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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 procedures used to develop this document and those intended for its further maintenance are described in the ISO/IEC Directives, Part 1. In particular, the different approval criteria needed for the different types of ISO document should be noted. This document was drafted in accordance with the editorial rules of the ISO/IEC Directives, Part 2 (see www.iso.org/directives).

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

This third edition cancels and replaces the second (ISO 27852:2016) edition, which has been technically revised.

The main changes are as follows:

— clarified that this document does not apply to non-LEO protected regions (e.g. GEO);

— harmonized terms and definitions with those in ISO 24113;

— updated to harmonize with IADC [1] and United Nations [2] –[3] guidelines;

— added a subclause on the use of the recommended solar forcing dataset for the Coupled Model Intercomparison Project 6.

Any feedback or questions on this document should be directed to the user's national standards body. A complete listing of these bodies can be found at www.iso.org/members.html.

Introduction

Constraining estimated orbit lifetime of human-made objects is increasingly important as space debris continues to increase (as documented in Annex A) and as such is one of the central tenets of the global space debris mitigation strategy. This document is a supporting document to ISO 24113, its derivative spacecraft disposal standard ISO 23312 and launch vehicle upper stage disposal technical report ISO/TR 20590. The purpose of this document is to provide a common, consensus-based approach to determining orbit lifetime, one that is sufficiently precise and easily implemented for the purpose of demonstrating conformity with ISO 24113. This document offers standardized guidance and analysis methods to estimate orbital lifetime for all LEO-crossing orbit classes. This document only deals with orbit lifetime issues (orbit decay out of orbits crossing the LEO protected region); for other important requirements related to how long a space object will, or will not, cross or occupy a protected region, the user is directed to ISO 24113 and its derivative ISO 23312.

Space systems — Estimation of orbit lifetime

1 Scope

This document describes a process for the long-duration orbit lifetime prediction of orbit lifetime for spacecraft, launch vehicles, upper stages and associated debris in LEO-crossing orbits after mission phase (including any mission lifetime extensions).

The document also clarifies:

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 are referred to in the text in such a way that some or all of their content constitutes requirements of this document. 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 following terms and definitions apply.

ISO and IEC maintain terminology databases for use in standardization at the following addresses:

— ISO Online browsing platform: available at https://www.iso.org/obp

— IEC Electropedia: available at https://www.electropedia.org/

3.1.1
disposal
actions performed by a *spacecraft* (3.1.22) or *launch vehicle orbital stage* (3.1.9) to permanently reduce its chance of accidental break-up and to achieve its required long-term clearance of the *protected regions* (3.1.17)

Note 1 to entry: Actions can include removing stored energy and performing post-mission orbital manoeuvres.

3.1.2
disposal phase
interval between the *end of mission* (3.1.5) of a *spacecraft* (3.1.22) or *launch vehicle orbital stage* (3.1.9) and its *end of life* (3.1.4)

3.1.3
Earth orbit
bound or unbound Keplerian *orbit* (3.1.14) with Earth at a focal point, or Lagrange point orbit which includes Earth as one of the two main bodies

3.1.4
end of life
instant when a *spacecraft* (3.1.22) or *launch vehicle orbital stage* (3.1.9):

a) is permanently turned off, nominally as it completes its *disposal phase* (3.1.2),

b) completes its manoeuvres to perform a *controlled re-entry* (3.1.18) into the Earth’s atmosphere, or

c) can no longer be controlled by the operator

3.1.5
end of mission
instant when a *spacecraft* (3.1.22) or *launch vehicle orbital stage* (3.1.9):

a) completes the tasks or functions for which it has been designed, other than its *disposal* (3.1.1),

b) becomes incapable of accomplishing its *mission* (3.1.12), or

c) has its *mission* (3.1.12) permanently halted through a voluntary decision

3.1.6
GEO
Earth orbit (3.1.3) having zero inclination, zero eccentricity, and an orbital period equal to the Earth's sidereal rotation period

3.1.7
high area-to-mass
having a ratio of area to mass exceeding 0,1 m^2/kg

3.1.8
launch vehicle
DEPRECATED: launcher
system designed to transport one or more payloads into outer space

3.1.9
launch vehicle orbital stage
complete element of a *launch vehicle* (3.1.8) that is designed to deliver a defined thrust during a dedicated phase of the launch vehicle’s operation and achieve *orbit* (3.1.14)

Note 1 to entry: Non-propulsive elements of a launch vehicle, such as jettisonable tanks, multiple payload structures or dispensers, are considered to be part of a launch vehicle orbital stage while they are attached.

3.1.10
LEO-crossing orbit
orbit (3.1.14) having perigee within the LEO protected zone, i.e. with perigee altitude of 2 000 km or less

Note 1 to entry: As shown in Figure A.3, orbits having this definition encompass the majority of the high spatial density spike of *spacecraft* (3.1.22) and *space debris* (3.1.20).

3.1.11
long-duration orbit lifetime prediction
orbit lifetime (3.1.15) prediction spanning two *solar cycles* (3.1.19) or more (e.g. 25-year orbit lifetime)

3.1.12
mission
set of tasks or functions to be accomplished by a *spacecraft* (3.1.22) or *launch vehicle orbital stage* (3.1.19), other than its *disposal* (3.1.1)

3.1.13
mission phase
phase where the space system fulfils its *mission* (3.1.12), beginning at the end of the launch phase and ending when the space system no longer performs its intended mission or purpose

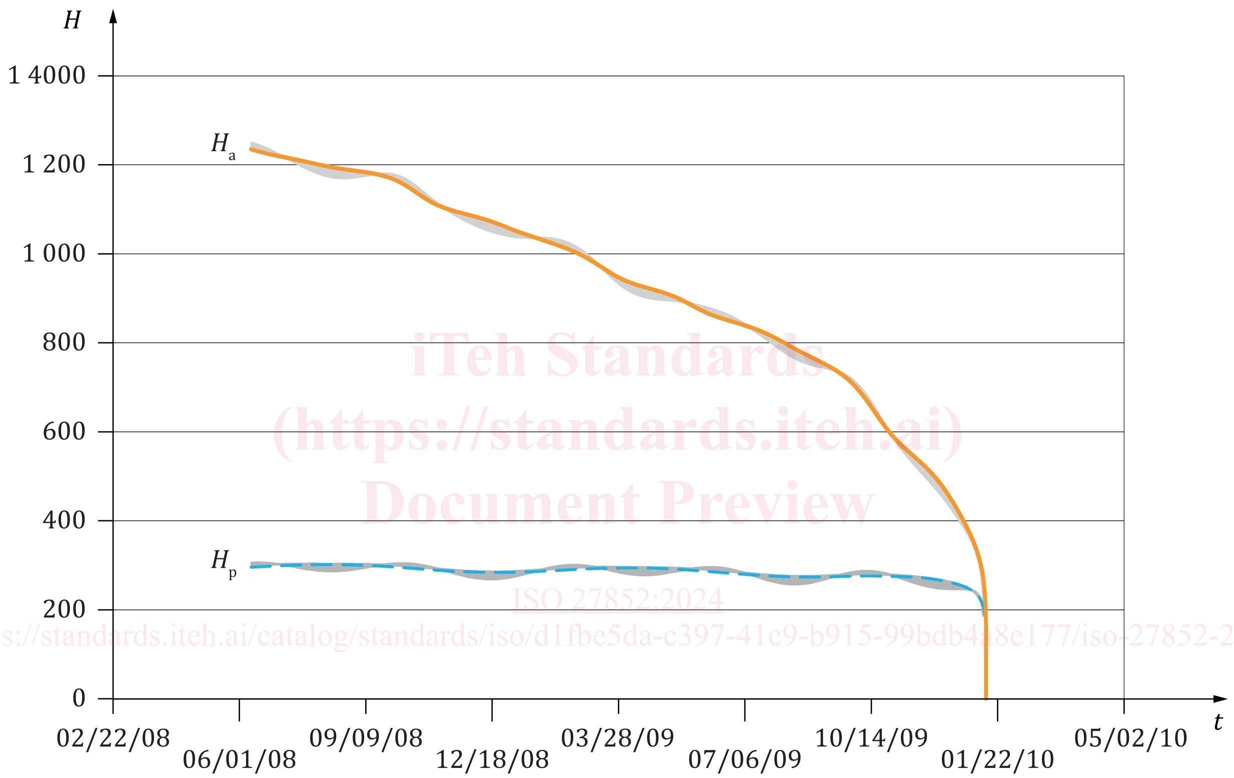
3.1.14
orbit
regular recurring path that a *space object* (3.1.21) takes about its primary attracting body

3.1.15
orbit lifetime
elapsed time between the orbiting *spacecraft's* (3.1.22) initial or reference position and its *re-entry* (3.1.18)

Note 1 to entry: Examples of "initial position" are the injection into *orbit* (3.1.14) of a spacecraft or *launch vehicle orbital stage* (3.1.9), or the instant when *space debris* (3.1.20) is generated. An example of a "reference position" is the orbit of a spacecraft or launch vehicle orbital stage at the *end of mission* (3.1.5).

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.

Note 3 to entry: Ballistic flight re-entry typically begins at 25 km to 50 km altitude.



Key

t time, expressed as calendar date

H_a apogee height, expressed in km

H_p perigee height, expressed in km

Figure 1 — Sample of orbit lifetime perigee and apogee decay profile

3.1.16
post-mission orbit lifetime
duration of the *orbit* (3.1.14) after completion of the *mission phase* (3.1.13)

Note 1 to entry: The *disposal phase* (3.1.2) duration is a component of the post-mission duration.

3.1.17
protected region
region in outer space that is protected with regard to the generation of *space debris* (3.1.20) to ensure its safe and sustainable use in the future

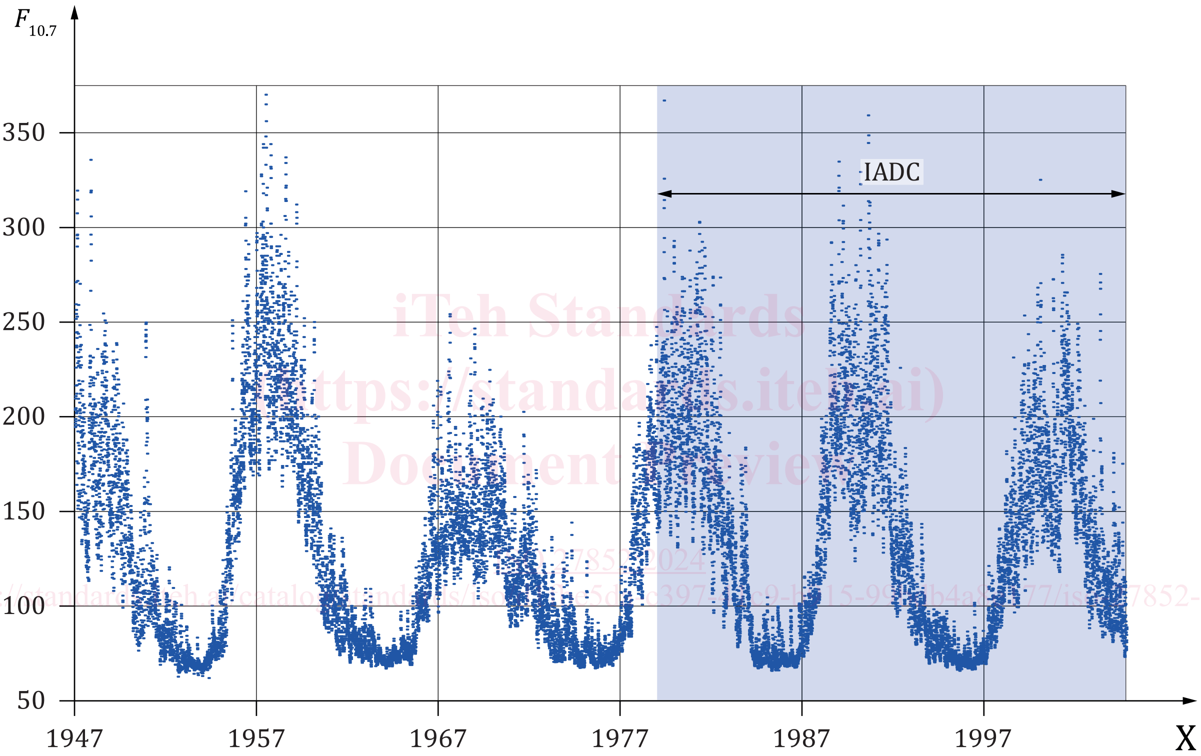
3.1.18
re-entry
permanent return of a *space object* (3.1.21) into the Earth's atmosphere

Note 1 to entry: Several alternative definitions are available for the delineation of a boundary between the Earth's atmosphere and outer space.

3.1.19
solar cycle
≈11-year time period which encompasses the 13-month oscillatory variation of solar radio flux, as observed by monthly sunspot number and highly correlated with the 13-month running mean of measurements taken 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* (3.1.16) 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.



Key

X year

$F_{10.7}$ adjusted daily Ottawa/Penticton solar radio flux (at 10,7 cm wavelength)

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

3.1.20
space debris
DEPRECATED: orbital debris
objects of human origin in *Earth orbit* (3.1.3) or re-entering the atmosphere, including fragments and elements thereof, that no longer serve a useful purpose

Note 1 to entry: *Spacecraft* (3.1.22) in reserve or standby modes awaiting possible reactivation are considered to serve a useful purpose.

3.1.21
space object
object of human origin which has reached outer space

3.1.22
spacecraft
system designed to perform a set of tasks or functions in outer space, excluding *launch vehicle* (3.1.8)

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
β	ballistic coefficient of spacecraft = $C_D * A / m$
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

GEO	geosynchronous Earth orbit
GTO	geosynchronous transfer orbit
LEO	low Earth orbit
RAAN	orbit right ascension of the ascending node (angle between vernal equinox and orbit ascending node, measured counter-clockwise in equatorial plane, looking in the –Z direction of the chosen inertial frame)
SRP	solar radiation pressure

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 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:

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 end of life;

f) spacecraft deorbit manoeuvres.

4.2 Definition of orbit lifetime estimation process

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

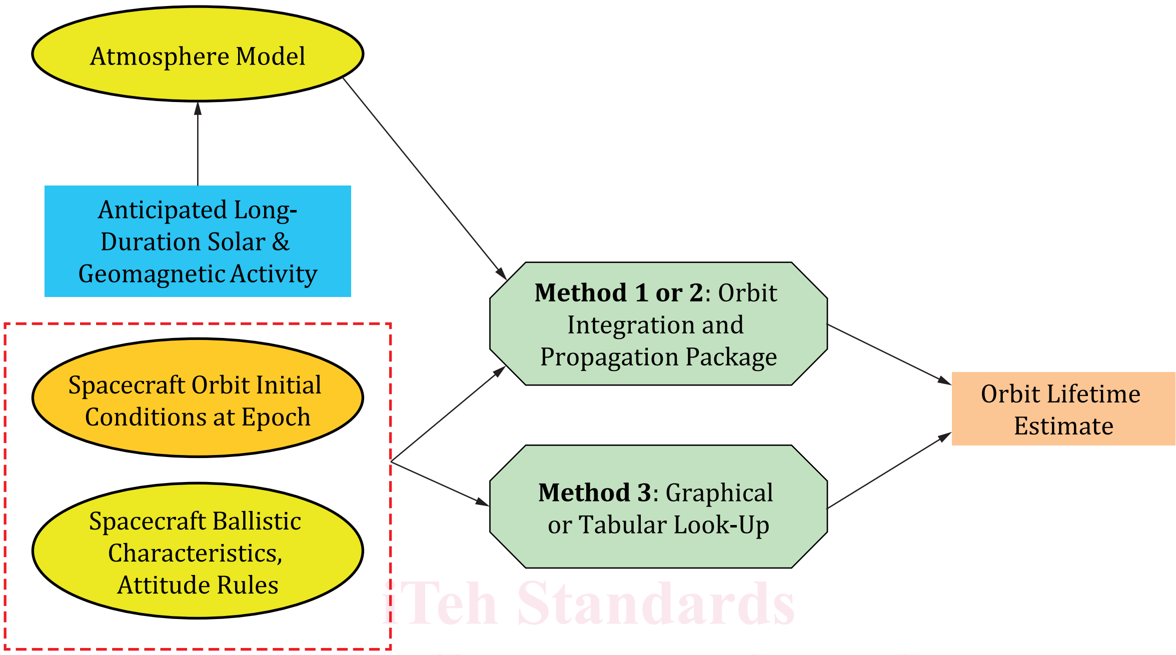


Figure 3 — Orbit lifetime estimation process[6]

5 Orbit lifetime estimation methods and applicability

5.1 General

There are three basic analysis methods used to perform a long-duration orbit lifetime prediction (ISO 24113), as depicted in Figure 1. 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 per cent) 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, equation fit**
Apogee < 2 000 km	No	No	Use β; no margin required	Use β; 5 % margin	Use β; 10 % margin	Use β; 25 % margin
Apogee < 2 000 km	No	Yes	Use β and SRP; no margin required	Use β and SRP; 5 % margin	Use β; 10 % margin	N/A
Apogee < 2 000 km	Yes	No	Use β; no margin required	Use β and SRP; 5 % margin	N/A	N/A
Apogee < 2 000 km	Yes	Yes	Use β and SRP; no margin required	Use β and SRP; 5 % margin	N/A	N/A
Apogee > 2 000 km	Either	Either	Use β and SRP and 3Bdy; no margin required	Use β and SRP and 3Bdy; 5 % margin	N/A	N/A

Key

N/A not applicable

β satellite ballistic coefficient

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,[7] - [8] 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 equations fit to pre-computed orbit lifetime estimation data obtained via the extensive and repetitive application of methods 1 or 2, or both.

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;

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 or 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. GTO) or other resonant orbits, Sun and Moon third-body perturbations and solar radiation pressure effects shall also be modelled.

5.3 Method 2: rapid semi-analytical orbit propagation

Method 2 analysis tools utilize semi-analytic propagation of mean orbit elements [7] - [8] 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 equation evaluations

In this final method, one uses tables, graphs and equations representing data that was generated by exhaustively using methods 1 and 2 (see 5.2 and 5.3). Graphs and equations provided in this document, along with other table lookup, analysis, and fit equations, can help the analyst crudely estimate orbit lifetime for their case of interest, permitting 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 document 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. 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.

5.5 Orbit lifetime sensitivity to Sun-synchronous orbit conditions

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 %.[6] 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, with a 5 % error margin required for the semi-analytic approach.

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

For high-eccentricity orbits (particularly 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.[9],[10] 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 [11]-[12] 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 [11] and [13].

A 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°
Area to mass ratio	5e-3 m^2/kg
Solar activity	Constant ($F_{10.7}$=140 sfu A_p=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 0 to 2π. A total of 2 500 different initial conditions were generated.