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Space systems — Estimation of orbit lifetime

Systèmes spatiaux — Estimation de la durée de vie en orbite

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Contents

Page

Foreword

ISO (the International Organization for Standardization) is a worldwide federation of national standards bodies (ISO member bodies). The work of preparing International Standards is normally carried out through ISO technical committees. Each member body interested in a subject for which a technical committee has been established has the right to be represented on that committee. International organizations, governmental and non-governmental, in liaison with ISO, also take part in the work. ISO collaborates closely with the International Electrotechnical Commission (IEC) on all matters of electrotechnical standardization.

International Standards are drafted in accordance with the rules given in the ISO/IEC Directives, Part 2.

The main task of technical committees is to prepare International Standards. Draft International Standards adopted by the technical committees are circulated to the member bodies for voting. Publication as an International Standard requires approval by at least 75 % of the member bodies casting a vote.

ISO 27852 was prepared by Technical Committee ISO/TC 20, *Aircraft and space vehicles*, Subcommittee SC 14, *Space systems and operations*.

Introduction

This standard is a supporting document to ISO 24113[3], Space systems -- Space debris mitigation requirements, and the GEO and LEO disposal standards that are derived from ISO 24113. The purpose of this 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 standard describes a process for the estimation of orbit lifetime for satellites, launch vehicles, upper stages and associated debris in LEO-crossing orbits.

The international standard also clarifies the following:

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

ii) resources for atmosphere model selection;

iii) approaches for satellite ballistic coefficient estimation.

2 Terms, definitions, symbols and abbreviated terms

2.1 Terms and definitions

For the purposes of this document, the following terms and definitions apply.

2.1.1
orbit lifetime
elapsed time between the orbiting satellite’s initial or reference position and orbit demise/reentry

NOTE 1 An example of the orbiting satellite's reference position is the post-mission orbit.

NOTE 2 The orbit’s decay is typically represented by the reduction in perigee and apogee altitudes (or radii) as shown in Figure 1.

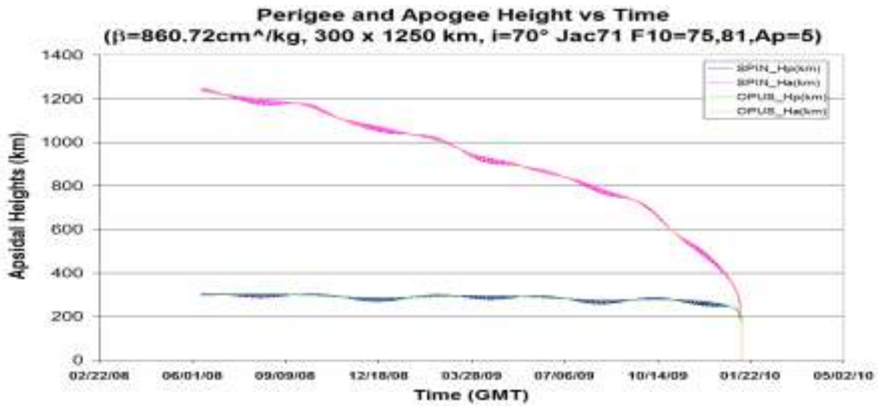

Figure 1: Sample of orbit lifetime decay profile

2.1.2
disposal phase
interval during which a spacecraft or launch vehicle orbital stage conducts its disposal actions. The disposal phase terminates upon the last exercise of control and communication of the space system.

2.1.3
earth equatorial radius
equatorial radius of the Earth

NOTE 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.

2.1.4
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 m2/kg

2.1.4
LEO-crossing orbit
Low-Earth Orbit, defined as an orbit with perigee altitude of 2000 km or less

NOTE As can be seen in Figure A.1, orbits having this definition encompass the majority of the high spatial density spike of satellites and space debris.

2.1.5
long-duration orbit lifetime prediction
orbit lifetime prediction spanning two solar cycles or more (e.g., 25-year orbit lifetime)

2.1.6
mission phase
phase where the space system fulfills its mission, beginning at the end of the launch phase and ending when the space system no longer performs its intended mission/purpose.

2.1.7
post-mission orbit lifetime
duration of the orbit after completion of the mission phase

NOTE The Disposal Phase duration is a component of Post-Mission duration.

2.1.8
satellite
system designed to perform specific tasks or functions in outer space

NOTE A spacecraft that can no longer fulfill its intended mission is considered non-functional. Spacecraft in reserve or standby modes awaiting possible reactivation are considered functional).

2.1.9
space debris
all man-made objects, including fragments and elements thereof, in Earth orbit or re-entering the atmosphere, that are non-functional.

2.1.10
space object
man-made object in outer space.

2.1.11
orbit
path followed by a space object.

2.1.12
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 10.7cm wavelength

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

NOTE 2 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.

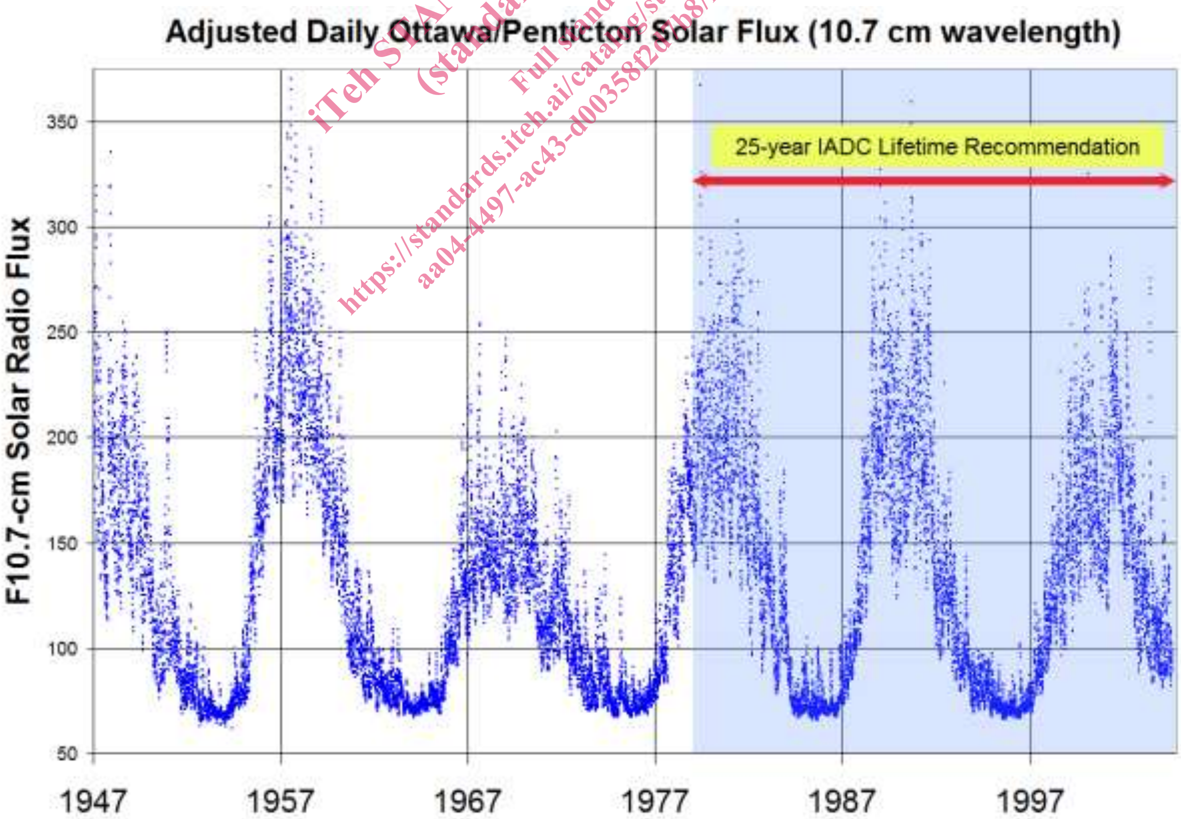

Figure 2: Solar cycle (≈11-year duration)

2.2 Symbols

a	Orbit semi-major axis
A	Satellite cross-sectional area with respect to the relative wind
A_p	Earth daily geomagnetic index
β	Ballistic coefficient of satellite = $C_D * A / m$
C_D	Satellite drag coefficient
C_R	Satellite reflectivity coefficient
e	Orbit eccentricity
$F_{10.7}$	Solar radio flux observed daily at 2800 MHz (10.7 cm) in solar flux units (10^{-22}W m^{-2} Hz^{-1})
$F_{10.7}$ *Bar*	Solar radio flux at 2800 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 satellite
R_e	Equatorial radius of the Earth

2.3 Abbreviated terms

GEO	Geosynchronous Earth Orbit
GTO	Geosynchronous Transfer Orbit
IADC	Inter-Agency Space Debris Coordination Committee
ISO	International Organization for Standardization
LEO	Low Earth Orbit
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).
STSC	Scientific and Technical Subcommittee of the Committee
UNCOPUOS	United Nations Committee on the Peaceful Uses of Outer Space

3 Orbit lifetime estimation

3.1 General requirements

The orbital lifetime of LEO-crossing mission-related objects shall be estimated using the processes specified in this document. 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 (1) initial orbit selection; (2) satellite vehicle design; (3) spacecraft launch and early orbit concepts of operation which minimize LEO-crossing objects; (4) satellite ballistic parameter modifications at EOL; and (5) satellite deorbit maneuvers.

3.2 Definition of orbit lifetime estimation process

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

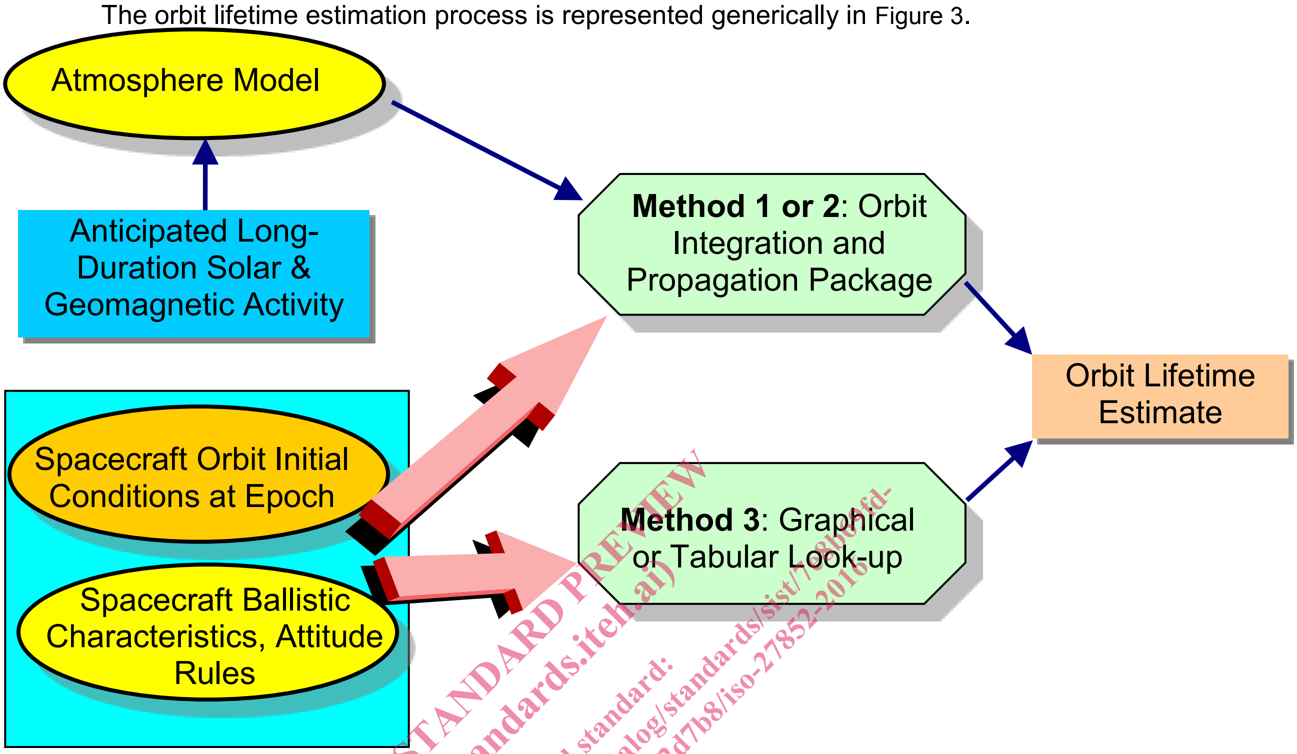

Figure 3: Orbit lifetime estimation process[4]

4 Orbit lifetime estimation methods and applicability

4.1 General

There are three basic analysis methods used to estimate orbit lifetime[4], 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 satellite as shown in Figure 4.

	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, Equation Fit**
Apogee<2000 km	No	No	No margin req'd	5% margin	10% margin	25% margin
Apogee<2000 km	No	Yes	No margin; use SRP	5% margin ; use SRP	10% margin IFF $C_r \approx 1.7$	N/A
Apogee<2000 km	Yes	No	No margin req'd	5% margin	N/A	N/A
Apogee<2000 km	Yes	Yes	No margin req'd; use SRP	5% margin ; use SRP	N/A	N/A
Apogee>2000 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*)

Figure 4: Applicable method with mandated conservative margins of error (in percent) and required perturbation modelling

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 satellite ballistic coefficient model. Method 2 utilizes a definition of mean orbital elements[5, 6], semi-analytic orbit theory and average satellite 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 equations fit to pre-computed orbit lifetime estimation data obtained via the extensive and repetitive application of Methods 1 and/or 2.

4.2 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 satellite 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 is (1) they run much slower than Method 2; (2) many of the detailed data inputs required to make this method realize its full accuracy potential are simply unavailable; and (3) 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.

4.3 Method 2: Rapid semi-analytical orbit propagation

Method 2 analysis tools utilize semi-analytic propagation of mean orbit elements[5, 6] 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 second versus 1700 seconds 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.

4.4 Method 3: Numerical Table Look-Up, Analysis and Fit Equation Evaluations

In this final method, one uses tables, graphs and equations representing data that was generated by exhaustively using Methods 1 & 2 (see above). The graphs and equations provided in this 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 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 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 ten-percent margin of error to account for table look-up interpolation errors. When using graphs and equations, the analyst shall impose a 25% margin of error.

4.5 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 percent[4]. This maximum difference (500 days) translates into a 5% error which can be corrected by knowing the local time of the orbit. As a result, Method 1 or 2 analyses of the actual sun-synchronous orbit condition shall be used when estimating the lifetime of Sun-synchronous orbits.

4.6 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 (ref. 30 and 31) 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 Ref. 30.

The following test case illustrates the complex dynamical properties of GTO. Initial parameters are provided in Figure 5:

Perigee altitude	200 km
Apogee altitude	GEO altitude
Inclination	2°
Area to mass ratio	5e-3 m²/kg
Solar Activity	Constant (F10.7=140 sfu Ap=15)
Drag coefficient	Constant = 2.2
Reflectivity coefficient	Constant = 2

Figure 5: GTO Initial Conditions for the Monte-Carlo simulation

Figure 6 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 0 to 2π. A total of 2500 different initial conditions were generated.

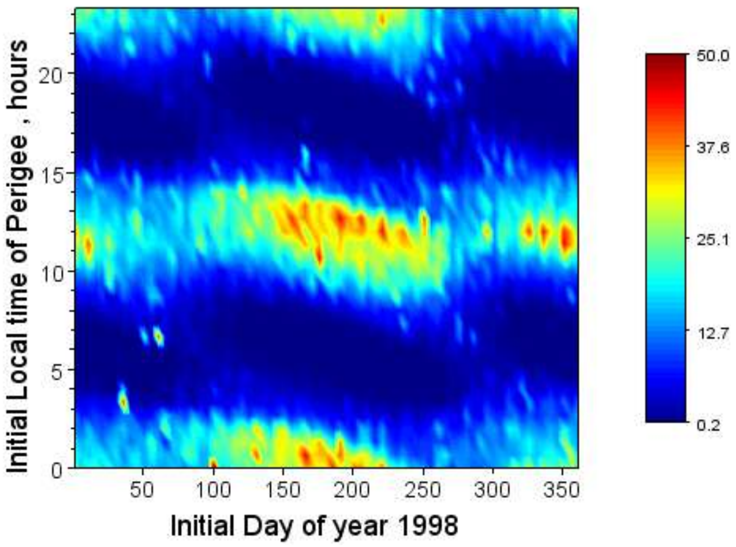

Figure 6: Lifetime variations with respect to initial date and local time of perigee (year)

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 Ref. 31 and 38) are illustrated here: the longest lifetimes are obtained for initial Sun-pointing (12h local time) or anti sun-pointing (24h 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 7h, 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 7 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.

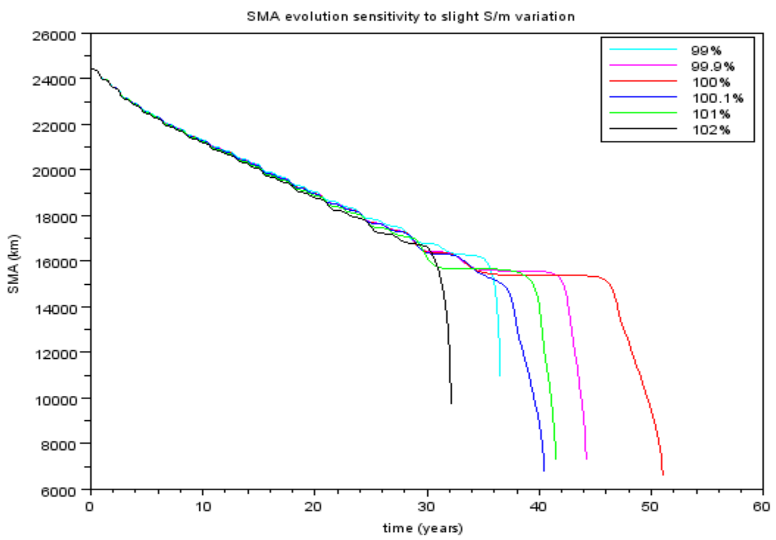

Figure 7: SMA evolution sensitivity to slight A/m variations (from 0.1 to 2%)