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Ariane 5 GTO debris mitigation using natural perturbations

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Abstract

GTO objects can potentially collide with operative satellites in LEO and GEO protected regions. Internationally accepted debris mitigation guidelines require that these objects exit these protected regions within 25 years, e.g. by re-entering and burning up in Earth's atmosphere. In this paper, an inventory of the GTO debris generated from Ariane 5 launches in the period 2012–2017 is provided, and it is expected that none of these objects will re-enter within 25 years. For future launches, natural perturbations can be exploited to increase compliance with mitigation guidelines without the use of extra propellant or complex de-orbiting systems, which is attractive from an economic point of view. The lifetime of GTO objects is very sensitive to initial conditions and some environmental and body-related parameters, mainly due to the effect of solar gravity on the perigee altitude. As a consequence, the lifetime of a specific GTO object cannot be predicted accurately, but its probability of re-entering in less than 25 years can be estimated with proper accuracy by following a statistical approach. By propagating the orbits of over 800 000 simulated Ariane 5 GTO objects, it was found that the launch time leading to the highest probability of compliance with debris mitigation guidelines for GEO launches from Kourou corresponds to about 2 PM local time, regardless of the date of launch, which leads to compliance rates ranging from 60 to 100%. Current practice is to launch at around 5–9 PM, so a change in procedures would be required in order to reach a higher degree of compliance with debris mitigation guidelines, which was predicted to be on average below 20% for the objects generated in the period 2012–2017.

Keywords: Space debris; Geostationary transfer orbit; Debris mitigation; Ariane 5; Orbital perturbations

1. Introduction

Low Earth orbit (LEO) and geostationary orbit (GEO) are two regions of near-Earth space with a relatively high probability of collisions between man-made objects due to their higher density of Earth-orbiting bodies. Since both areas are highly attractive, this has led to the definition of two protected regions around LEO and GEO, as depicted in Fig. 1. Debris mitigation guidelines have been proposed by the Inter-Agency Space Debris Coordination Committee (IADC, 2007) in order to minimise the generation of debris in the future, including measures such as passivation of all stored energy after the end of life to prevent in-orbit explosions and break-ups. Additionally, non-functional objects should not remain in or cross protected regions beyond 25 years after their end of life.

Objects such as depleted rocket stages used during the launch of geostationary satellites typically follow a geostationary transfer orbit (GTO), with perigee altitude below 2 000 km and apogee altitude near geostationary altitude, which crosses both the LEO and GEO protected regions. It can take these objects years or even decades to re-enter in Earth's atmosphere by natural means (Sharma et al., 2004), which increases the risk of collisions and generation of additional debris. A prediction for the year 2200 revealed that the number of total orbital debris larger than 1 mm, 1 cm and 10 cm could be reduced by about 62, 57 and 43%, respectively, only by implementing de-orbiting strategies for all upper stages launched after 2010 with perigee altitude below 2 000 km (Anselmo et al., 1999).

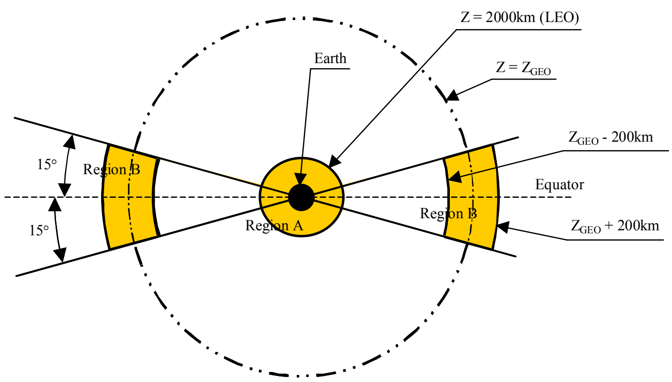

Fig. 1: Graphical definition of the near-Earth protected regions (IADC, 2007). Region A: spherical region extending from Earth's surface up to an altitude of 2 000 km. Region B: segment of a spherical shell extending 200 km below and above the GEO altitude (35 786 km) and with a declination between −15 and 15 degrees.

Previous studies have shown that natural orbital per-

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turbations such as those caused by Earth's irregular gravity field, luni-solar gravity and atmospheric drag can affect the lifetime of GTO objects significantly (Sharma et al., 2004). Siebold and Reynolds (1995) found that the lifetime of GTO objects resulting from launches from Kennedy Space Center (with an initial inclination of 28.5 degrees) depends on the initial perigee altitude, the initial right ascension of the ascending node (RAAN) and the initial position of the Sun. Potentially, the lifetime of GTO debris could be reduced by wisely choosing these parameters, leading to a higher rate of compliance with debris mitigation guidelines. However, the accurate long-term propagation of GTOs poses some challenges, mainly due to the coupling of the effects of different perturbations when the semi-major axis of the orbit is close to 15 000 km, a phenomenon known as solar apsidal resonance; this can result in the lifetime being extremely sensitive to initial orbital conditions and environmental and body-related parameters (Wang and Gurfil, 2017).

Fisher and David (2014) surveyed the GTO debris population generated in the period 2004–2012 resulting from GEO launches and found that 84% of the objects that are not compliant with debris mitigation guidelines and cross both the LEO and GEO protected regions were generated during Ariane 5 launches. In this paper, the orbital evolution of Ariane 5 debris in GTO, namely upper stages and payload carriers, will be studied for different initial conditions, with the aim to determine whether it is possible to choose initial launch conditions such that the rate of compliance with debris mitigation guidelines is increased by exploiting the effects of natural orbital perturbations, without expending additional propellant and with a minimal impact, if any, on mission planning and design.

2. GTO debris from Ariane 5 launches

On average, more than 32 objects were purposely left in GTO every year in the period 2004–2012 as a result of geostationary satellite launches. More than 69% of these objects are non-compliant with debris mitigation guidelines (Fisher and David, 2014), i.e. they are expected to cross the protected regions of near-Earth space for more than 25 years. To put these numbers into perspective: there were only 39 operative satellites in elliptical orbits as of November 2017 (Union of Concerned Scientists, 2017).

Launches carried out with Ariane 5 result in the generation of debris that are purposely left in GTO according to mission planning. The Ariane debris mitigation policy allows for up to one piece of debris per launched satellite (Johnson, 2005). Typically, these are the final stage and, in some cases, additional payload carriers. These objects are some of the least compliant with debris mitigation guidelines: 94% of them are non-compliant, 96% of which cross both the LEO and the GEO protected regions, as can be seen in Fig. 2. This unarguably makes Ariane 5 the major contributor to the GTO debris population, also when expressed in absolute numbers.

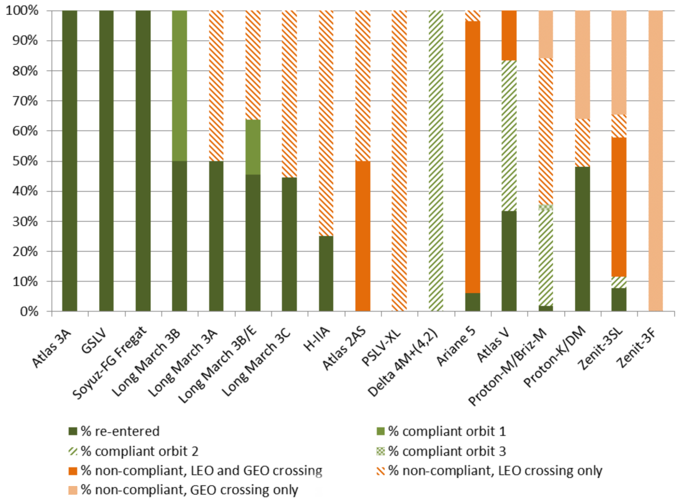

Fig. 2: Orbits of upper stages and mission-related debris from GEO launches in the period 2004–2012 broken down by launch vehicle (Fisher and David, 2014). Compliant orbit 1: perigee below 200 km to ensure re-entry within 25 years; compliant orbit 2: between LEO and GEO protected regions; compliant orbit 3: super-synchronous.

To illustrate how the orbits of Ariane 5 debris evolve over long periods of time, the apogee and perigee altitudes of an Ariane 5 rocket body used during the launch of Spainsat in March 2006 is given in Fig. 3, generated from the two-line elements provided by the Joint Force Space Component Command (JFSCC, 2018). The initial apogee and perigee altitudes of this object were 32 888 and 255 km, respectively, its initial orbital inclination was 4.53 degrees, and the launch took place at 19:33 local time from Kourou (JFSCC, 2018; Wikipedia, 2018). After more than 12 years, its perigee altitude is still above 250 km, which shows that these objects can take very long to re-enter.

An analysis of the satellite-tracking data for the period 2012–2017 provided by the JFSCC (2018) reveals that objects generated during recent launches are likely to show a similar behaviour, given that comparable initial launch conditions hold, leading to the presence of debris in protected regions over long periods of time, as can be seen in Table 1. Although only a small fraction of the generated debris cross the GEO protected region because most have apogee altitudes below the 35 586 km limit, all of them cross the LEO protected region and have perigee altitudes above 200 km, which makes it unlikely for them to comply with debris mitigation guidelines. Previous studies have used the rough limit of 200 km as a rule of thumb to predict which GTO objects are likely to re-enter in less than 25 years (Fisher and David, 2014), although this is insufficient due to the complex dynamics of GTOs. In Subsection 4.2, it will be shown that the probability of re-entry in less than 25 years is not above 72% for any of the 39 objects generated by Ariane 5 launches.

In order to determine whether the initial launch conditions can be chosen such that the lifetime of the resulting GTO debris is minimised, it is convenient to characterise the orbits of these objects by studying past launches. The

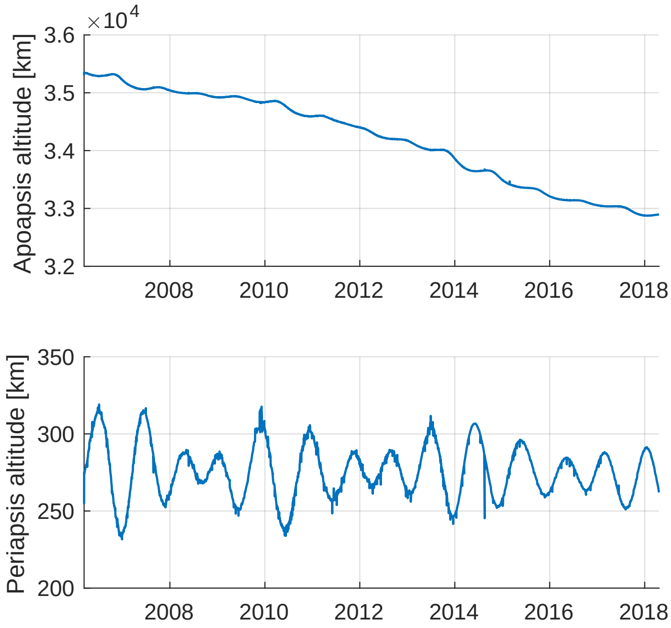

Fig. 3: Time evolution of the apogee and perigee altitudes of the Ariane 5 rocket body used during the launch of Spainsat in March 2006 (JFSCC, 2018).

initial two-line elements of the 39 GTO objects generated in the period 2012–2017 from Ariane 5 launches provided by the JFSCC (2018), as well as the local times of launch (Wikipedia, 2018), were analysed (cf. Fig. 4). It was found that the initial values of most orbital parameters cannot be chosen freely and thus are not prone to being selected as optimisation variables. The initial argument of perigee is always close to 180 degrees (cf. Fig. 4**e**), since it has to be close to either 0 or 180 degrees for the satellite to be within the GEO ring when apogee altitude is reached. The range of possible initial inclinations is bounded by two factors: the need to use near-equatorial transfer orbits, so that the use of propellant is minimum when the payload is injected into geostationary orbit, and the latitude of the European Spaceport in Kourou, i.e. 5.36 degrees. However, in-orbit manoeuvres can decrease the initial inclination of the GTO debris below 5.36 degrees, although most of the objects (20 out of 39) have inclinations above this value (cf. Fig. 4**d**).

Since Ariane 5 carries out a direct ascent to GEO, the apogee altitude is typically close to GEO altitude. Except for one case, all objects had initial apogee altitudes below GEO altitude, with most of them (87%) below the limit of 35 586 km defining the beginning of the GEO protected region. About half of the objects had apogee altitudes between 34 800 km and GEO altitude (cf. Fig. 4**c**). Although the nominal perigee altitude of the transfer orbit used in Ariane 5 launches to GEO is 250 km (Arianespace, 2016; Fisher and David, 2014), the actual values were found to range between 209 and 309 km (cf. Fig. 4**b**).

All launches in the period 2012–2017 took place between 16:54 and 20:45 Kourou local time (cf. Fig. 4**a**), except for one that took place at 2:20. Most of the objects (85%) were generated during launches between 17:30 and 19:30. Despite the fact that the range of launch times is limited, the initial right ascensions of the ascending nodes of the orbits of the GTO debris were widely distributed (cf. Fig. 4**f**), as the orientation of the orbit with respect to inertial space changes throughout the year for a given launch time and location.

Table 1: Debris generated during the launch of geostationary satellites with Ariane 5 in the period 2012–2017. An object is said to be compliant with debris mitigation guidelines if its initial perigee altitude is below 200 km. Based on data from the JFSCC (2018).

Year of launch	**Number of launches**	**Number of debris objects**	**Compliant with guidelines**	**LEO crossing only**	**LEO and GEO crossing**
2012	6	6	0	6	0
2013	3	3	0	3	0
2014	5	5	0	5	0
2015	6	7	0	7	0
2016	6	8	0	8	0
2017	5	10	0	5	5
Total	**31**	**39**	**0**	**34**	**5**

3. Predicting the lifetime of GTO objects

3.1. Relevant perturbations

GTO objects resulting from the launch of geostationary satellites reach the end of their operational life soon after launch, typically some ten minutes later (Arianespace, 2016), but can remain in orbit for years, if not decades. These objects are non-functional, uncontrolled and unpowered during most of their lifespan, and thus they are seldom affected by orbital perturbations other than those of natural origin. Due to their high (initial) eccentricity, these objects undergo different perturbations at different positions in their orbit. When close to perigee (200 to 300 km altitude for Ariane 5 debris), the most relevant perturbations are those caused by Earth's irregular gravity field (mainly the J_2 term) and atmospheric drag (Fortescue et al., 2011). On the other hand, when close to apogee (GEO altitude), the J_2 term and the gravitational attraction of the Sun and the Moon all cause perturbing accelerations of the same order of magnitude, while atmospheric drag becomes negligible and solar radiation pressure (SRP) can become relevant over long periods of time (Armellin et al., 2015).

The relevant perturbations that cannot be neglected when propagating the orbit of GTO objects are low-degree zonal terms of the spherical harmonics expansion of Earth's gravity field –at least J_2–, lunar and solar gravity, and atmospheric drag (Lamy et al., 2011; Morand et al., 2012;

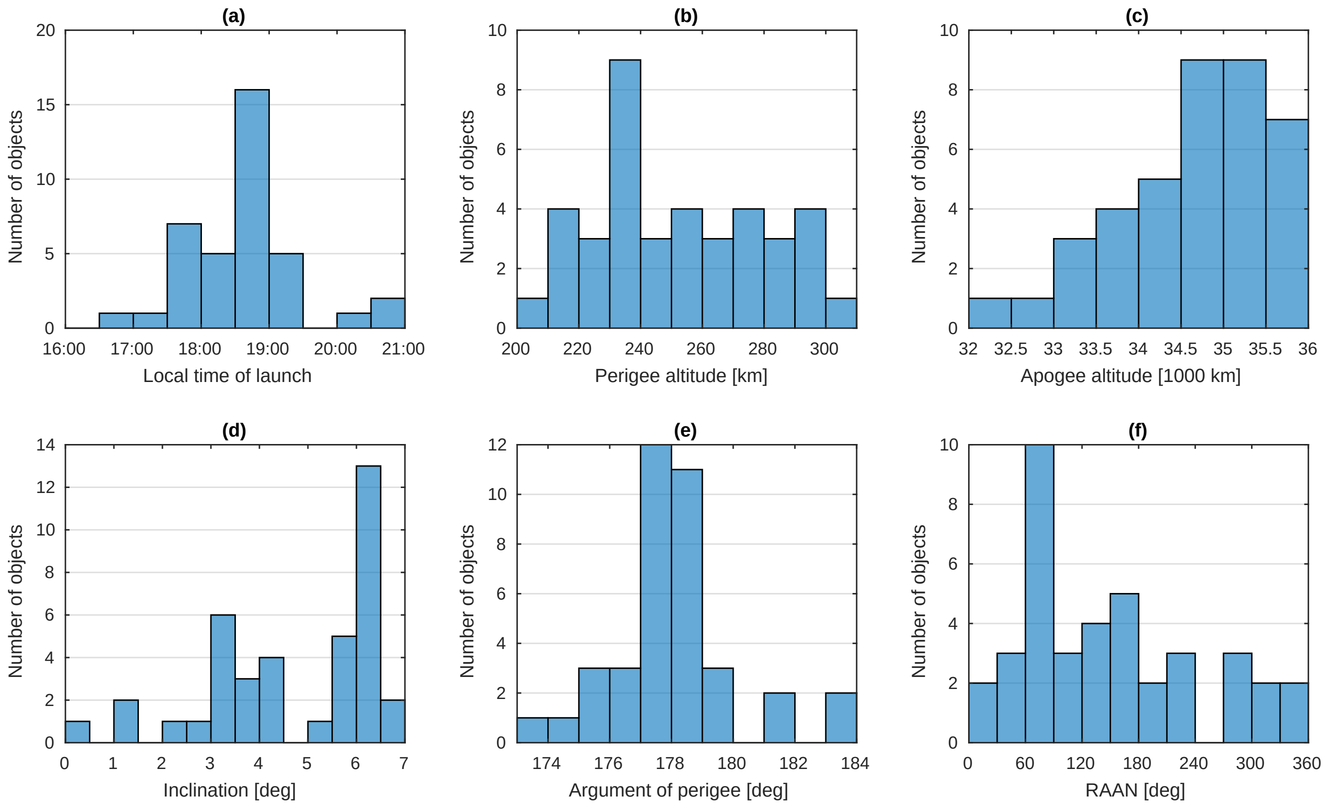

Fig. 4: Characteristics of the debris generated during the launch of geostationary satellites with Ariane 5 in the period 2012–2017: **(a)** Kourou local time of launch (one object omitted), **(b)** initial perigee altitude, **(c)** initial apogee altitude, **(d)** initial orbital inclination, **(e)** initial argument of perigee (one object omitted) and **(f)** initial right ascension of the ascending node.

Wang and Gurfil, 2016, 2017). Additionally, some authors have also included some dedicated tesseral harmonics and solar radiation pressure –optionally accounting for occultations of the Sun by Earth's shadow– in their acceleration models (Morand et al., 2012). In the current paper, zonal terms are included up to degree 7, tesseral terms are neglected and solar radiation pressure is considered but without taking occultations into account. It has been verified that the contribution of tesseral terms and occultations is negligible for GTO lifetime assessments.

3.2. Solar apsidal resonance

A very important aspect of estimating the lifetime of GTO objects is a coupling between orbital perturbations, namely those caused by J_2 and solar gravity, which, in combination with the effects of atmospheric drag, can lead to very different lifetimes for slightly different initial conditions (Sharma et al., 2004). An exhaustive study of this phenomenon, known as solar apsidal resonance, was recently carried out by Wang and Gurfil (2017).

The only perturbation causing secular variations on the semi-major axis of GTOs is atmospheric drag. Thus, the initial perigee altitude should be expected to be the dominant factor determining the lifetime of GTO objects. The Sun causes long-period variations on the value of the eccentricity, and thus on the perigee altitude. The mean rate of change of the perigee altitude over one orbital revolution due only to solar gravity, $\Delta_{2\pi}h_p$, is given by the expression derived from Morand et al. (2012):

$$\Delta_{2\pi}h_p = \frac{15}{2}\pi\frac{\mu_S}{\mu_E}\left(\frac{a}{r_S}\right)^3 ae\sqrt{1-e^2}\cos^2\lambda\sin 2\Lambda \quad (1)$$

with μ_S and μ_E the Sun and Earth standard gravitational parameters, respectively, a the orbit semi-major axis, r_S the distance to the Sun, e the orbit eccentricity, and λ and Λ the Sun declination and azimuth angles, as defined in Fig. 5.

The main perturbation causing variations in the value of the RAAN, Ω, and the argument of perigee ω, is J_2. The mean drift of the perigee due to J_2 is defined as (Vinti, 1998):

$$\dot{\Omega} + \dot{\omega} = \widetilde{J}_2 n\left[-\cos i + \frac{1}{2}\left(5\cos^2 i - 1\right)\right] \quad (2)$$

with n the mean motion, i the orbital inclination, and

$$\widetilde{J}_2 = \frac{3}{2}\frac{J_2 R_E^2}{p^2} \quad (3)$$

with $J_2 = 1.0826357 \times 10^{-3}$ (Tapley et al., 2005), R_E Earth's reference radius, and $p = a(1-e^2)$ the orbit semi-latus rectum.

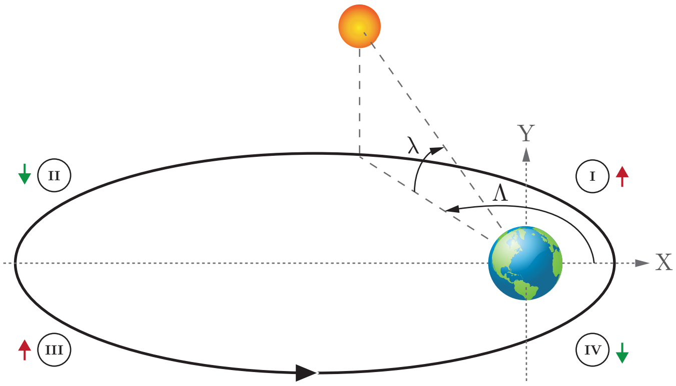

Fig. 5: Definition of the Earth-centred perifocal reference frame, with the X-axis towards the perigee and the Z-axis in the direction of the orbital angular momentum. The relevant angles for modelling the effects of solar gravity on the perigee altitude are: the Sun azimuth angle Λ, measured in the orbital plane from the X-axis to the projection of the Earth-Sun line, and the Sun declination angle λ between the orbital plane and the Earth-Sun line. The Sun causes the perigee altitude to increase when it is in quadrants I or III and to decrease when it is in quadrants II or IV.

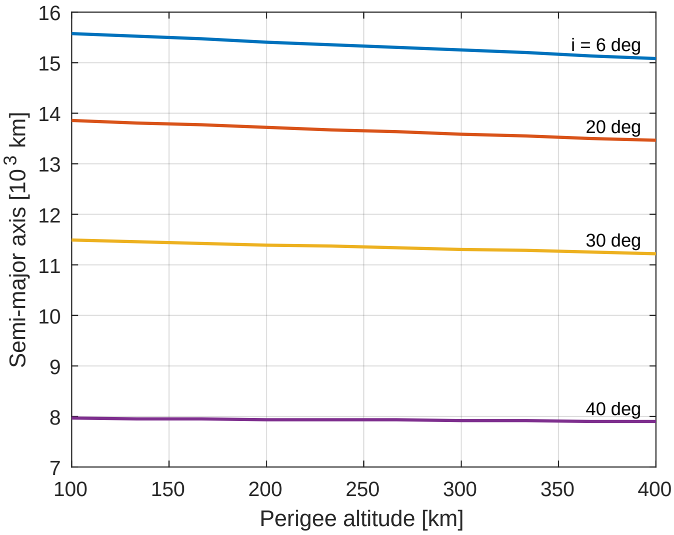

Fig. 6: Combinations of perigee altitudes and semi-major axes for which the mean drift of the perigee due to J_2 coincides with Earth's mean motion for different orbital inclinations, potentially triggering solar apsidal resonances.

When the value of the mean drift of the perigee gets close to Earth's mean motion, i.e. about 0.986 degrees per day, the Sun azimuth angle remains constant for long periods of time, causing quasi-secular variations (i.e. perturbations with periods of several years) on the perigee altitude according to Eq. (1), which can lead the GTO object to undergo very different drag accelerations. For a GTO with a perigee altitude of 250 km and an inclination of 6 degrees, the resonance condition is met when the semi-major axis becomes roughly 15 300 km, as can be seen in Fig. 6. Resonance cannot happen for GTOs with inclinations above 43 degrees, since the resonance conditions would only be met after the object has re-entered in Earth's atmosphere. For inclinations above 46.4 degrees, resonance conditions are never met, as the term between square brackets in the right-hand side of Eq. (2) becomes negative.

The apsidal solar resonance does not only have the potential to lead to very different lifetimes for slightly different initial conditions, but can also be the cause of counterintuitive effects on the orbital evolution. For instance, an increase of atmospheric drag (by considering larger atmospheric densities, area-to-mass ratios and/or drag coefficients) can lead to longer lifetimes, as observed in Fig. 7. This is due to the fact that an increase in drag causes the semi-major axis to decrease initially at a faster rate, and as a consequence the solar apsidal resonance conditions are reached at a different epoch for which the orbit-Earth-Sun geometry is different. Thus, claiming that a GTO object will re-enter with certainty in less than e.g. 25 years, based on simulations showing that for the most unfavourable conditions of low atmospheric drag it would do so in 25 years, is misleading.

3.3. Semi-analytical propagation

The lifetime of GTO objects is very sensitive to several factors that cannot be known with sufficient accuracy, such as the solar activity levels and instantaneous cross-sectional area throughout their lifespan, and some initial orbital conditions, especially the initial perigee altitude and RAAN. Morand et al. (2012) and Sharma et al. (2004) found that a small deviation in the values of these parameters, which would have negligible effects for the propagation of LEO and GEO spacecraft, can change the lifetime of GTO objects by an order of magnitude in some cases, mainly due to the presence or absence of solar apsidal resonance. Thus, they decided to follow a statistical approach when predicting the lifetime of GTO objects, in which many orbits with slightly different (initial) conditions are propagated. In this way, it is possible to estimate the probability of complying with debris mitigation guidelines, i.e. re-entering in Earth's atmosphere in less than 25 years.

Since the orbits of GTO objects may have to be propagated for several years (decades in some cases), the statistical approach becomes computationally unfeasible when a fully-numerical propagator (such as Cowell or Encke) is used. In order to obtain the results presented in this paper, a propagator based on the semi-analytical satellite theory described by Danielson et al. (1999) has been used. This semi-analytical propagator uses equinoctial elements, which are non-singular for GTOs. By averaging the effects of orbital perturbations over one orbital revolution, it is possible to integrate the mean elements of the GTO instead of its osculating elements, allowing the use of much larger integration step-sizes (in the order of one day). Recently, Wang and Gurfil (2016) used Milankovitch elements for the long-term propagation of GTOs, with similar

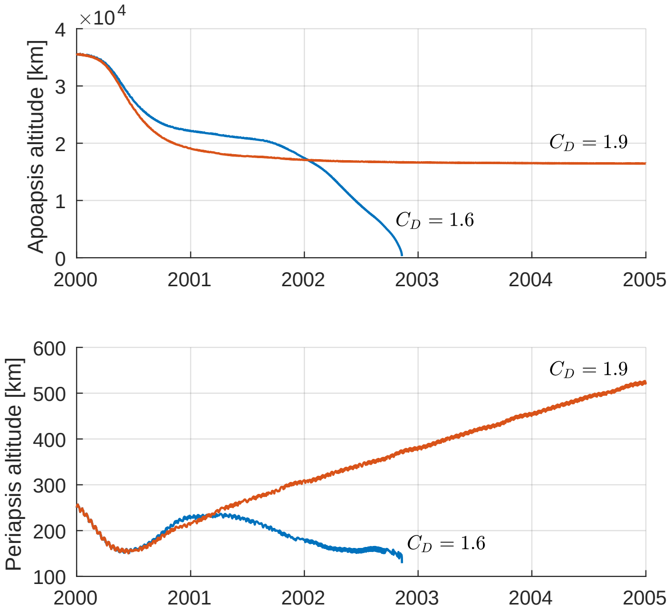

Fig. 7: Time evolution of two objects in a GTO with identical initial conditions, except for the drag coefficient. Note that, counterintuitively, an increase of drag can lead to a longer lifetime.

advantages.

The effects caused by zonal terms of Earth's geopotential, luni-solar gravity and SRP (neglecting Solar occultations) can all be averaged analytically over one orbital revolution using the expressions provided by Danielson et al. (1999). Thus, the only relevant perturbation that cannot be averaged analytically is atmospheric drag. It was found that most of the computation time was spent solving the averaging integrals of atmospheric drag:

$$A_i = \frac{\mathrm{d}a_i}{\mathrm{d}t} = \frac{1}{2\pi\sqrt{1-e^2}} \int_{L_1}^{L_2} \left(\frac{r}{a}\right)^2 \left(\frac{\partial a_i}{\partial \dot{\boldsymbol{r}}} \cdot \boldsymbol{q}\right) \mathrm{d}L \tag{4}$$

where

- A_i are the mean element rates due to atmospheric drag;
- a_i are the mean equinoctial elements:

$$a_i = \begin{pmatrix} a_1 \\ a_2 \\ a_3 \\ a_4 \\ a_5 \\ a_6 \end{pmatrix} = \begin{pmatrix} a \\ e\sin(\omega+\Omega) \\ e\cos(\omega+\Omega) \\ \tan i/2 \,\sin\Omega \\ \tan i/2 \,\cos\Omega \\ M+\omega+\Omega \end{pmatrix} \tag{5}$$

  with M the mean anomaly;
- $L = f + \omega + \Omega$ is the true longitude, with f the true anomaly;
- $\boldsymbol{r}$ is the position vector of the orbiting body in an Earth-centred inertial reference frame; and
- $\boldsymbol{q}$ is the perturbing acceleration caused by atmospheric drag:

$$\boldsymbol{q} = -\frac{1}{2}\rho\frac{C_D A}{m} V_r^2 \frac{\boldsymbol{V_r}}{V_r} \tag{6}$$

  with ρ the atmospheric density, C_D, A and m the drag coefficient, cross-sectional area and mass of the orbiting body, and $\boldsymbol{V_r} = \dot{\boldsymbol{r}} - \omega_E \times \boldsymbol{r}$ the relative velocity of the orbiting body with respect to Earth's atmosphere, with $\boldsymbol{\omega_E}$ Earth's rotational velocity, which can be assumed to be constant and equal to $[0, 0, \omega_E]^T$, with $\omega_E = 7.292115 \times 10^{-5}$ rad/s. To obtain the results presented in this paper, ρ was determined using the NRLMSISE-00 model (Picone et al., 2002), and C_D and A were assumed to be constant and equal to 2.2 and 15 m^2, respectively (Fisher and David, 2014; Morand et al., 2012).

The averaging integrals of drag in Eq. (4) can be solved numerically using a Gaussian quadrature. The integration limits are $L_1 = -\overline{f} + \omega + \Omega$ and $L_2 = \overline{f} + \omega + \Omega$, where $\overline{f}$ is the critical true anomaly:

$$\overline{f} = \arccos \frac{\frac{a(1-e^2)}{R_E + \overline{h}} - 1}{e} \tag{7}$$

where R_E is Earth's radius and $\overline{h}$ is a critical altitude above which the effects of atmospheric drag can be neglected. It was found that this limit could be set as low as 300 km and an odd number of at least 11 quadrature nodes was required to obtain accurate results, i.e. each integral had to be evaluated at at least 11 points close to perigee passage.

In order to speed up orbital propagation, it was investigated whether it was possible to derive a pseudo-empirical expression relating the mean element rates A_i due to atmospheric drag and the values of some instantaneous parameters, so that numerical integration could be avoided. By using the orbital and body characteristics of 673 GTO objects put into orbit between January 2010 and April 2017 provided by the JFSCC (2018), the following expression was derived:

$$A_i \approx \left\{ \left[\left(\frac{h_p}{A} + B \right) (F_{10.7} - C)^D \right] + E \right\} [A_i]_p \tag{8}$$

with $A = 3.65 \times 10^6$ m, $B = 0.152$, $C = 45.3$, $D = 0.07$ and $E = 0.011$. The value of the solar 10.7-cm radiation flux, $F_{10.7}$, is identical to the input for the NRLMSISE-00 atmosphere model. The mean element rates $[A_i]_p$ are obtained from Eq. (4) by solving the integrals by evaluating them just at one node (at perigee, i.e. for $L = \omega + \Omega$) and assuming $\overline{h} = 600$ km in order to find L_1 and L_2. This makes the estimation of the effects of atmospheric drag about ten times faster and the overall propagation process about twice as fast, while keeping proper accuracy levels.

In the semi-analytical propagator, short-period terms were neglected. Consequently, the re-entry condition (defined as $h_p < 100$ km) was evaluated using the mean elements, i.e. the mean semi-major axis and the mean eccentricity. The amplitude of the short-period variations of the perigee altitude was estimated to be about 5 km for a characteristic GTO with a perigee altitude of 200 km (or around 2.5%). Although short-period terms were neglected during propagation, the short-period terms caused by J_2 *were* taken into account at the beginning of the propagation in order to convert the initial osculating elements to mean elements.

The semi-analytical propagator was verified and benchmarked against the accurate (but slower) fully-numerical Cowell propagator. It was found that the root-mean-square (RMS) deviation of the lifetime predictions of the semi-analytical propagator compared to those of the Cowell propagator was about 4.1%, whilst the overall computation times were reduced to just 2 to 2.5%.

4. Mitigation of Ariane 5 debris

4.1. Methodology

In order to determine the optimal launch conditions leading to the shortest lifetimes for the GTO debris resulting from Ariane 5 launches, some parameters will be fixed and others will be set to vary within a range of interest with constant step-sizes, leading to the generation of a set of GTO objects (following different orbits) referred to as *study domain*. This domain will be characterised by propagating the orbits of all objects therein until the re-entry condition –mean perigee altitude below 100 km– is met. The time difference between the re-entry epoch and the initial epoch is defined as the *lifetime* of the object. In case re-entry had not happened within 30 years for any of the objects in the study domain, the propagation of that object's orbit would be terminated, thus obtaining a lower limit for its actual lifetime.

The variable parameters were chosen to be the epoch of injection into GTO, t_0, and the initial RAAN, Ω_0, due to three reasons: they were identified as two of the three parameters the lifetime of GTO objects is most sensitive to (Siebold and Reynolds, 1995); their values are not constrained by Ariane launch procedures (Arianespace, 2016), leading to a widespread distribution in comparison to other parameters (cf. Fig. 4); and they can be easily changed at late stages of the mission with little or no impact. The remainder of the relevant parameters were set to have a constant nominal value because they did not fulfil these three conditions.

The epoch of injection into GTO is defined as the epoch at which the rocket engines are turned off and the payload separated, beginning a period of unpowered flight. Since Ariane 5 performs a direct ascent to GEO, the time difference between the epoch of injection into GTO and the actual launch epoch is small, typically about ten minutes (Arianespace, 2016). Thus, since the step-size for sampling this variable will be larger than ten minutes, this time difference can be (and will be) neglected, and t_0 will be referred to as launch epoch (or launch date) hereinafter.

Table 2: Values of the variable parameters.

Parameter	**Mininum**	**Maximum**	**Step-size**
Launch date	1 Jan 1985	31 Dec 1985	4 days
Initial RAAN	0 deg	360 deg	4 deg

Table 3: Values of the fixed parameters.

Parameter	**Nominal**	**1-σ uncty.**
Initial perigee altitude [km]	250	25
Initial apogee altitude [km]	34 700	900
Initial inclination [deg]	6	1
Initial argument of perigee [deg]	178	5
Initial true anomaly [deg]	0	–
Object mass [kg]	3 000	30
Object cross-sectional area [m^2]	15	5
Object drag coefficient [–]	2.2	–

As discussed in Section 3, it is necessary to perform a statistical analysis of the study domain, given the extreme sensitivity of the lifetime of GTO objects to initial orbital conditions and environmental and body-related parameters. As a consequence, instead of performing a single propagation for each (t_0, Ω_0) pair (or *case*) in the study domain, 100 propagations (or *subcases*) were studied for each, by varying the value of the fixed parameters through the introduction of pseudo-random, normally-distributed noise to account for deviations from the nominal values. This effectively increases the number of objects in the study domain by two orders of magnitude and lets us obtain statistical indicators for each case in the study domain, such as the mean/median lifetime or the probability of the lifetime being shorter than a given number of years.

The variable parameters were changed within the ranges and with the step-sizes specified in Table 2. The nominal values for the fixed parameters are provided in Table 3. The standard deviation σ of the normally-distributed noise (with zero mean) is also provided for a subset of these parameters, while for the rest it is assumed to be zero (i.e. no noise). The values presented in Table 3 were selected based on the characterisation of the Ariane 5 GTO debris generated in the period 2012–2017 (cf. Fig. 4), Ariane 5 mission procedures and launch constraints (Arianespace, 2016), and previous studies on the propagation of GTOs (Fisher and David, 2014; Morand et al., 2012). Invalid subcases (e.g. objects with negative cross-sectional area) were excluded, leading to some cases with less than 100 subcases. Invalid subcases were extremely rare: 1 109 out of 837 200, or 0.13%.

The initial true anomaly was set to zero for all propagations, consistent with the assumption that the period of time from launch until injection into GTO is negligible. As a consequence, the ground track of any GTO object starts from the European Spaceport in Kourou. Under

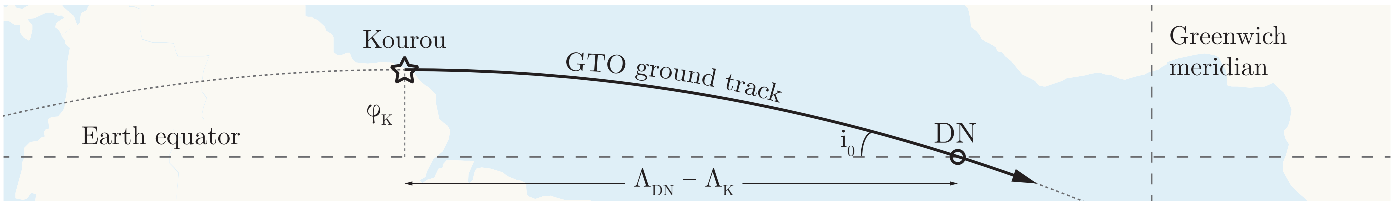

Fig. 8: Simplified geometry for converting between RAAN and local time of launch. The launch site is denoted as Kourou and its latitude and longitude as φ_K and Λ_K, respectively; the projection of the orbit's descending node on Earth's surface as DN, and its longitude as Λ_{DN}; and the orbit initial inclination as i_0. The spherical triangle defined by the GTO ground track, Earth equator and the meridian passing through the launch site is used to determine Λ_{DN}, from which the RAAN can be derived for a given epoch.

this assumption, and assuming no further manoeuvres, the launch time is uniquely defined by the day of launch and the initial RAAN, i.e. it is not possible to choose an arbitrary launch time on a given day if the target RAAN is fixed. In practice, the initial RAAN of the GTO is not chosen but derives from the choice of the launch date and time, and the orbit inclination and argument of perigee. As a consequence, any pair of launch day and initial RAAN values can be converted to a pair of launch day and launch time values. This transformation was carried out using the expressions provided by Wertz (2001), and assuming the initial ground-track geometry in Fig. 8. Clearly, this is only valid for objects with orbital inclinations larger than Kourou's latitude, i.e. 5.36 degrees. Half of the GTO objects generated in the period 2012–2017 from Ariane 5 launches had smaller inclinations (cf. Fig. 4**d**), presumably because of inclination-correction manoeuvres taking place before injection into GTO. The RAAN–time conversion expressions were validated using the data from the Ariane 5 GTO debris with initial inclinations of at least 5.36 degrees provided by the JFSCC (2018). For these objects it was assumed that no inclination-correction manoeuvre had taken place. It was found that the RMS deviation of the RAANs estimated using the aforementioned expressions compared to the actual RAANs from satellite-tracking data is 8.6 degrees, which is considered to be sufficiently small taking into account all the assumptions that have been made and possible measuring errors in the two-line elements data.

The launch date has been chosen to range from the first to the last day of 1985. At first, one may be tempted to believe that this choice would make the obtained results unusable, as it would be interesting to know what the lifetimes of GTO debris generated in the period 2012–2017 would have been if different launch conditions had held, and what the optimal launch conditions will be for future launches. Thus, it would appear reasonable to choose the launch date to range from 2012 onwards. However, since the propagations are typically carried out over tens of years before re-entry takes place, predictions about the solar activity levels would be needed, which can introduce significant errors in the lifetime predictions (Morand et al., 2012). Solar flux heats up the atmosphere, making it expand, which causes lower, denser layers of the atmosphere to raise, leading to the debris experiencing more atmospheric drag when solar flux is near maximum. The solar activity cycle has a period of 11 years, and the most relevant parameter, $F_{10.7}$, can range from around 50 solar flux units (sfu) at solar minima up to 250 sfu at solar maxima (Tapping, 2013). This is clearly the long-period perturbation with the longest period. The other relevant long-period perturbations have much shorter periods, such as the ones caused by the gravity of the Sun and the Moon, with periods of 6 months and 2 weeks, respectively (Wertz, 2001). This means that, if the effects of the solar activity levels on the lifetimes of Ariane 5 GTO debris were small, the results for 1985 would also be applicable to any year in the future (and in the past). During the discussion of the results, it will be shown that, although the lifetime of GTO debris can change significantly depending on whether the launch takes place close to a solar minimum or a solar maximum, the optimal launch conditions remain rather constant throughout the whole solar cycle. Thus, when presenting the results, the year of launch will be omitted, so the first variable parameter will effectively be the day of the year at which launch takes place.

4.2. Results

The characterisation of the study domain defined in Tables 2 and 3 was carried out by completing a total of 836 091 orbit propagations. For each case in the study domain, the probability of re-entry in less than 25 years, i.e. the probability of compliance with debris mitigation guidelines, was determined by assessing the lifetimes of all of its subcases, leading to the results provided in Fig. 9. As can be seen, the actual GTO debris generated in the period 2012–2017 correspond to cases with relatively low compliance probabilities, ranging from 0 to 72%, and with a mean compliance rate of a mere 18%. The same results are provided in Fig. 10 after performing the RAAN–local time conversion discussed in Subsection 4.1 according to the geometry assumed in Fig. 8. It can be seen that the optimal launch times, i.e. leading to the highest compliance probabilities, remain rather constant throughout the year. Two bands of interest are identified: one at around 2 AM and another one at around 2 PM (Kourou local time), both with similar compliance probabilities. Since most launches in the period 2012–2017 took place well after noon (cf. Fig. 4**a**), the (local) optima around 2 AM will not be considered further, and only the optimum launch conditions

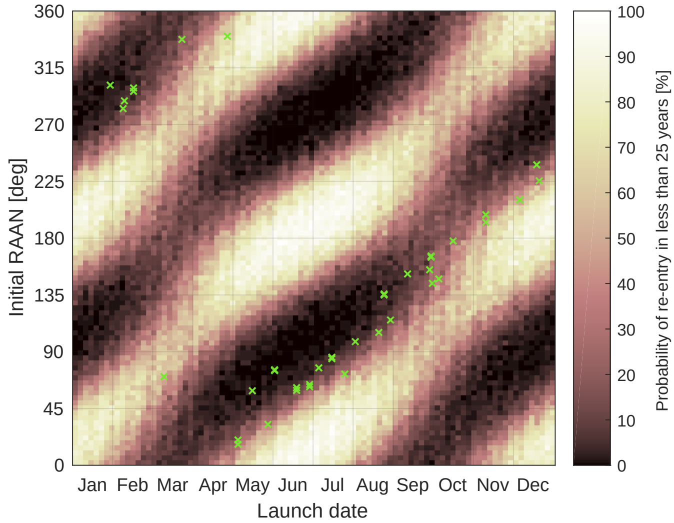

Fig. 9: Probability of compliance with debris mitigation guidelines for GTO debris from Ariane 5 GEO launches as a function of day of launch and initial right ascension of the ascending node. Debris from actual launches in the period 2012–2017 are marked with crosses.

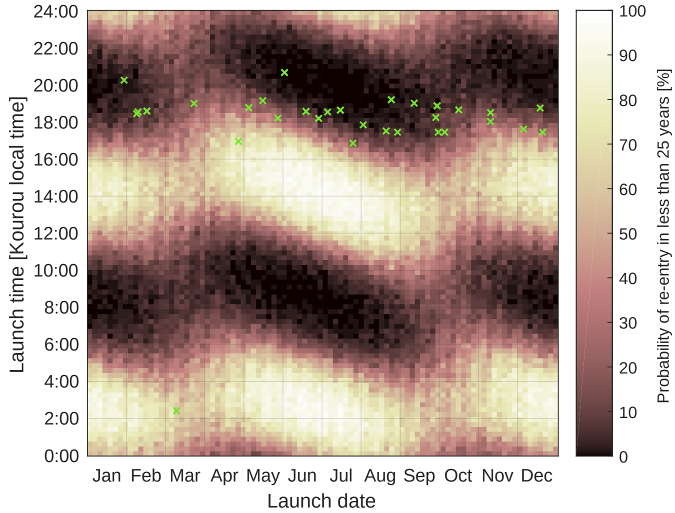

Fig. 10: Probability of compliance with debris mitigation guidelines for GTO debris from Ariane 5 GEO launches as a function of day and time of launch. Debris from actual launches in the period 2012–2017 are marked with crosses.

centred around the 2 PM band will be discussed.

The optima in the 2 PM band are provided in Fig. 11 (top) as a solid line. As already stated, the launch time leading to the highest rate of compliance with debris mitigation guidelines remains rather constant, ranging from 12:15 to 16:00 in the studied domain. However, the highest compliance rate does change significantly throughout the year, as shown in Fig. 11 (bottom). For launches in the months of March or October, the probability of complying with debris mitigation guidelines cannot be higher than about 60%. Launches from May to August are better, with compliance rates above 90%. As seen in Fig. 11 (top), the actual launches in the period 2012–2017 mostly took place around 6–7 PM. Only one of the launches (late April) has a compliance probability of more than 50%. By bringing forward the launch time by about four hours, compliance rates could be improved significantly. Since geostationary satellites co-rotate with Earth, a change in the launch time would have little or no impact on the mission, since the same slot in the GEO ring (or phasing orbit) would be reached regardless of the launch time. The main limitation when choosing the launch time has to do with the preparatory operations that have to be carried out at the launch site before launch. If anything were to go wrong during these preparations, and the launch time would be delayed by a few hours, this could have severe consequences from a debris mitigation perspective. Thus, it may seem reasonable to target a launch time slightly before the optimum launch time (e.g. 20–30 minutes earlier than the optimum, depending on the season of the year), which would still lead to a high probability of compliance with debris mitigation guidelines in case the launch was postponed by no more than one hour. However, the determination of the *right* time offset is highly dependent on the historical launch delays of Ariane 5 launches and the confidence with which the targeted launch time can be met. Ariane launch procedures specify a launch window of one hour, within which launch should take place as soon as possible (Arianespace, 2016).

As already mentioned in Subsection 4.1, the results presented in this paper correspond to launches in 1985, which have been assumed to be generalisable to any year. The validity of such assumption was investigated by repeating the study for the same objects and orbits but for 1980. In 1985, solar activity levels were close to a minimum (which took place in 1986), while in 1980 they were at a maximum (Tapping, 2013). The predicted probability of compliance with debris mitigation guidelines can vary significantly depending on the studied year, as shown in Fig. 12. The deviation in the predicted rate of compliance ranges from −35 to 35 percentage points (pp), with an RMS deviation of 8.5 pp and a mean deviation of about 2 pp, i.e. the average rate of compliance is slightly higher when launching at a solar activity maximum, but it can become smaller for certain launch dates and times because of the counterintuitive effects introduced by the apsidal solar resonance. A larger atmospheric drag during the first years can lead to reaching solar resonance conditions at a different geometry (cf. Fig. 5), which can have a big impact on the long-term evolution of the perigee altitude (cf. Fig. 7).

Although the highest rates of compliance with debris mitigation guidelines can change throughout solar cycles, the optimal launch times stay rather constant regardless of the year in which the launch takes place. It was found that the optimal launch times around the 2 PM band for launches in 1980 differ between −112 and 112 minutes, with a mean difference of just one minute and an RMS deviation of 42 minutes. All optimal launch times were within 12:30 and 16:00. Thus, even though solar activity

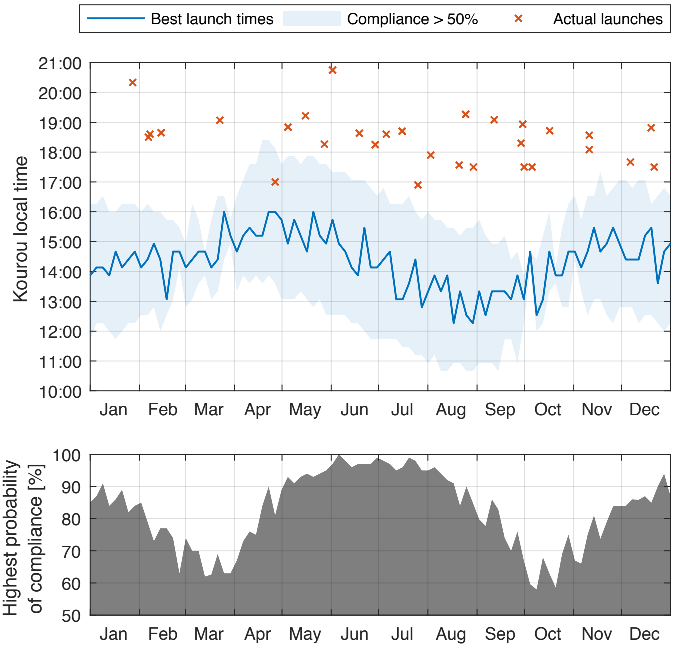

Fig. 11: Launch times leading to the highest probability of compliance with debris mitigation guidelines throughout the year (top), and corresponding highest probabilities of compliance (bottom). The shaded area in the top plot represents the launch times leading to a probability of compliance of at least 50%. Actual launches in the period 2012–2017 are marked with crosses (a launch at 2:20 has been omitted).

levels do have some impact on the rate of compliance with debris mitigation guidelines, the results presented in this paper for 1985 and the conclusions deriving therefrom can be said to be generalisable to any year at the expense of a limited loss in accuracy. For the determination of the optimal launch times for future Ariane 5 GEO missions it is recommended to use the results presented here just as a first, rough estimate, and to carry out propagations of the orbits for the targeted day or possible days of launch starting in the actual year of launch. However, predictions about solar activity levels will be needed, which may also introduce significant inaccuracies in the lifetime predictions. Yet, seriously improved compliance rates are to be expected.

5. Conclusions

It has been shown that the rate of compliance with debris mitigation guidelines for objects generated from the launch of geostationary satellites using the Ariane 5 launcher can be significantly increased by choosing a launch time close to 2 PM local time. It was found that the dependence of the optimal launch time on the date of launch is small, as it was found to always range from 12:15 to 16:00, regardless of the day of year and of whether the launch takes place at a solar maximum or minimum.

However, the highest probability of compliance, i.e. of re-entering in less than 25 years, was found to vary significantly throughout the year, with rates ranging from about

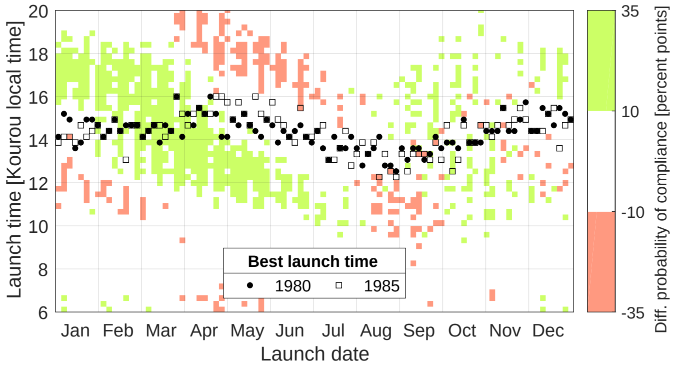

Fig. 12: Difference in the probability of compliance with debris mitigation guidelines for launches in 1980 (close to a solar maximum) compared to launches in 1985 (close to a solar minimum). Best launch times are marked with dots for launches in 1980 and with empty squares for launches in 1985.

60% for launches in March and October, up to probabilities above 90% from May to August. The rates of compliance depend less strongly on the solar activity levels: when comparing predicted compliance rates for launches in 1980 (close to a solar maximum) to those for launches in 1985 (close to a solar minimum), it was found that, on average, they were just about two percentage points higher.

It was found that all Ariane 5 launches of geostationary satellites in the period 2012–2017 took place outside of the launch window leading to high compliance rates with debris mitigation guidelines, with 30 out of 31 launches between 16:54 and 20:45 Kourou local time. Only one of the generated GTO objects is likely to re-enter in less than 25 years. By bringing the launch time close to 2 PM, the rate of compliance could be increased from a mean value of 18% to values raging from 60 to 100%. Since geostationary satellites co-rotate with Earth, changing the launch time would have little or no impact on the payload and mission development, and the only foreseeable consequences would be for the logistics of preparatory procedures at the launch site.

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