INTERNATIONAL STANDARD

ISO 27852

First edition
2011-07-15

Space systems — Estimation of orbit lifetime

Systèmes spatiaux — Estimation de la durée de vie en orbite

Reference number
ISO 27852:2011(E)

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.

Attention is drawn to the possibility that some of the elements of this document may be the subject of patent rights. ISO shall not be held responsible for identifying any or all such patent rights.

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

Introduction

A spacecraft is exposed to the risk of collision with orbital debris and operational satellites throughout its launch, early orbit and mission phases. This risk is especially high during passage through or operations within the LEO region.

To address these concerns, the Inter-Agency Space Debris Coordination Committee (IADC) recommended to the United Nations[2] (section 5.3.2 "Objects Passing Through the LEO Region"): "Whenever possible space systems that are terminating their operational phases in orbits that pass through the LEO region, or have the potential to interfere with the LEO region, should be de-orbited (direct re-entry is preferred) or where appropriate manoeuvred into an orbit with a reduced lifetime. Retrieval is also a disposal option." and "A space system should be left in an orbit in which, using an accepted nominal projection for solar activity, atmospheric drag will limit the orbital lifetime after completion of operations. A study on the effect of post-mission orbital lifetime limitation on collision rate and debris population growth has been performed by the IADC. This IADC and some other studies and a number of existing national guidelines have found 25 years to be a reasonable and appropriate lifetime limit."

The Scientific and Technical Subcommittee (STSC) of the United Nations Committee on the Peaceful Uses of Outer Space (UNCOPUOS), acknowledging the benefits of the IADC guidelines, established the Working Group on Space Debris to develop a set of recommended guidelines[3] based on the technical content and the basic definitions of the IADC space debris mitigation guidelines, taking into consideration the United Nations treaties and principles on outer space. Consistent with the IADC recommendations (listed above), STSC Guideline 6 states that space mission planners, designers, manufacturers and operators should "Limit the long-term presence of spacecraft and launch vehicle orbital stages in the low-Earth orbit (LEO) region after the end of their mission." STSC guidelines also state, "For more in-depth descriptions and recommendations pertaining to space debris mitigation measures, Member States and international organizations can refer to the latest version of the IADC space debris mitigation guidelines and other supporting documents, which can be found on the IADC website (www.iadc-online.org)."

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 IADC guidelines. This International Standard offers standardized guidance and analysis methods to estimate orbital lifetime for all LEO-crossing orbit classes.

This International Standard is a supporting document to ISO 24113[1] and the GEO and LEO disposal standards that are derived from ISO 24113.

Space systems — Estimation of orbit lifetime

IMPORTANT — The electronic file of this document contains colours which are considered to be useful for the correct understanding of the document. Users should therefore consider printing this document using a colour printer.

1 Scope

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

It also clarifies the following:

— modelling approaches and resources for solar and geomagnetic activity modelling;

— resources for atmosphere model selection;

— 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.

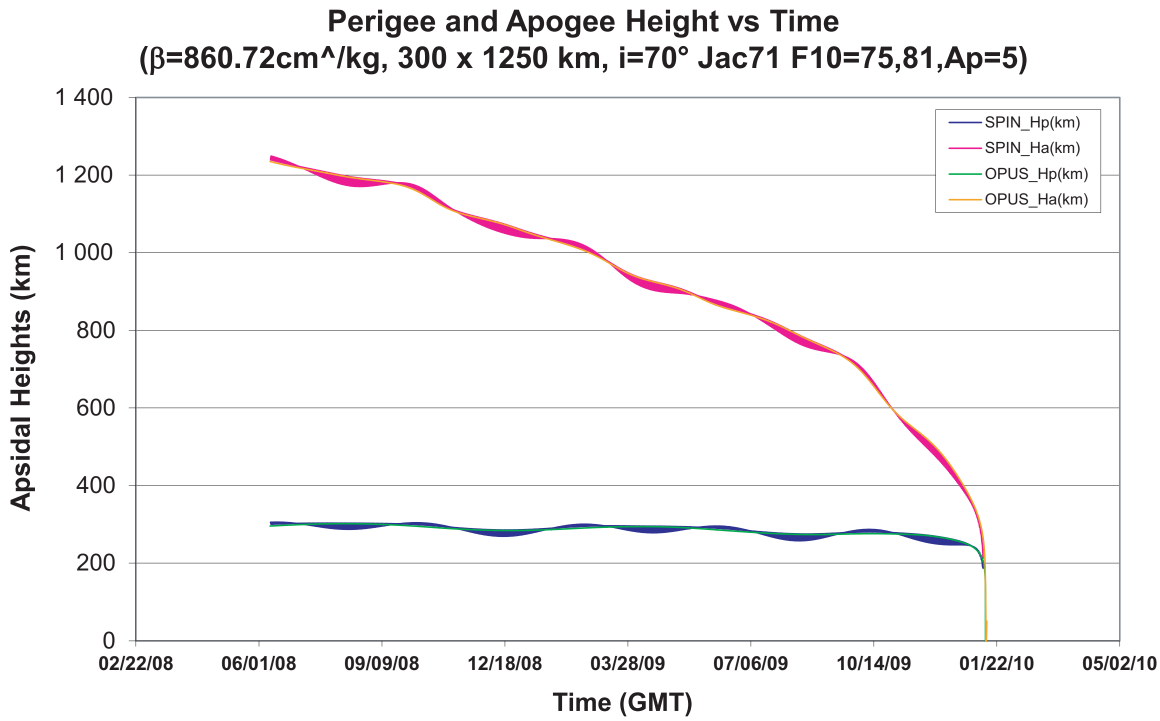

Figure 1 — Sample of orbit lifetime decay profile

2.1.2
disposal phase
interval during which a spacecraft or launch vehicle orbital stage completes its disposal actions

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
LEO-crossing orbit
low-earth orbit, defined as an orbit with perigee altitude of 2 000 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

NOTE Begins at the end of the launch phase and ends at the beginning of the disposal phase.

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 as 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 10,7 cm wavelength

NOTE 1 Historical records back to the earliest recorded data (1947) are shown in Figure 2.

NOTE 2 For reference, the current 25-year post-mission IADC orbit lifetime recommendation is overlaid onto the historical data; it can be seen that multiple solar cycles are encapsulated by this long time duration.

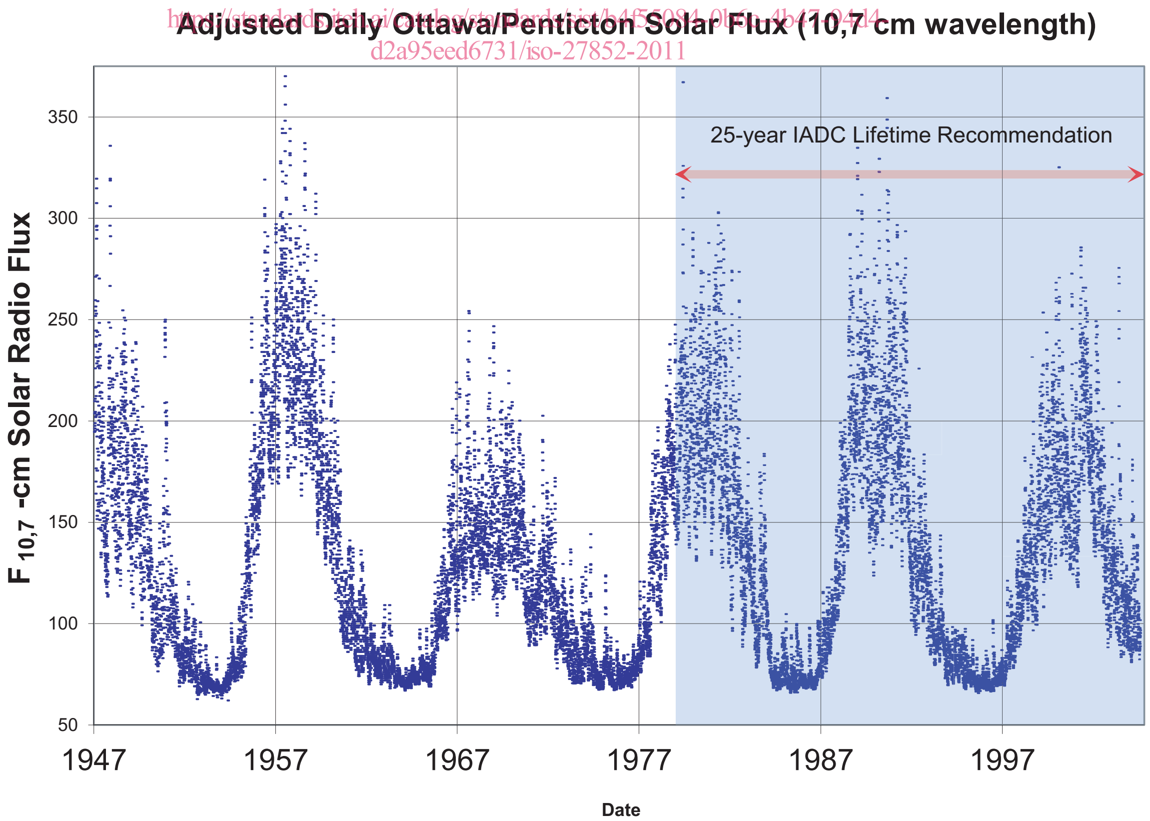

Figure 2 — Solar cycle (≈11 year duration)

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, equal to $C_{\mathrm{D}} \times A/m$
C_{D}	satellite drag coefficient
C_{R}	satellite 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})
$\bar{F}_{10,7}$	solar radio flux at 2 800 MHz (10,7 cm), averaged over three solar rotations
H_{a}	apogee altitude, equal to $a(1 + e) - R_{\mathrm{e}}$
H_{p}	perigee altitude, equal to $a(1 - e) - R_{\mathrm{e}}$
m	mass of a 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 (the angle between the vernal equinox and the orbit ascending node, measured CCW in the equatorial plane, looking in the –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 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 as per IADC recommendations using a combination of the following:

a) initial orbit selection;

b) satellite vehicle design;

c) spacecraft launch and early orbit concepts of operation which minimize LEO-crossing objects;

d) satellite ballistic parameter modifications at EOL;

e) satellite deorbit manoeuvres.

3.2 Definition of orbit lifetime estimation process

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

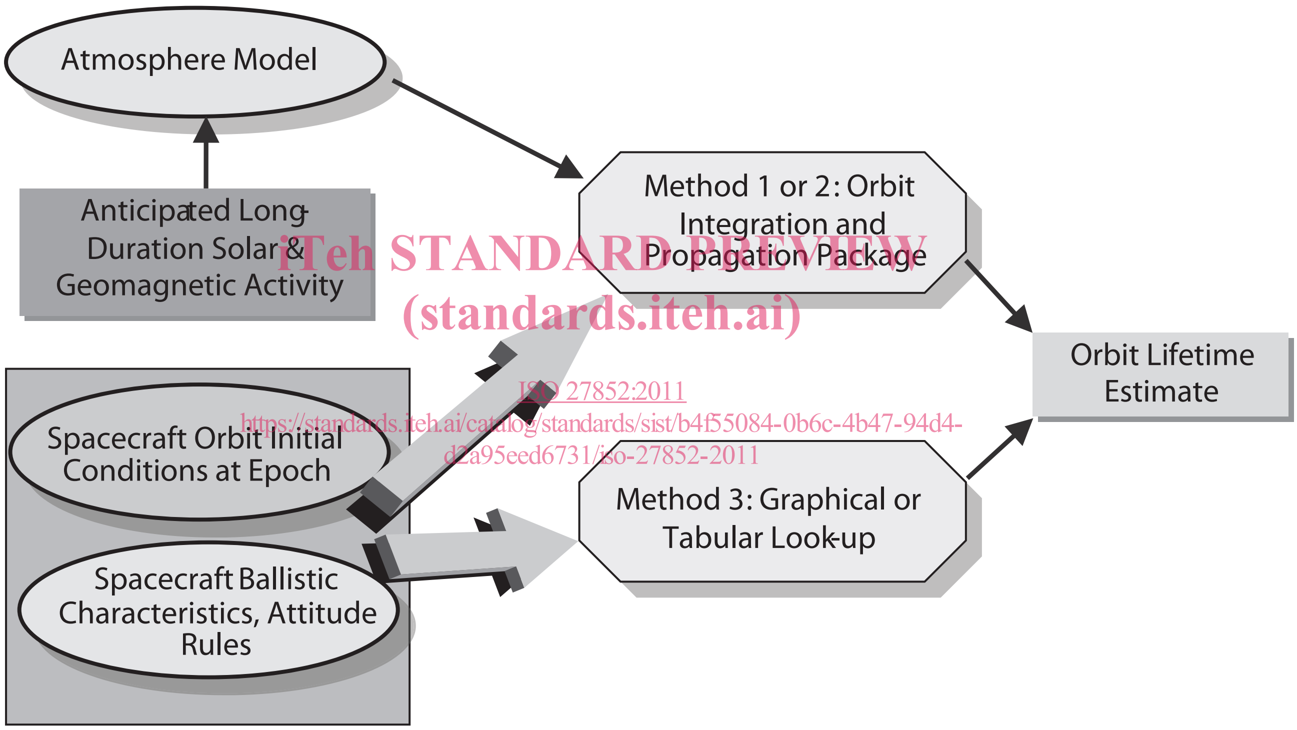

Figure 3 — Orbit lifetime estimation process[4]

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 Table 1.

Table 1 — Applicable method with mandated conservative margins of error and required perturbation modelling

Orbit apogee altitude km	Special orbit:		Conservative margin applied to each method:			
	Sun-sync?	High area-to-mass?	Method 1: Numerical integration	Method 2: Semi-analytic	Method 3a: Table look-up	Method 3b: Graph, equation fit
Apogee < 2 000	No	No	No margin req'd	5 % margin	10 % margin	25 % margin
Apogee < 2 000	No	Yes	No margin; use SRP	5 % margin; use SRP	10 % margin IFF $C_r \approx 1{,}7$	N/A
Apogee < 2 000	Yes	No	No margin req'd	5 % margin	N/A	N/A
Apogee < 2 000	Yes	Yes	No margin req'd; use SRP	5 % margin; use SRP	N/A	N/A
Apogee > 2000	Either	Either	No margin req'd; use 3Bdy+SRP	5 % margin; use 3 Bdy + 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 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 a very rapid integration of the equations of motion, while still retaining reasonable accuracy. methods 3a and 3b are 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 are the following.

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/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 average ballistic coefficient. In the case of high apogee orbits (e.g. geosynchronous transfer orbits), sun and moon third-body perturbations 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

