addressed on a case by case basis, strongly dependent as it is from the initial orbital and celestial conditions. In certain cases, it should be possible to constrain the eccentricity growth from several decades to a few centuries, in others the eccentricity would grow to values so large to cause orbital decay in several decades. For very high inclinations and appropriate initial conditions, the interaction between the Kozai-Lidov effect and the other perturbations might lead to the emergence of a dynamical protection mechanism of the geosynchronous protected region, effective at least for several decades.

As a consequence of this situation, no easy to apply, general and cost-effective end-of-life disposal solution can be recommended for the re-orbiting of near circular geosynchronous objects with $i_0 > 30°$. Nevertheless, if the total number of these objects will not grow by more than two orders of magnitude in the next century, even simply abandoning them at the end of their missions would not lead to a significant increase of the collision probability in the geostationary ring and in the geosynchronous protected region, because they would be spread in a huge volume of space by the perturbations, spending only occasional and relatively short periods of time in the most populated geosynchronous belt. However, in order to further reduce the average long-term collision probability with objects residing in the geosynchronous protected region, an end-of-life re-orbiting above the geosynchronous altitude of a few hundred km might be in any case beneficial, even with an unbounded eccentricity.

6. ACKNOWLEDGMENTS

The work described in this paper was carried out in the framework of the Agreement No. 2015-028-R.0 between the Italian Space Agency (ASI) and the National Institute of Astrophysics (INAF) on "Space Debris: Support to IADC Activities and Pre-Operational Validation for SST".

7. REFERENCES

1. Steering Group & Working Group 4 (2004; 2007). Space Debris Mitigation Guidelines, Document IADC-02-01 & Revision 1, Inter-Agency Space Debris Coordination Committee (IADC).
2. Anselmo, L. & Pardini, C. (2008). Space Debris Mitigation in Geosynchronous Orbit. Advances in Space Research 41(7), 1091–1099.
3. Beutler, G. (2005). Methods of Celestial Mechanics, Volume II: Application to Planetary System, Geodynamics and Satellite Geodesy, Springer, Berlin, Germany, pp153–155.
4. Lemmens, S. (Ed.) (2017). Classification of Geosynchronous Objects, Issue 19, Revision 1, ESA's Space Debris Office, European Space Agency, Darmstadt, Germany.
5. Zhao, C.-Y., Zhang, M.-J., Wang, H.-B., Xiong, J.-N., Zhu, T.-L. & Zhang, W. (2015). Analysis on the Long-term Dynamical Evolution of the Inclined Geosynchronous Orbits in the Chinese BeiDou Navigation System. Advances in Space Research 56(3), 377–387.
6. Zhu, T.-L., Zhao, C.-Y. & Zhang, M.-J. (2017). Analytic Model for the Long-term Evolution of Circular Earth Satellite Orbits Including Lunar Node Regression. arXiv:1703.00655v1.
7. Allan, R.R. & Cook, G.E. (1964). The Long-period Motion of the Plane of a Distant Circular Orbit. Proceedings of the Royal Society A: Mathematical, Physical and Engineering Sciences 280(1380), 97–109.
8. Breiter, S. (2001). Lunisolar Resonances Revisited. Celestial Mechanics and Dynamical Astronomy 81(1-2), 81–91.
9. Lidov, M.L. (1961) Evolution of Artificial Planetary Satellites under the Action of Gravitational Perturbations Due to External Bodies. Iskusstviennye Sputniki Zemli (in Russian) 8, 5–45.
10. Kozai, Y. (1962) Secular Perturbations of Asteroids with High Inclination and Eccentricity.

The Astronomical Journal 67(9), 591–598.
11. Tamayo, D., Burns, J.A., Hamilton, D.P. & Nicholson, P.D. (2013). Dynamical Instabilities in High-obliquity Systems. The Astronomical Journal 145(54), 12pp.
12. Wytrzyszczak, I., Breiter, S. & Borczyk, W. (2007). Regular and Chaotic Motion of High Altitude Satellites. Advances in Space Research 40(1), 134–142.
13. Cook, G.E. (1962). Luni-solar Perturbations of the Orbit of an Earth Satellite. Geophys. J.R. Astron. Soc. 6, 271–291.
14. Hughes, S. (1980). Earth Satellite Orbits with Resonant Lunisolar Perturbations. I. Resonances Dependent Only on Inclination. Proc. R. Soc. Lond. A 372, 243–264.
15. Hughes, S. (1981). Earth Satellite Orbits with Resonant Lunisolar Perturbations. II. Resonances Dependent on the Semi-major Axis, Eccentricity and Inclination. Proc. R. Soc. Lond. A 375, 379–396.
16. Yokoyama, T. (1999). Dynamics of Some Fictitious Satellites of Venus and Mars. Planetary and Space Science 47, 619–627.
17. Anselmo, L. & Pardini, C. (2011). Orbital Evolution of the First Upper Stages Used for the New European and Chinese Navigation Satellite Systems. Acta Astron. 68, 2066–2079.
18. Bordovitsyna, T.V., Tomilova, I.V. & Chuvashov, I.N. (2014). Secular Resonances as a Source of Dynamic Chaoticity in the Long-term Orbital Evolution of Uncontrolled Satellites. Solar Syst. Res. 48, 259–268.
19. Roncoli, R.R. (2005). Lunar Constants and Models Document. Report JPL D-32296, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California, USA.
20. Anonymous (2016). STELA User's Guide, Version 3.1.1. Centre National d'Études Spatiales (CNES), Toulouse, France.
21. Pardini, C. & Anselmo, L. (1994). SATRAP: Satellite Reentry Analysis Program. Internal Report C94-17, CNUCE Institute, Consiglio Nazionale delle Ricerche (CNR), Pisa, Italy.