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# Space systems — Estimation of orbit lifetime

*Systèmes spatiaux — Estimation de la durée de vie en orbite*



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# Foreword

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The committee responsible for this document is ISO/TC 20, *Aircraft and space vehicles*, Subcommittee SC 14, *Space systems and operations*.



This second edition cancels and replaces the first edition (ISO 27852:2011), which has been technically revised.

# Introduction

This International Standard is a supporting document to ISO 24113 and the GEO and LEO disposal standards that are derived from ISO 24113. The purpose of this International Standard is to provide a common consensus approach to determining orbit lifetime, one that is sufficiently precise and easily implemented for the purpose of demonstrating compliance with ISO 24113. This project offers standardized guidance and analysis methods to estimate orbital lifetime for all LEO-crossing orbit classes.

# Space systems — Estimation of orbit lifetime

## 1 Scope

This International Standard describes a process for the estimation of orbit lifetime for spacecraft, launch vehicles, upper stages and associated debris in LEO-crossing orbits.

This International Standard also clarifies the following:

a) modelling approaches and resources for solar and geomagnetic activity modelling;

b) resources for atmosphere model selection;

c) approaches for spacecraft ballistic coefficient estimation.

## 2 Normative References

The following documents, in whole or in part, are normatively referenced in this document and are indispensable for its application. For dated references, only the edition cited applies. For undated references, the latest edition of the referenced document (including any amendments) applies.

ISO 24113, *Space systems — Space debris mitigation requirements*



## 3 Terms, definitions, symbols and abbreviated terms



### 3.1 Terms and definitions

For the purposes of this document, the terms and definitions given in ISO 24113 and the following apply.

**3.1.1**
**orbit lifetime**
elapsed time between the orbiting spacecraft's initial or reference position and orbit demise/reentry

Note 1 to entry: An example of the orbiting spacecraft's reference position is the post-mission orbit.

Note 2 to entry: The orbit's decay is typically represented by the reduction in perigee and apogee altitudes (or radii) as shown in Figure 1.

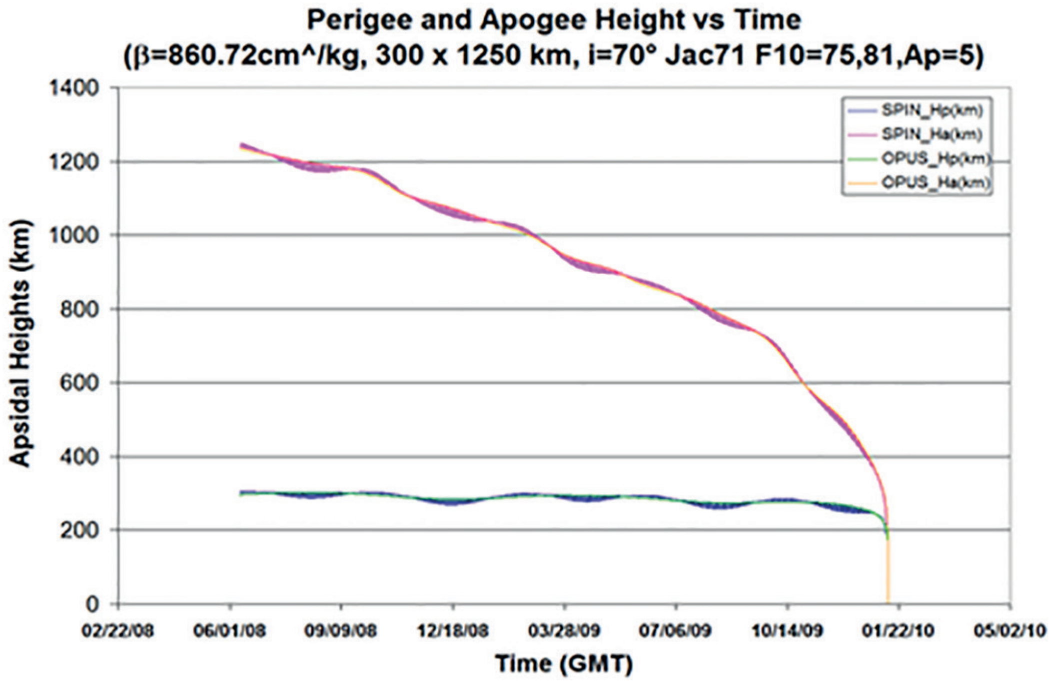


**Figure 1 — Sample of orbit lifetime decay profile**



**3.1.2**
**earth equatorial radius**
equatorial radius of the Earth



Note 1 to entry: The equatorial radius of the Earth is taken as 6 378,137 km and this radius is used as the reference for the Earth's surface from which the orbit regions are defined.

**3.1.3**
**high area-to-mass**
**HAMR**
space objects are considered to be high area-to-mass (or HAMR) objects if the ratio of area to mass exceeds 0,1 $m^2$/kg

**3.1.4**
**LEO-crossing orbit**
low-earth orbit, defined as an orbit with perigee altitude of 2 000 km or less

Note 1 to entry: As can be seen in Figure A.1, orbits having this definition encompass the majority of the high spatial density spike of spacecraft and space debris.

**3.1.5**
**long-duration orbit lifetime prediction**
orbit lifetime prediction spanning two solar cycles or more (e.g. 25-year orbit lifetime)

**3.1.6**
**mission phase**
period of a mission during which specified communications characteristics are fixed.

Note 1 to entry: The transition between two consecutive mission phases may cause an interruption of the communications services.

**3.1.7**
**post-mission orbit lifetime**
duration of the orbit after completion of all mission phases

Note 1 to entry: The disposal phase duration is a component of post-mission duration.

 

**3.1.8**
**space object**
man-made object in outer space

**3.1.9**
**orbit**
path followed by a space object

**3.1.10**
**solar cycle**
≈11-year solar cycle based on the 13-month running mean for monthly sunspot number and is highly correlated with the 13-month running mean for monthly solar radio flux measurements at the 10,7 cm wavelength

Note 1 to entry: Historical records back to the earliest recorded data (1945) are shown in Figure 2.

Note 2 to entry: For reference, the 25-year post-mission orbit lifetime constraint specified in ISO 24113 is overlaid onto the historical data; it can be seen that multiple solar cycles are encapsulated by this long time duration.

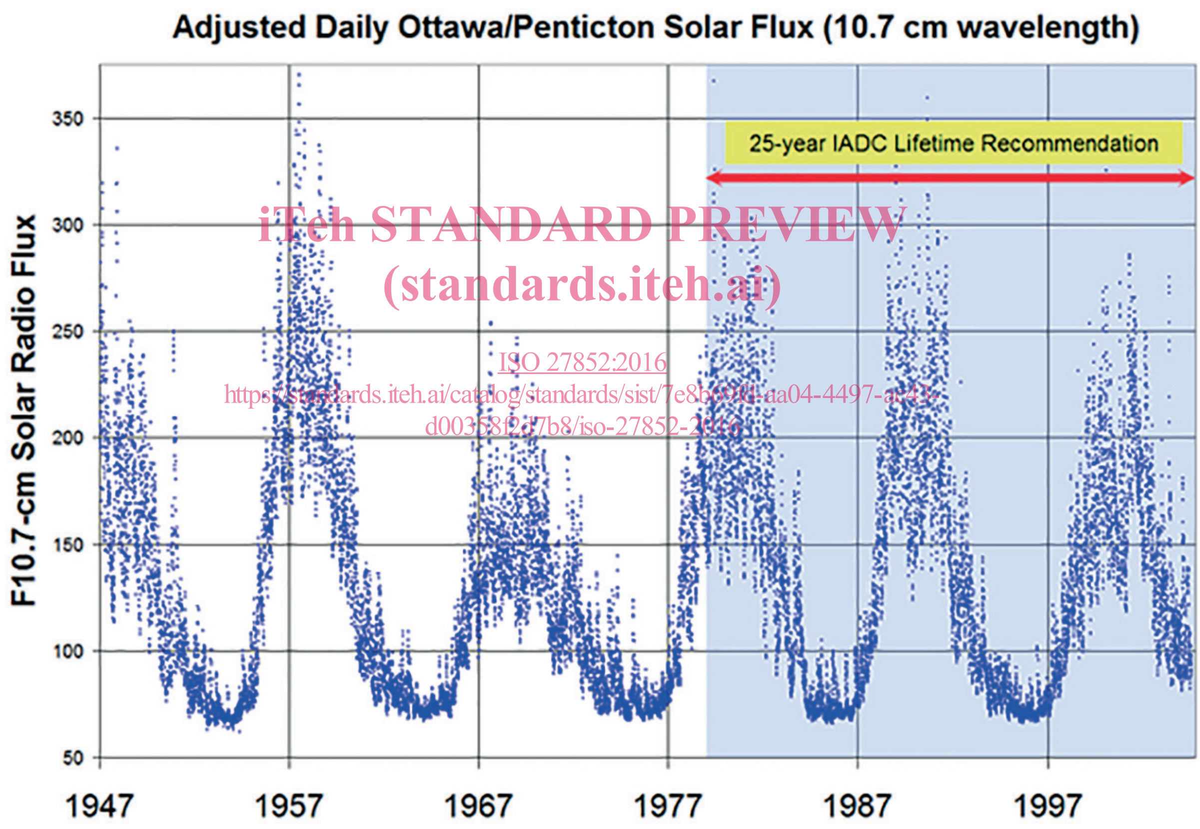



**Figure 2 — Solar cycle (≈11-year duration)**

## 3.2 Symbols

| | |
|---|---|
| $a$ | orbit semi-major axis |
| $A$ | spacecraft cross-sectional area with respect to the relative wind |
| $A_p$ | earth daily geomagnetic index |
| $\beta$ | ballistic coefficient of spacecraft = $C_D \cdot A/m$ |
| $C_D$ | spacecraft drag coefficient |
| $C_R$ | spacecraft reflectivity coefficient |

| | |
|---|---|
| $e$ | orbit eccentricity |
| $F_{10,7}$ | solar radio flux observed daily at 2 800 MHz (10,7 cm) in solar flux units ($10^{-22}$W $m^{-2}$ $Hz^{-1}$) |
| $F_{10,7}$ Bar | solar radio flux at 2 800 MHz (10,7 cm), averaged over three solar rotations |
| $H_a$ | apogee altitude = a (1 + $e$) − $R_e$ |
| $H_p$ | perigee altitude = a (1 − $e$) − $R_e$ |
| $m$ | mass of spacecraft |
| $R_e$ | equatorial radius of the Earth |

## 3.3 Abbreviated terms

| | |
|---|---|
| 3Bdy | third-body (perturbations) |
| CAD | computer-aided design |
| GEO | geosynchronous earth orbit |
| GTO | geosynchronous transfer orbit |
| HAMR | high area-to-mass ratio |
| IADC | Inter-Agency Space Debris Coordination Committee |
| ISO | International Organization for Standardization |
| LEO | low earth orbit |
| N/A | not applicable |
| RAAN | orbit right ascension of the ascending node (angle between vernal equinox and orbit ascending node, measured CCW in equatorial plane, looking in–Z direction) |
| SRP | solar radiation pressure |
| STSC | Scientific and Technical Subcommittee of the Committee |
| UNCOPUOS | United Nations Committee on the Peaceful Uses of Outer Space |



# 4 Orbit lifetime estimation

## 4.1 General requirements

The orbital lifetime of LEO-crossing mission-related objects shall be estimated using the processes specified in this International Standard. In addition to any user-imposed constraints, the post-mission portion of the resulting orbit lifetime estimate shall then be constrained to a maximum of 25 years per ISO 24113 using a combination of (a) initial orbit selection, (b) spacecraft vehicle design, (c) spacecraft launch and early orbit concepts of operation which minimize LEO-crossing objects, (d) spacecraft ballistic parameter modifications at EOL, and (e) spacecraft deorbit maneuvers.

## 4.2 Definition of orbit lifetime estimation process

The orbit lifetime estimation process is represented generically in Figure 3.

 

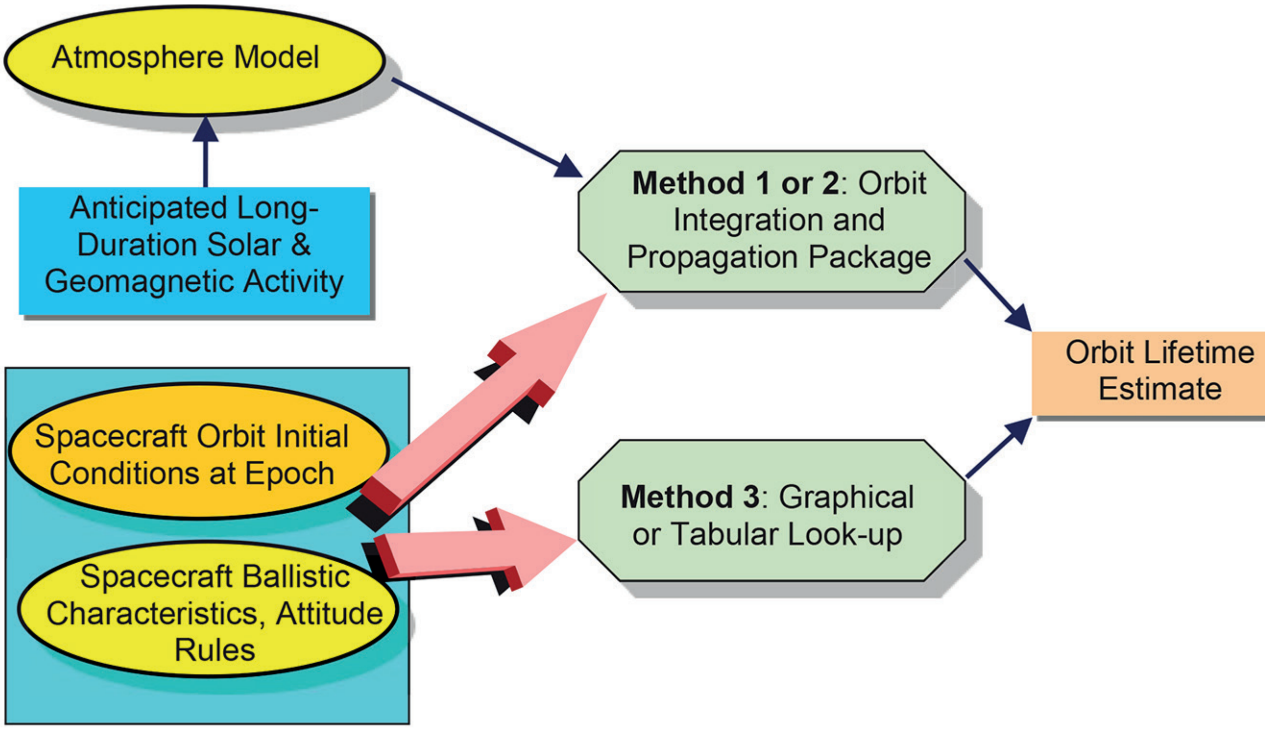


**Figure 3 — Orbit lifetime estimation process[4]**



## 5 Orbit lifetime estimation methods and applicability



### 5.1 General

There are three basic analysis methods used to estimate orbit lifetime,[3] as depicted in Figure 3. Determination of the method used to estimate orbital lifetime for a specific space object shall be based upon the orbit type and perturbations experienced by the spacecraft as shown in Table 1.

**Table 1 — Applicable method with mandated conservative margins of error (in percent) and required perturbation modelling**

| | Special orbit | | Conservative margin applied to each method | | | |
|---|---|---|---|---|---|---|
| **Orbit apogee altitude**, km | **Sun-sync?** | **High area-to-mass?** | **Method 1:** **Numerical integration** | **Method 2:** **Semi-analytic** | **Method 3:** **Table look-up** | **Method 3** **Graph, formula fit** |
| Apogee < 2 000 km | No | No | No margin req'd | 5 % margin | 10 % margin | 25 % margin |
| Apogee < 2 000 km | No | Yes | No margin; use SRP | 5 % margin; use SRP | 10 % margin IFF $C_r \approx 1{,}7$ | N/A |
| Apogee < 2 000 km | Yes | No | No margin req'd | 5 % margin | N/A | N/A |
| Apogee < 2 000 km | Yes | Yes | No margin req'd; use SRP | 5 % margin; use SRP | N/A | N/A |
| Apogee > 2 000 km | Either | Either | No margin req'd; use 3Bdy+SRP | 5 % margin; use 3Bdy+ SRP | N/A | N/A |

N/A = not applicable

3Bdy = third-body perturbations

SRP = solar radiation pressure



Method 1, certainly the highest fidelity model, utilizes a numerical integrator with a detailed gravity model, third-body effects, solar radiation pressure, and a detailed spacecraft ballistic coefficient model. Method 2 utilizes a definition of mean orbital elements,[4] [5] semi-analytic orbit theory and average spacecraft ballistic coefficient to permit the very rapid integration of the equations of motion while still retaining reasonable accuracy. Method 3 is simply a table lookup, graphical analysis or evaluation of formulae that have been fit to pre-computed orbit lifetime estimation data obtained via the extensive and repetitive application of Methods 1 and/or 2. It is worth noting that all methods (1 through 3) shall include at gravity zonals $J_2$ and $J_3$ at a minimum.

## 5.2 Method 1: High-precision numerical integration

Method 1 is the direct numerical integration of all accelerations in Cartesian space, with the ability to incorporate a detailed gravity model (e.g. using a larger spherical harmonics model to address resonance effects), third-body effects, solar radiation pressure, vehicle attitude rules or aero-torque-driven attitude torques, and a detailed spacecraft ballistic coefficient model based on the variation of the angle-of-attack, with respect to the relative wind. Atmospheric rotation at the Earth's rotational rate is also easily incorporated in this approach. The only negative aspects to such simulations are (a) they run much slower than Method 2, (b) many of the detailed data inputs required to make this method realize its full accuracy potential are simply unavailable, and (c) any gains in orbit lifetime prediction accuracy are frequently overwhelmed by inherent inaccuracies of atmospheric modelling and associated inaccuracies of long term solar activity predictions/estimates. However, to analyse a few select cases where such detailed model inputs are known, this is undoubtedly the most accurate method. At a minimum, Method 1 orbit lifetime estimations shall account for $J_2$ and $J_3$ perturbations and drag using an accepted atmosphere model and an averaged ballistic coefficient. In the case of high apogee orbits (e.g. geosynchronous transfer orbits) or other resonant orbits, sun and moon third-body perturbations and solar radiation pressure effects shall also be modelled (see Reference [28] for additional discussion).

 

### 5.3 Method 2: Rapid semi-analytical orbit propagation

Method 2 analysis tools utilize semi-analytic propagation of mean orbit elements[4] [5] influenced by gravity zonals $J_2$ and $J_3$ and selected atmosphere models. The primary advantage of this approach over direct numerical integration of the equations of motion (Method 1) is that long-duration orbit lifetime cases can be quickly analysed (e.g. 1 s versus 1 700 s CPU time for a 30-year orbit lifetime case). While incorporation of an attitude-dependent ballistic coefficient is possible for this method, an average ballistic coefficient is typically used. At a minimum, Method 2 orbit lifetime estimations shall account for $J_2$ and $J_3$ perturbations and drag using an accepted atmosphere model and an average ballistic coefficient. In the case of high apogee orbits (e.g. GTO), sun and moon third-body perturbations shall also be modelled.

### 5.4 Method 3: Numerical table look-up, analysis and fit formula evaluations

In this final method, one uses tables, graphs and formulae representing data that was generated by exhaustively using Methods 1 and 2 (see 5.2 and 5.3). The graphs and formulae provided in this International Standard can help the analyst crudely estimate orbit lifetime for their particular case of interest; the electronic access to tabular look-up provided via this International Standard (at www.CelesTrak.com) permits the analyst to estimate orbit lifetime for their particular case of interest via interpolation of Method 1 or Method 2 gridded data; all such Method 3 data in this International Standard were generated using Method 2 approaches. At a minimum, Method 3 orbit lifetime products shall be derived from Method 1 or Method 2 analysis products meeting the requirements stated above. When using this method, the analyst shall impose at least a 10 % margin of error to account for table look-up interpolation errors. When using graphs and formulae, the analyst shall impose a 25 % margin of error.



### 5.5 Orbit lifetime sensitivity to sun-synchronous



For sun-synchronous orbits, orbit lifetime has some sensitivity to the initial value of RAAN due to the density variations with the local sun angle. Results from numerous orbit lifetime estimations show that orbits with 6:00 am local time have longer lifetime than orbits with 12:00 noon local time by about 5,5 %. [3] This maximum difference (500 d) translates into a 5 % error which can be corrected by knowing the local time of the orbit. As a result, Methods 1 or 2 analyses of the actual sun-synchronous orbit condition shall be used when estimating the lifetime of sun-synchronous orbits (see References [28] and [38], where more details are given).

### 5.6 Orbit lifetime statistical approach for high-eccentricity orbits (e.g. GTO)

For high-eccentricity orbits (particularly geosynchronous transfer orbits or GTO), it can be difficult to iterate to lifetime threshold constraints due to the coupling in eccentricity between the third-body perturbations and the drag decay. Due to this convergence difficulty, only Method 1 or 2 analyses shall be used when determining initial conditions which achieve a specified lifetime threshold for such orbits.

Sample analyses of GTO launcher stages (see References [29] and [30]) highlight this orbit lifetime sensitivity to initial conditions (orbit, spacecraft characteristic and force model), leading to a wide spectrum of orbital lifetimes.

Some theoretical considerations about the dynamical properties of GTO orbits are provided in References [29] and [36].

The following test case illustrates the complex dynamical properties of GTO. Initial parameters are provided in Table 2.

**Table 2 — GTO initial conditions for the Monte Carlo simulation**

| | |
|---|---|
| Perigee altitude | 200 km |
| Apogee altitude | GEO altitude |
| Inclination | 2° |

**Table 2** *(continued)*

| | |
|---|---|
| Area to mass ratio | 5e-3 $m^2$/kg |
| Solar activity | Constant (F10.7 = 140 sfu Ap = 15) |
| Drag coefficient | Constant = 2,2 |
| Reflectivity coefficient | Constant = 2 |

Figure 4 shows lifetime results (years) when varying the initial date and the initial local time of perigee. This latest parameter is defined as the angle in the equator between the sun direction and the orbit perigee, measured in hours. The date was chosen from day 1 to 365 in year 1998 and the local time of perigee was chosen by varying the right ascension of ascending node from 0π to 2π. A total of 2 500 different initial conditions were generated.

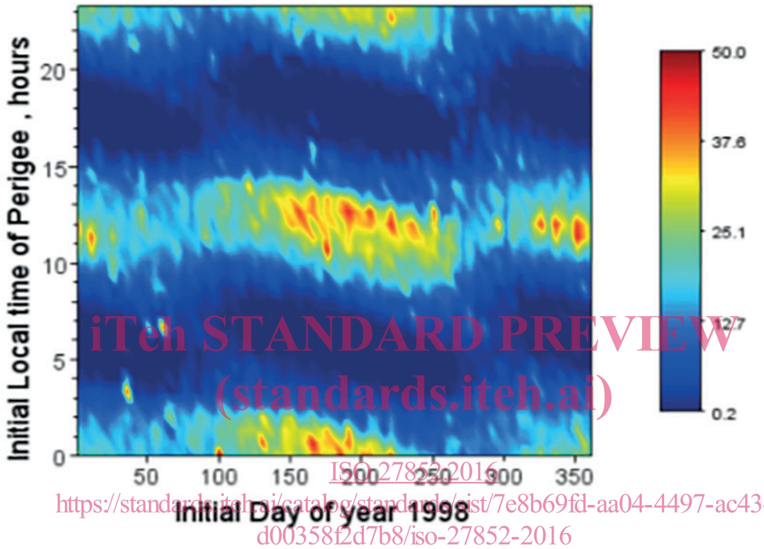




**Figure 4 — Lifetime variations with respect to initial date and local time of perigee (year)**

The shapes of the lifetime contours confirm that initial day of year and local time of perigee are initial conditions that make sense to describe GTO evolution since strong patterns are visible. The amplitudes of lifetimes variations are worth noting: from several months to more than 50 years. Previous results (see References [30] and [37]) are illustrated here: the longest lifetimes are obtained for initial sun-pointing (12 h local time) or anti sun-pointing (24 h local time) perigee with an initial date around the solstices. Note that the dark red pixels drawn in dark blue areas, as seen for initial day 60 and local time 7 h, are an indication of the presence of strong resonance phenomena. We know that the year also has an influence, to a lesser extent, through the moon perturbation.

Figure 5 shows semi-major axis evolution for several propagations of a typical low-inclined GTO. The different curves correspond to changes of 0,1 % or 1 % in the area to mass ratio of the object (A/m), which is far below the level of incertitude on this parameter. These dispersions lead to variations of decades in the re-entry duration. Such a strong non-linear behaviour is explained by the aforementioned resonances. One can see that semi-major axis evolutions are quite similar between all propagation cases until the entrance in the coupling between J2 and sun perturbations, for a semi-major axis equal to about 15 500 km. The duration of the resonance (period when the semi-major axis remains constant) and, thus, the rest of the propagation are completely different. A similar figure can be plotted by keeping the area to mass ratio constant and slightly changing the solar activity.

 

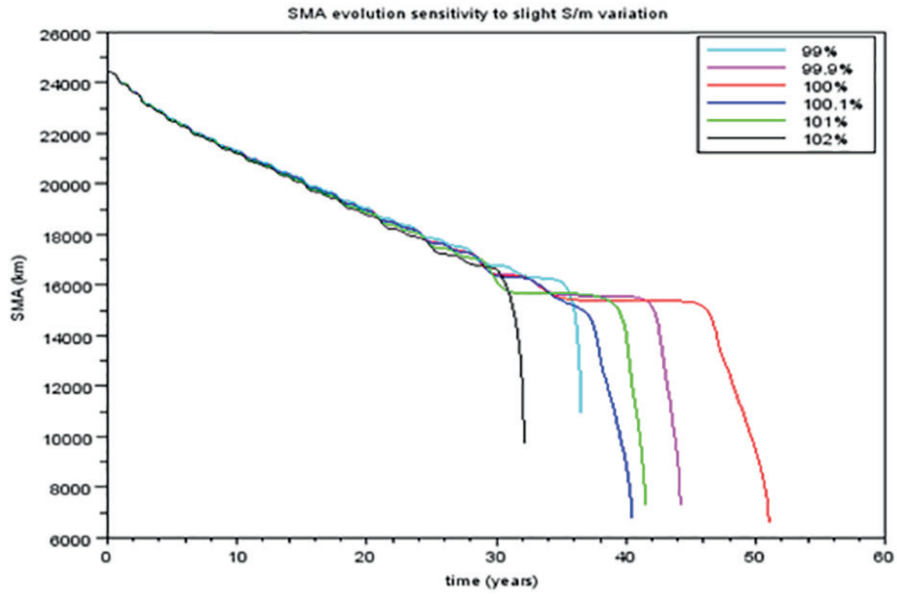


**Figure 5 — SMA evolution sensitivity to slight A/m variations (from 0,1 to 2 %)**

These examples show that resonance phenomena have substantial impacts on orbital elements evolution that can neither be predicted nor managed. Cumulated uncertainties on drag force between the extrapolation start (mission disposal manoeuvre, for example) and the instant when the resonance occurs make the entry condition in this resonance prone to strong variations. As a consequence, trying to estimate lifetime of GTOs using only one extrapolation may lead to erroneous conclusion since tiny changes in the initial conditions, spacecraft characteristics or force models end in very different lifetime results. Exceptions to that would be objects on a GTO whose semi major axis has already decreased enough to avoid resonances or to be very close to them. However, since resonance conditions change with regards to the possible resonant angles, one can see that performing several propagation cases is advised to get robust results. As a conclusion, only statistical results are adequate to estimate the strong variations of GTO lifetimes.



As a consequence, one should not say “this object’s lifetime is Y years” in GTO but rather “the lifetime of this object is shorter than Y years with a probability p”, coming from a cumulative distribution function (see example below).

The key parameter uncertainties to be taken into account in the lifetime estimation are

— initial conditions (date, orbit parameters),

— ballistic coefficient and drag coefficient, and

— solar activity.

The following test case (see Reference [32]) provides results of Monte Carlo simulations. Initial parameters are described in Table 3. A total of 2 500 different initial conditions were generated.

 