CNES (FRENCH SPACE AGENCY)

TECHNICAL AND DIGITAL DIRECTORATE
FLIGHT DYNAMICS SUB-DIRECTORATE
ORBITOGRAPHY DEPARTMENT

STELA

STELA-NT-ETUDES-363-CNES
Issue : 01 Date : 03/04/2024
Revision : 00 Date : 03/04/2024

Ref.: intentionally left blank
Distribution code : E

TECHNICAL NOTE STELA TOOL USER GUIDE FOR SATELLITE APPLICATIONS

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INDEX SHEET

Confidentiality:	KEY WORDS: STELA, ORBIT PROPAGATION, RE-ENTRY, FSOA

DOCUMENT TITLE:
STELA TOOL USER GUIDE FOR SATELLITE APPLICATIONS

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ABSTRACT: The purpose of this STELA user guide is to provide operators subject to the French Space Operations Act with a set of recommendations for implementing a long-term orbit propagation calculation in order to verify compliance with the requirements of the Technical Regulations.

ASSOCIATED DOCUMENTS: This document stands on its own.	Tracking :

Volume: 1	TOTAL NO. OF PAGES: 29 of which cover PAGES: 6 NO. OF ADDITIONAL PAGES : 0	COMPOSITE DOCUMENT: N	Language : EN

CONF. MANAGEMENT : No	MANAGER CONF. MGMNT. :

REASON FOR CHANGE: Document creation

CONTRACT: None

Host system:

INTERNAL DISTRIBUTION

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EXTERNAL DISTRIBUTION

Name	Acronym	Comments

CHANGES LOG

Iss.	Rev.	Date	Reference, Author(s), Reasons for change
01	00	03/04/2024	Document created based on the *Best Practice Guide for Controlling a Space Object - Ed4.0 05/02/2014 - LOS-GR-CNF-8-CNES* Grégoire Laur DOA/SME/LOS Fabien Gachet DTN/DV/OR Etienne Montagnon DTN/DV/MS2

CONTENTS

1. GLOSSARY AND LIST OF TBC & TBD PARAMETERS

GEO	Geostationary Orbit
GTO	Geostationary Transfer Orbit
FSOA	French Space Operations Act (see LOS in French)
LEO	Low Earth Orbit
LOS	Loi Opérations Spatiales (FSOA in English)
MEO	Medium Earth Orbit
RR	Random Re-Entry
TR	Technical Regulations (FSOA)
STELA	Semi Analytic Tool for End of Life Analysis

List of TBC parameters:

List of TBD parameters:

2. GENERAL

2.1. REFERENCE DOCUMENTS

DR1	USER MANUAL - STELA *(PDF supplied with the software)*
DR2	Lamy et al, "Resonance Effects on lifetime of Low Earth Orbit Satellites", 23rd ISSFD,2012
DR3	Fraysse, H. et al "Long term orbit propagation techniques developed in the frame of the French Space Act". 22nd ISSFD, 2011
DR4	Lamy, A. et al. "Analysis of Geostationary Transfer Orbit Long-Term Evolution and Lifetime". 22nd ISSFD, 2011
DR5	Morand, V. et al. "Dynamical properties of Geostationary Transfer Orbits over long time scales: consequences for mission analysis and lifetime estimation" AIAA, 2012
DR6	Le Fèvre, C. et al. "Compliance of disposal orbits with the French Space Act: the Good Practices and the STELA tool", 63rd IAC, 2012

2.2. APPLICABLE DOCUMENTS

3. PURPOSE OF THE DOCUMENT

The purpose of this note is to provide a set of recommendations for implementing a long-term orbit propagation calculation for a space object as part of a request for technical compliance under the French Space Operations Act.

To ensure that this document is sufficiently self-contained, it will include some information on using STELA but does not replace the User Manual provided with the software (see DR1).

Recommendations or comments will be incorporated into the main text.

4. BASIC PRINCIPLES FOR USING THE STELA SOFTWARE

The STELA (Semi analytic Tool for End of Life Analysis) software is the tool used by CNES to verify the long-term evolution of orbits within the framework of the French Space Operation Act (FSOA). The User Manual (DR1) gives a more detailed description of its model, its functionalities and its scope of use.

> Using the latest version of STELA and the parameters recommended in this document (atmospheric model, Cx, "equivalent constant" solar activity, etc.) to extrapolate the orbit **ensures compliance with the TR**.
>
> **The latest version of the tool recommended for FSOA use is available on the following website: *https://www.connectbycnes.fr/en/fsoa***

Figure 4-1: STELA logo

STELA is an orbit propagator based on semi-analytic integration of the space object centre of gravity movement equations. The dynamics equations were averaged to retain only medium and long-term effects of perturbations on the orbital parameters. These equations can therefore be integrated with a large time step (greater than the orbital period) to gain in calculation time. The main short periods are then analytically added to compute the osculating parameters for the required dates.

The dynamic model used is adapted to each type of orbit. It can therefore be different for LEO, GEO and GTO orbits.

STELA integrates the averaged parameters, in coherence with the dynamical model used.

STELA notably produces a summary file of the orbit extrapolation performed (file in "_sim.txt" format containing the description of inputs, outputs and calculation parameters, and indicating the compliance with FSOA criteria).

The content of this "_sim.txt" file must be given in the FSOA file in addition to the corresponding file in "_sim.xml" format.

STELA also contains iterative modules to help determine the end-of-life orbit and a tool to determine the cross sectional mean area. If the latter is used, the corresponding descriptor file ("_shap.xml" format file) must be provided.

5. LONG-TERM ORBIT PROPAGATION MODELLING

5.1. FOREWORD

The aim of this section is to provide the necessary elements of orbital mechanics so that the operator can process their FSOA file without ambiguity about the recommended definitions, conventions and standards.

These rudiments are used in particular in the mission analysis which forms part of the FSOA file submitted by the operator and which includes:

- The identification of the launcher injection orbit and the strategy adopted to reach the operational orbit,
- The identification of the desired end-of-life orbit and the manoeuvring strategy implemented to reach this orbit,
- The ΔV (orbital velocity increments) and propellant budget over the operational life of the satellite, including provisions for anti-collision manoeuvres and disposal.

In addition, these concepts are necessary for the use of the various tools recommended by CNES so that the operator can meet the various requirements of the FSOA: STELA for calculating orbital propagation, DEBRISK and ELECTRA for estimating lethal risks due to re-entering fragments of space objects.

Lastly, a great deal of attention is paid to the propagation of the end-of-life orbit to check compliance with criteria relating to atmospheric re-entry time or passage through one of the protected areas (LEO or GEO). These checks can be carried out using the STELA tool, which implements the practices in this document. This can be used to simplify the description of hypotheses and methods used.

5.2. DEFINITIONS

5.2.1.1. Description of orbit parameters

An orbit will be defined by its orbit parameters on a given date. These parameters will be described, specifying the reference frame, the time scale associated with the date, their type and their nature.

5.2.1.2. Time scales

Below are the main time scales used:

TAI: International Atomic Time

UTC: Coordinated Universal Time

TU1 or UT1: Universal Time

TT: Terrestrial Time

Where:

- TT = TAI + 37.184 sec. (2021)
- UTC = TAI - (seconds integer)

The integer number of seconds is given by the IERS (e.g. 37 seconds on 01/01/2021).
The UTC-UT1 variable discrepancy is given by the IERS. It is such that |UTC – UT1| < 0.9 seconds.
This discrepancy is negligible in the framework of these analyses.

5.2.1.3. Reference frames

State vectors will be expressed in one of the following reference frames:

- ICRF (International Celestial Reference Frame),
- GCRF (Geocentric Celestial Reference Frame),
- EME2000 (Earth Mean Equator at J2000 reference Epoch),
- CIRF (Celestial Intermediate Reference Frame),
- TIRF (Terrestrial Intermediate Reference Frame)

The IERS Technical note No.36 "IERS CONVENTIONS (2010)" contains the definition of these reference frames, in chapter 5 in particular.

5.2.1.4. Types of vector

State vectors will be expressed in one of the following formats:

- Date, x, y, z, vx, vy, vz
- Date, a, e, i, Ω, ω, M,
- Date, a, e, i, MLTAN, ω, M,
- Date, ha, hp, i, Ω, ω, M
- Date, a, e.cos(ω), e.sin(ω), i, Ω, ω+M,
- Date, a, e.cos($\omega+\Omega$), e.sin($\omega+\Omega$), sin(i/2).cos(Ω),sin(i/2).sin(Ω), $\Omega+\omega$+M,

Where:

- a: semi-major axis
- e: eccentricity
- i: inclination
- Ω: right ascension of ascending node
- ω: argument of perigee
- M: mean anomaly
- ha: apogee geocentric altitude (apogee radius – Earth radius)
- hp: perigee geocentric altitude (perigee radius – Earth radius)
- MLTAN: mean local time of the ascending node (mean in the sense of the time equation):

$$MLTAN = UT1 + \frac{24}{360} Long$$

- UT1 in hours,
- Long: East longitude of the orbit's ascending node (in degrees).

5.2.1.5. Nature of vector

It will be specified whether the state vector corresponds to:

- osculating orbit: translation into Keplerian parameters of the instantaneous position-velocity at the vector date,
- averaged orbit: osculating orbit from which the effects of periods strictly below the orbital period are removed.

NB: in the following paragraphs, when the position of the space object at time t is referred to, this is the position that corresponds to the osculating orbit at this time.

NB: STELA takes these two types of vector into account.

5.2.1.6. Constants and physical parameters

Physical constant values used by STELA:

- Earth radius used to check FSOA criteria and calculate apogee and perigee altitudes: *Rearth* = 6378 km
- Potential model, Earth radius and Earth oblateness coefficients used to calculate the dynamics and the geodetic altitude for the atmospheric model, the state vector type conversions: those of the Grim5-S1 model
- Astronomical unit (AU): 1.49598022291 10^{11} m
- Solar radiation pressure at 1UA: 0.45605 10^{-5} N/m^2
- The Sun's gravitational constant: 1.32712440018 10^{20} $m^3s^{-2}kg^{-1}$
- The Moon's gravitational constant: 4.9027779 10^{12} $m^3s^{-2}kg^{-1}$

- Using STELA **<u>and its default settings</u>** (atmospheric model, Cx, "equivalent constant" solar activity, etc.) to extrapolate the orbit **ensures compliance with the TR**.

5.3. LOW ORBITS

The following paragraphs describe the methods and hypotheses that can be used to calculate:

- Atmospheric re-entry time for a space object in Low Earth Orbit,
- The long-term evolution of an orbit 'above' area A, to check compliance with the criterion of non-penetration into this protected area.

Any orbit with an apogee and perigee falling within the LEO protected area (area A), *or in the vicinity of the latter* and likely to penetrate it, is considered to be in the low orbit range.

The *atmospheric re-entry* instant is considered to be reached when the object's geocentric altitude becomes less than **120 km**. The corresponding criterion is:

Vector radius (t) < Rearth + 120 km

Where Vector radius (t) = distance between Earth center and the space object at time t.

In addition, the area A penetration criterion is:

- First time t for a given period, i.e.:

Vector radius (t) < Rearth + (2000+h) km

Where Vector radius (t) = distance between Earth center and the space object at time t.

The value of the margin "h" depends on simplifications taken into account in the dynamic modelling in relation to a complete forces model. The value taken into account by STELA is specified in its User Manual.

Orbit evolution and the computed atmospheric re-entry time depend on initial conditions, the dynamic modelling used to propagate the orbit, and the object characteristics (drag area, mass, aerodynamic drag coefficient). The following paragraphs specify the recommended dynamic modelling and the associated entry parameters.

5.3.1. Dynamic model for long-term orbit extrapolation

At least the following forces must be taken into account:

- Earth's gravitational force: effect of Earth potential zonal terms up to degree 4. For orbits with an inclination in the vicinity of the "critical" inclination (63.4° and 116.6° approx.) additional modelling is required. A good level of accuracy is obtained by taking into account the effect of zonal terms up to degree 7. Additional modelling is also required for resonant orbits as described below. A good level of accuracy is obtained by taking into account the effect of zonal terms up to degree 15.

- The atmospheric drag force,

- The gravitational force of the Moon and the Sun,

- The Solar Radiation Pressure (SRP) force.

The luni-solar gravitational potential has little impact on re-entry time except for sun-synchronous or quasi-sun-synchronous end-of-life orbits (approximately up to one year), orbits near the upper limit of the LEO area, the most eccentric orbits, and in some cases resonant orbits for which the secular drift of angles due to J2 (Earth oblateness) causes a coupling with the luni-solar perturbations which cause a secular drift of the eccentricity (effect on the re-entry time up to more than ten years). The most sensitive cases occur when the secular drift of angles caused by J2 results in the following type of relations:

$$\dot{\Omega} - \dot{\alpha_{L/S}} \pm \dot{\omega} \approx 0, \quad \dot{\Omega} - 2\dot{\alpha_{L/S}} + 2\dot{\omega} \approx 0, \quad \dot{\Omega} \pm 2\dot{\omega} \approx 0$$

where $\dot{\alpha_{L/S}}$ is the apparent speed of rotation of the Moon (index L) or the Sun (index S) around the Earth

$$(\dot{\alpha_L} = \frac{2\pi}{27.3}\, rad/day \quad \text{and} \quad \dot{\alpha_S} = \frac{2\pi}{365.25}\, rad/day)$$

These conditions are fulfilled for some low orbits in the [40.80] deg and [110.130] deg inclination range. The evolution of these orbits is therefore sensitive to the initial right ascension of the node, of the argument of perigee and the right ascension of the sun (see DR2).

NB: "critical" inclination occurs when the secular drift of the argument of perigee due to Earth potential zonal terms is zero or practically zero, which corresponds to the condition: $\dot{\omega} \approx 0$

5.3.2. Calculation parameters for long-term orbit extrapolation

5.3.2.1. Atmospheric drag force

This chapter describes the practices applied to determine the parameters used to calculate the atmospheric drag force (aerodynamic force Fa):

$$\overrightarrow{Fa} = -\frac{1}{2}\rho S C_x V_r \vec{V_r}$$

Where:

- ρ: atmospheric density
- S: drag cross-section (surface projected onto a perpendicular plane at V_r)
- C_x: aerodynamic drag coefficient
- V_r: velocity of the spacecraft in relation to the atmosphere

5.3.2.2. Atmospheric model

The use of the MSIS00 atmospheric model is recommended to calculate the atmospheric density. Any other atmospheric model used will be specified and justified.

Solar activity and geomagnetic index:

Solar activity greatly influences the atmospheric re-entry time of low orbits. With prediction of level of solar cycles and their "timing" (duration, phasing over time) being uncertain over several decades, a calculation principle of atmospheric re-entry time based on an "**equivalent constant solar activity**" was defined (constant magnetic indices and fluxes with respect to time).

For a satellite defined by its surface, its Cx and its mass, and for a given orbit, this standardisation principle is used to render the result of the atmospheric re-entry time calculation independent of the effective end-of-life date in relation to solar cycles.

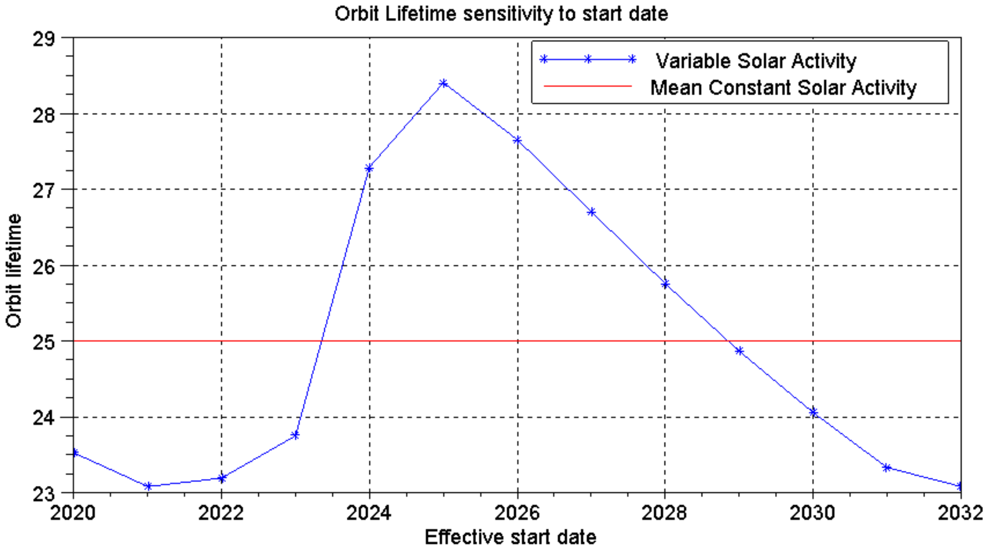


Figure 5-1: Orbit Lifetime sensitivity to start date

Its level was set based on measured past solar activity (last six solar cycles) and using a statistical method, to obtain for all space objects applying this method an atmospheric re-entry time equal to 25 years on average (see DR3).

This means that for a given satellite and orbit (set to re-enter in 25 years with the average solar activity level), if we calculate the re-entry times by statistically varying future solar activity (in magnitude and in timing by randomly selecting a sequence of n cycles from the last six), we should obtain a distribution function for this time such as the probability P (time < 25 years) = 50%. This function would have a form such as that below:

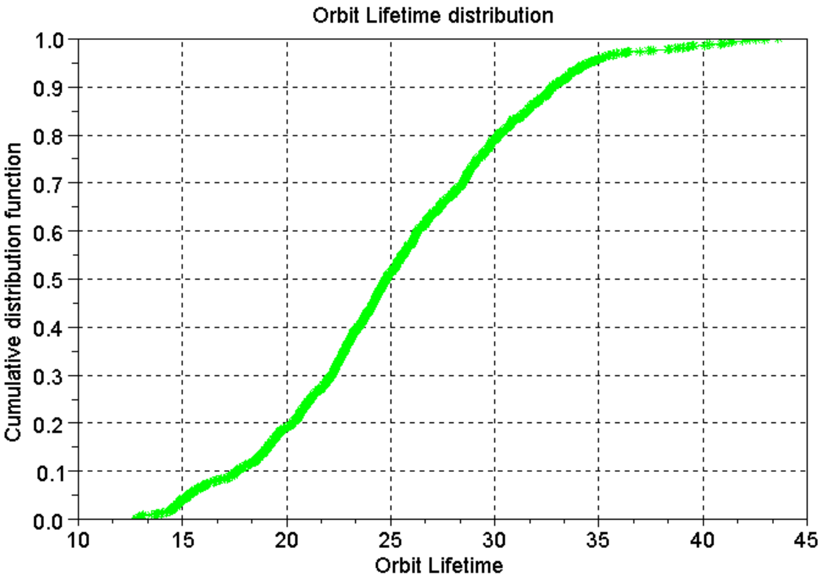


Figure 5-2: Orbital Lifetime distribution

The level of solar activity and the geomagnetic index will therefore be fixed by using "equivalent constant" solar activity (with respect to time) defined by the following expressions based on the ballistic coefficient and the initial altitude of the apogee of the end-of-life medium Earth orbit:

$$AP = 15$$

$$F_{10.7} = 194.4 + 3.17\ln\left(\frac{SC_x}{m}\right) - 6.86\ln(ha)$$

- AP: geomagnetic index
- $F_{10.7}$: solar flux in sfu
- SCx/m: ballistic coefficient in m^2/kg
- ha: apogee radius of the initial medium Earth orbit – Earth radius in km
- ln: Napierian logarithm

NB: if Cx varies as a function of the altitude, $F_{10.7}$ is calculated where Cx = 2.2.

The following graph represents the above function for different S/m values:

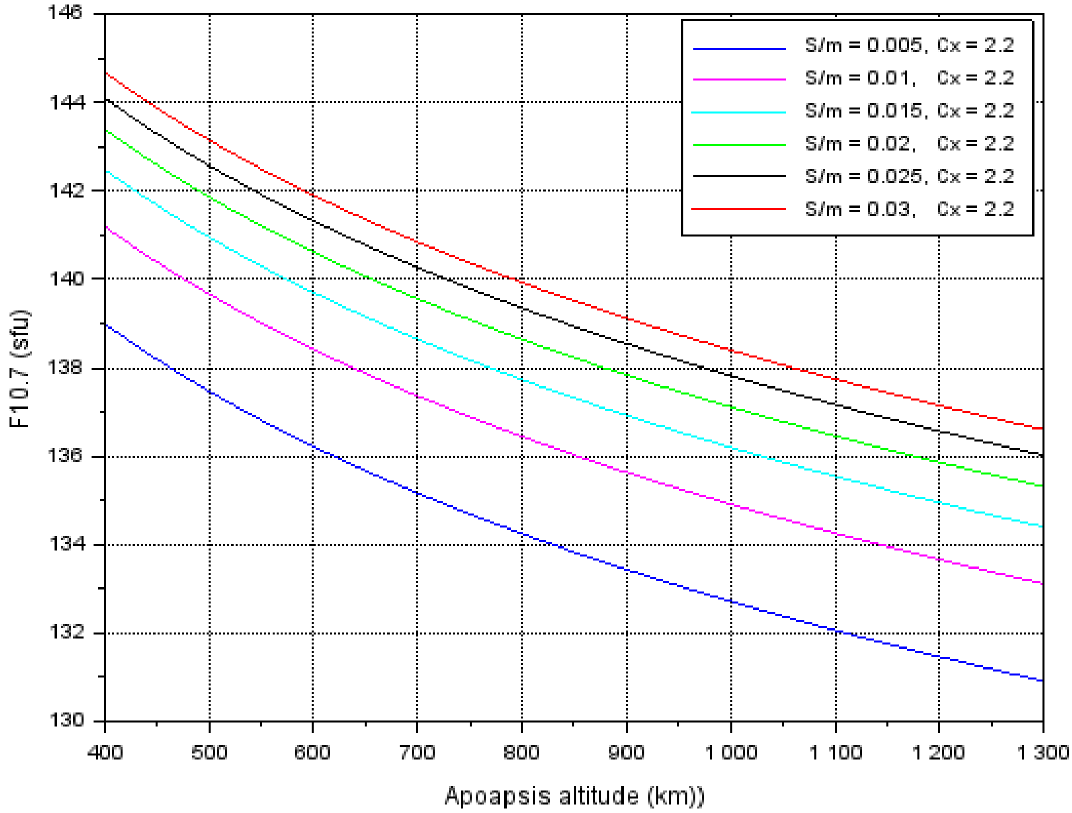


Figure 5-3: Equivalent constant solar flux value as a function of S/m

These parameters will be entered in the atmospheric model using the following configuration:

- Solar flux (instantaneous and averaged) equal to $F_{10.7}$,
- Three-hourly geomagnetic indices all equal to AP.

This approach therefore practically negates (excluding orbit with an inclination in the previously defined "sensitive" orbit range and which is resonant) the dependence of the atmospheric re-entry time on the disposal date for a satellite and a given orbit (orbit defined by its apogee, perigee, inclination and local time for sun-synchronous orbits).

Dependence on the initial date therefore remains very low (of around one month in 25 years). This can be managed by a margin of 1% over the intended re-entry time (nominally 25 years) when choosing the end-of-life orbit.

NB: the influence of initial local time on the atmospheric re-entry time is only perceptible for quasi-sun-synchronous end-of-life orbits (of around ± one year), and for orbits with an inclination in the previously defined "sensitive" orbits range and that are resonant.

The use of an "equivalent constant" solar activity is recommended for all long-term propagation and must be definitive for the entire operation.

5.3.2.3. Drag area

The drag cross-section will be taken into account based on an average area of constant value for the orbit extrapolation time. This area will be calculated taking into account the probable attitude of the spacecraft:

- If a stable equilibrium attitude of the spacecraft is possible (aerodynamic balance, gravity gradient stabilisation, inertial pointing, etc.), this will be described and used to determine the drag area.
- In other cases, it can be considered that the spacecraft's attitude in relation to the direction of the airspeed will be arbitrary and variable.

The dimensions of the spacecraft taken into account for this calculation will be supplied. Any simplifications made to the geometry of the object to calculate this area must be indicated and made to minimise the surface.

If the utility programme included in STELA is used to calculate the average area, the descriptor file must be supplied with the satellite drawing in the three axes.

5.3.2.4. Atmospheric drag coefficient Cx

The values taken into account (constant or variable) for the atmospheric drag coefficient Cx must be indicated and justified.

For "conventionally" shaped space objects (geometry based on flat plates, sphere), use of the typical value of Cx dependant on the altitude shown below is recommended. This produces the following graph:

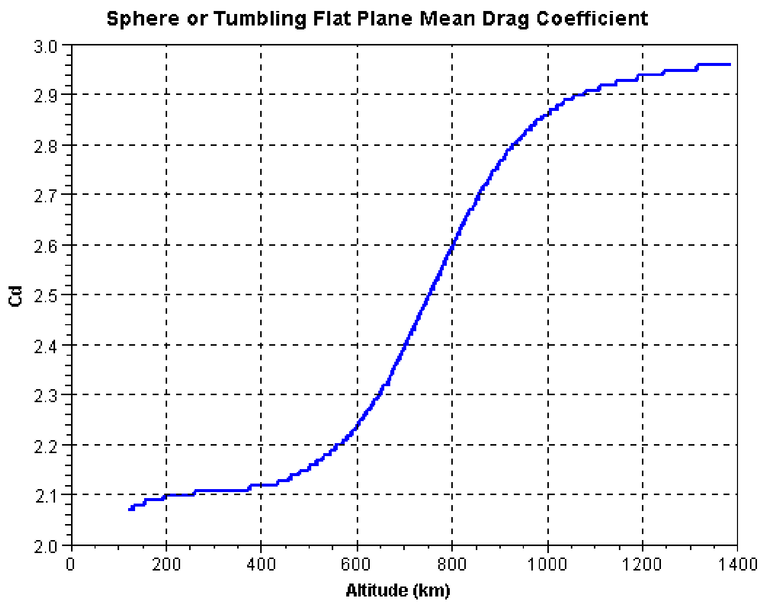


Figure 5-4: Cx as a function of geodetic altitude

For streamlined aerodynamic shaped space objects, a constant or variable Cx value as a function of the altitude will be proposed and justified.

NB: if, taking into account the equilibrium attitude and the aerodynamic shape, it would be relevant to also consider an aerodynamic lift force, this will be taken into account when calculating the atmospheric re-entry time and characterised by a proposed and justified lift coefficient.

5.3.2.5. Solar radiation pressure force

This chapter describes the practices applied to determine the parameters used to calculate the solar radiation pressure force (not within eclipse periods) expressed as follows:

$$\overrightarrow{F_p} = C_R P_0 \, S \left(\frac{d_0}{d}\right)^2 \vec{u}$$

Where:

- $\vec{u}$ sun-satellite direction unit vector
- d satellite-sun distance
- d_0 average Earth-sun distance = 1UA
- P_0 solar radiation pressure at distance d_0
- S satellite's reflecting area (surface projected onto a perpendicular plane at u)
- C_R reflectivity coefficient.

5.3.2.6. Reflecting area

The reflecting area will be taken into account based on an average area of constant value for the orbit extrapolation time. This area will be calculated taking into account the probable attitude of the spacecraft:
- if a stable equilibrium attitude of the craft is possible (balance, inertial pointing, etc.), this will be described and used to determine the average reflecting area,
- in other cases, it will be considered that the spacecraft's attitude in relation to the direction of the sun will be arbitrary and variable.

The dimensions of the spacecraft taken into account for this calculation will be supplied. Any simplifications made to the geometry of the object to calculate this area must be indicated and made to maximise the area.

If the utility programme included in STELA is used to calculate the average area, the descriptor file must be supplied with the satellite drawing in the three axes.

5.3.2.7. Reflectivity coefficient

This reflectivity coefficient C_R may vary by 1 (completely absorbent surface) to 2 (completely reflecting surface).

The reflectivity coefficient value used will be provided and justified. By default, this can be taken at 1 to be conservative, excluding orbits with an inclination in the previously defined "sensitive" orbits range and that are resonant.

5.4. GEOSTATIONARY OR QUASI-GEOSTATIONARY ORBITS

Reminder: The disposal orbit for satellites operating in the GEO orbit must be selected above the protected area B.

The following paragraphs describe the methods and hypotheses that can be used to calculate the long-term evolution of an orbit in the vicinity of the geostationary arc and to check compliance with the criterion of non-penetration into the GEO protected area.

To select a disposal orbit that complies with the criterion of non-penetration into the protected area B, it is also possible to use the formula given in ISO 26872 "Space systems - Disposal of satellites operating at geosynchronous altitude" as long as the eccentricity of this orbit is less than 0.003 and its inclination is less than 20 deg.

Any orbit with an apogee and perigee falling within the GEO protected area, or in the vicinity of the latter and likely to penetrate it, is considered to be in the GEO orbit range.

The radius to be taken into account for the geostationary orbit altitude is equal to **42,164 km.**

The area B penetration criterion, allowing a margin of "h" km to take into account any inaccuracy of the dynamic modelling, is as follows:

- First time t for a given period, i.e.:

<u>42164 – (200+h) km < Vector radius (t) < 42164 + (200+h) km</u>

where Vector radius (t) = distance between Earth center and the space object at time t.

and

-15 deg < geocentric latitude (t) < 15 deg

Orbit evolution and in particular apogee and perigee altitudes depend on initial conditions, the dynamic modelling used to propagate the orbit, the object characteristics (reflecting surface, mass, reflectivity coefficient). The following paragraphs specify the dynamic modelling deemed necessary and the associated entry parameters.

The value of the margin "h" depends on simplifications taken into account in the dynamic modelling in relation to a complete forces model. The value taken into account by STELA is specified in its User Manual.

5.4.1. Dynamic model for long-term orbit extrapolation

At least the following forces must be taken into account:
- the Earth's gravitational force: complete model up to degree 4 and scale 4
- the Solar Radiation Pressure (SRP) force.
- the gravitational force of the Moon and the Sun.

NB: the luni-solar gravitational potential and the Solar Radiation Pressure effect significantly influences the orbit's evolution, in particular the evolution of its eccentricity. Based on this, the orbit's evolution will largely depend on the disposal date and the choice of initial eccentricity vector.

5.4.2. Calculation parameters for long-term orbit extrapolation

This paragraph describes the practices applied to determine the parameters used to calculate the solar radiation pressure force expressed as follows:

$$\overrightarrow{F_p} = C_R P_0 \, S \left(\frac{d_0}{d}\right)^2 \vec{u}$$

Where:

- $\vec{u}$ sun-satellite direction unit vector
- d satellite-sun distance
- d_0 average Earth-sun distance = 1UA
- P_0 solar radiation pressure at distance d_0
- S satellite's reflecting surface (surface projected onto a perpendicular plane at u)
- C_R reflectivity coefficient.

5.4.2.1. Reflecting surface

The reflecting surface will be taken into account based on an average surface of constant value for the orbit extrapolation time. This surface will be calculated taking into account the probable attitude of the spacecraft:
- if a stable equilibrium attitude of the craft is possible (balance, inertial pointing, etc.) this will be described and used to determine the average reflecting surface.
- in other cases, it will be considered that the spacecraft's attitude in relation to the direction of the sun will be arbitrary and variable.

The dimensions of the spacecraft taken into account for this calculation will be supplied. Any simplifications made to the geometry of the object to calculate this surface must be indicated and made to maximise the surface.

If the utility programme included in STELA is used to calculate the average surface, the descriptor file must be supplied with the satellite drawing in the three axes.

5.4.2.2. Reflectivity coefficient

This reflectivity coefficient C_R may vary by 1 (completely absorbent surface) to 2 (completely reflecting surface).

The reflectivity coefficient value used will be provided and justified. In all cases, this value must not be less than 1.5.

5.5. GEOSTATIONARY TRANSFER ORBITS (GTO)

Geostationary Transfer Orbits (GTO): Any orbit with an apogee inside, in the vicinity of or above ("super synchronous" orbits) the GEO protected area and the perigee inside or above the LEO protected area is considered to be in the GTO range. Eccentricity is typically greater than 0.25).

The LEO area re-entry criterion is:

- First time t for a given period, i.e.:

Vector radius (t) < Rearth + (2000+h) km

Where Vector radius (t) = distance between Earth center and the space object at time t.

In addition, the atmospheric re-entry instant is considered to be reached when the object altitude becomes less than **80 km**. The corresponding criterion is:

Vector radius (t) < Rearth + 80 km

Where Vector radius (t) = distance between Earth center and the space object at time t.

NB: this value of 80 km is different from the value used for LEO orbits (120 km), given the different long-term dynamics of these orbits.

The radius to be taken into account for the geostationary orbit altitude is equal to **42,164 km.**

The GEO area exit criterion, allowing a margin of "h" km to take into account any inaccuracy of the dynamic modelling, is as follows:

- Latest time t for a given period, i.e.:

42164 – (200+h) km < Vector radius (t) < 42164 + (200+h) km

Where Vector radius (t) = distance between Earth center and the space object at time t.

and

-15 deg < geocentric latitude (t) < 15 deg

The GEO area penetration criterion, allowing a margin of "h" km to take into account any inaccuracy of the dynamic modelling, is as follows:

- First time t for a given period, i.e.:

42164 – (200+h) km < Vector radius (t) < 42164 + (200+h) km

where Vector radius (t) = distance between Earth center and the space object at time t.

and
-15 deg < geocentric latitude (t) < 15 deg

The value of the margin "h" depends on simplifications taken into account in the dynamic modelling in relation to a complete forces model. The value taken into account by STELA is specified in its User Manual.

Orbit evolution and in particular apogee and perigee altitudes depend on initial conditions, the dynamic modelling used to propagate the orbit, the object characteristics (drag surface, mass, aerodynamic drag coefficient, reflecting surface, reflectivity coefficient).

The specific nature of GTO type orbits is that their evolution can be very sensitive to initial conditions (in particular the positioning of the orbit in relation to the Moon and the Sun) and to perturbing force model parameters (mass, ballistic coefficient, drag coefficient, reflectivity coefficient, solar activity, etc.). This is due to gravitational resonance phenomena created by the pull of the Moon and the Sun (association between the secular drift of angles due to J2 and luni-solar perturbations of the same type as those presented for low orbits but with a more significant effect due to the high eccentricities of the orbits, see DR4 and DR5).

This means in particular that the evolution of the disposal orbit can be very sensitive to the disposal date.

The figure below shows an example of the atmospheric re-entry time calculation in years of a quasi-equatorial GTO orbit as a function of the disposal day in the year and the initial local time of the perigee (key parameter influencing the orbit's evolution). This latter parameter is the angle, expressed in hours (given than 0 deg corresponds to 12 hours), between the direction of the sun and the direction of the perigee projected into the equator.

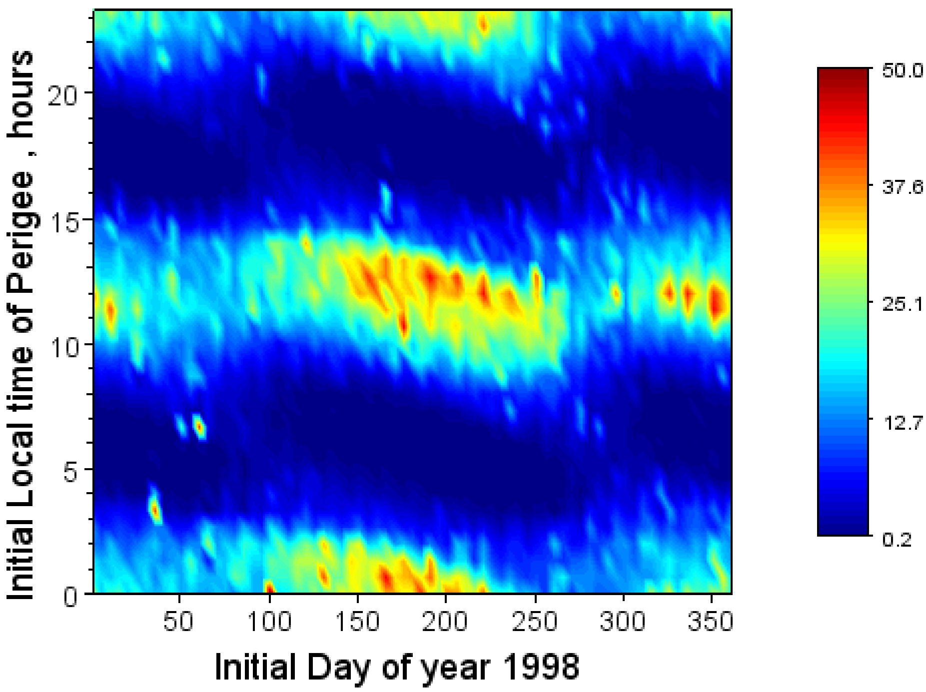


Figure 5-5: Atmospheric re-entry time (years)

The figure below shows an example of evolutions of the semi-major axis (SMA) of a quasi-equatorial GTO orbit at constant initial conditions (orbit parameters and date), but for which the surface over mass S/m ratio value of the associated object has been very slightly modified (by around 1%):

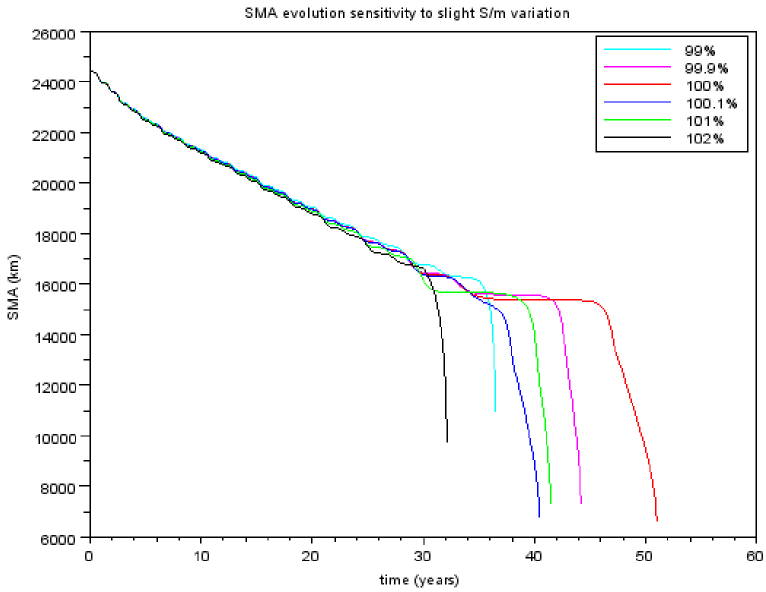


Figure 5-6: Example of an orbit's semi-major axis evolution sensitivity to S/m

This sensitivity to initial conditions and to the configuration of perturbing force models creates a need to statistically calculate the above criteria.

The following paragraphs specify the dynamic modelling required for this calculation, the associated entry parameters and the usable statistic method.

5.5.1. Dynamic model for long-term orbit extrapolation

At least the following forces must be taken into account:

- Earth's gravitational force: effect of Earth potential zonal and tesseral terms up to degree and scale 7. For orbits with an inclination nearing the "critical" inclination (approximately 63.4° and 116.6°) the effect of zonal terms up to degree 7 needs to be taken into account to obtain the required level of accuracy,
- the solar radiation pressure force,
- the atmospheric drag force,
- the gravitational force of the Moon and the Sun.

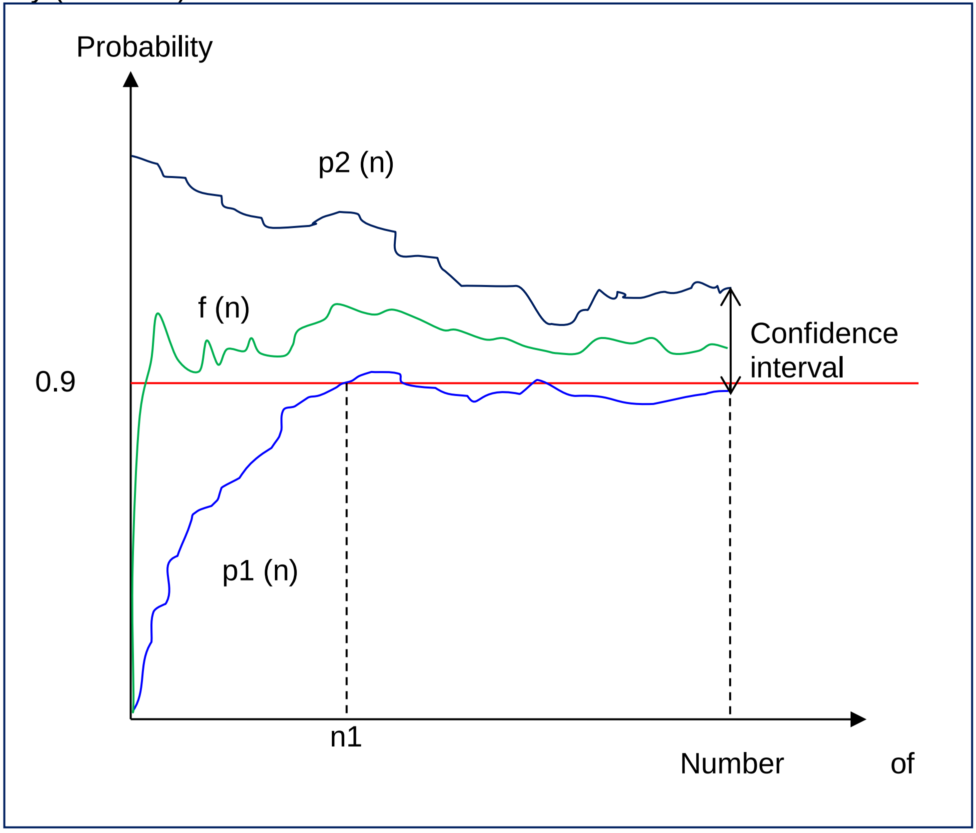
Tesseral terms are taken into account to enable the evolution of some resonant orbits with the Earth's gravitational potential to be better characterised, i.e. those for which the period is commensurable with the Earth's rotation period. The main resonances to be considered are: 2/1 (semi-major axis ≈ 26.560 km), 3/2 (semi-major axis ≈ 32.177 km) and at higher altitude 4/3 (semi-major axis ≈ 34.800 km). n/m signifies that "n orbital revolutions" are made by the spacecraft whilst the Earth spins on its axis m times.

5.5.2. Criteria and Statistical Method

It is considered that a law criterion of type:
- Atmospheric re-entry time < 25 years,
- Non-penetration in the LEO protected area,
- Exit from the GEO protected area in less than one year (launchers only),
- Non-penetration in the GEO protected area,

is checked if the probability of compliance with this criterion is greater than 0.9.

This check is based on a "Monte Carlo" type statistical method which involves randomly dispersing uncertain key parameters (initial orbit parameters, force model parameters) according to a suitable law of probability, to propagate the orbit n times with these dispersed parameters and calculate the probability of occurrence associated with each applicable criterion. This is based on the ratio between the number of propagated orbits complying with the criterion and the total number of propagated orbits (observed probability: f) and a confidence interval associated with the estimation of the statistical parameter to take into account the fact that the statistic is applied to a finite number of random selections (sample). The probabilities corresponding to confidence interval ranges [p1, p2] are compared with the target probability (see DR6).



The confidence interval [p1(n), p2(n)] is calculated based on "Wilson with continuity correction" formulas with a confidence level of 95%: a confidence interval at 95% will provide a correct probability framework that we are attempting to estimate ninety-five times out of one hundred. This means that if this estimation is repeated many times, confirming that the parameter to be estimated (i.e. the actual probability) falls within the confidence interval at 95% each time, an error will occur on average five times out of one hundred. The value of the p1 and p2 ranges of the Wilson confidence interval is expressed by the following formulas:

$$p1 = \frac{2nf + u_{\alpha/2}^{2} - 1 - u_{\alpha/2}\sqrt{u_{\alpha/2}^{2} - 2 - 1/n + 4f(n(1-f)+1)}}{2(n + u_{\alpha/2}^{2})}$$

$$p2 = \frac{2nf + u_{\alpha/2}^{2} + 1 + u_{\alpha/2}\sqrt{u_{\alpha/2}^{2} + 2 - 1/n + 4f(n(1-f)-1)}}{2(n + u_{\alpha/2}^{2})}$$

Where:

- n: number of propagated orbits (number of selections),
- f: "observed" probability: number of orbits complying with the criterion divided by n,
- p1: range below the confidence interval,
- p2: range above the confidence interval,
- $u_{\alpha/2} = \Phi^{-1}(1-\alpha/2) = 1.96$ for a 95% confidence interval (is the normal law distribution function).

For a number n of selections (n propagated orbits):

- if the range below the confidence interval is greater than or equal to 0.9, the criterion is checked (compliant),
- if the range above the confidence interval is less than or equal to 0.9, the criterion is not checked (non-compliant),
- if neither of the confidence interval ranges confirms either of these two conditions, it is not possible to draw a conclusion regarding the criterion check. The number of selections will therefore need to be increased to reduce the breadth of the confidence interval.

A minimum number of selections is required. Therefore (based on this number of selections) for 5% of potentially inaccurate statistical frameworks (Monte Carlo), the error between the observed probability and the actual probability is low.

For a confidence level of 0.95, $n_{min} = 45$.

The following uncertain key parameters must be dispersed:

- Initial conditions (orbit parameters),
- Ballistic coefficient via surface and/or mass of the spacecraft and/or reflectivity coefficient,
- Solar activity and geomagnetic index for orbits subject to atmospheric drag,
- Ballistic coefficient via the surface and/or mass of the spacecraft and/or the drag coefficient for orbits subject to atmospheric drag.

5.5.3. Calculation parameters for long-term orbit extrapolation

5.5.3.1. Atmospheric drag force

DEFINITION OF THE FORCE

See §5.3.2.1 on LEO orbits

ATMOSPHERIC MODEL

See §5.3.2.2 on LEO orbits

SOLAR ACTIVITY AND GEOMAGNETIC INDEX:

Solar activity and the geomagnetic index significantly influence the evolution of GTO orbits crossing the LEO protected area. The predicted level of solar cycles and their "timing" (duration, phasing over time) is uncertain over several decades. The statistical calculation of solar activity and the future geomagnetic index (in level and in timing) can be calculated by randomly selecting a sequence of n cycles from the last six solar cycles (see example of figures below for 5 former cycles), and a random departure day in the first cycle. The data of a cycle is daily flux data ($F_{10.7}$) and three-hourly geomagnetic index data (Ap).

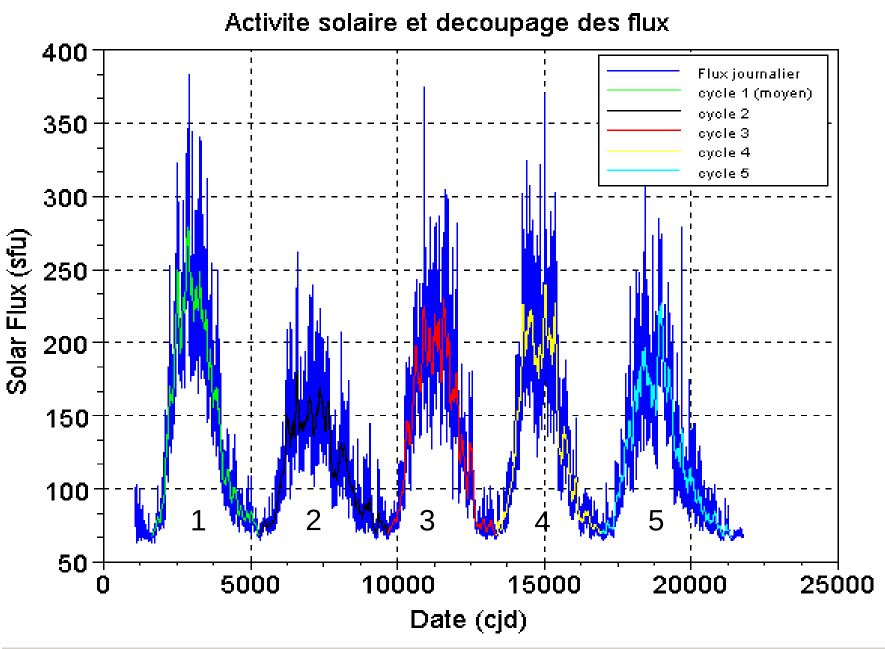


Figure 5-7: Example of solar cycle breakdown - solar fluxes

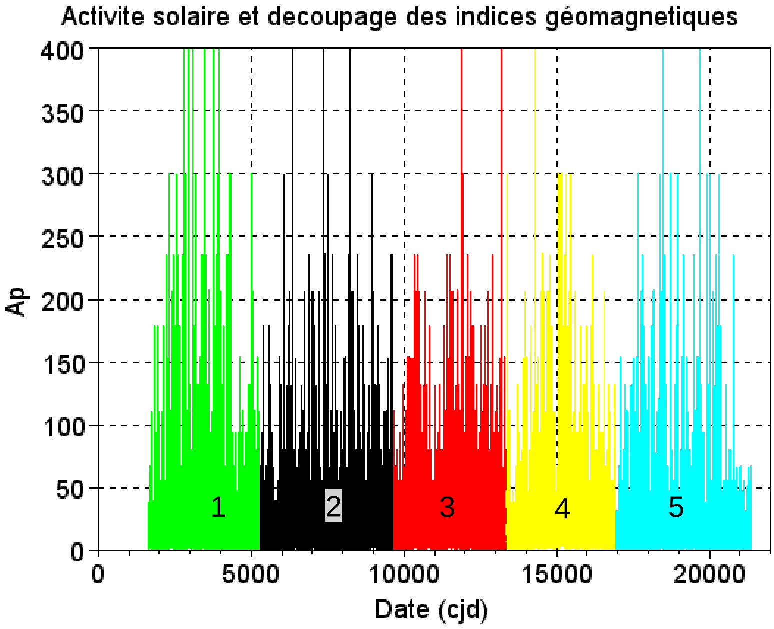


Figure 5-8: Example of solar cycle breakdown - geomagnetic indices

DRAG SURFACE:

See §5.3.2.3 on LEO orbits

ATMOSPHERIC DRAG COEFFICIENT CX:

See §5.3.2.4 on LEO orbits

DISPERSION OF THE DRAG FORCE OVER THE M/(SC$_X$) COEFFICIENT

The drag force is dispersed:

- on the one hand via the dispersion over atmospheric density due to the dispersion of solar activity (solar flux and geomagnetic index)
- on the other hand via dispersions taken into account over the m/(SCx) coefficient.

The m/(SCx) coefficient above must be dispersed to take into account uncertainties over the drag surface, aerodynamic coefficients and mass. This dispersion will be defined in form (uniform, Gaussian) and in range, as a function of the estimated uncertainties.

A uniform uncertainty of +/-20% will at least be used.

5.5.3.2. Solar radiation pressure (SRP) force

DEFINITION OF THE FORCE

See §5.4 on GEO orbits

REFLECTING SURFACE

See §5.4 on GEO orbits

REFLECTIVITY COEFFICIENT

This reflectivity coefficient C_R may vary by 1 (completely absorbent surface) to 2 (completely reflecting surface).

The reflectivity coefficient value used will be provided and justified.

DISPERSION OF THE SOLAR RADIATION PRESSURE FORCE OVER THE M/(SC$_R$) COEFFICIENT

The solar radiation pressure force is dispersed via the dispersions taken into account over the m/(S/C_R) coefficient.

The m/(SC_R) coefficient above must be dispersed to take into account uncertainties over the reflecting area, the reflectivity coefficient and the mass. This dispersion will be defined in form (uniform, Gaussian) and in range, as a function of the estimated uncertainties.

A uniform uncertainty of +/-20% will at least be used.

5.6. MEO ORBITS

The focus here is on the operational orbits between areas A and B (GNSS-type orbits), whose graveyard orbits must guarantee non-interference with areas A and B over a time span of 100 years.

STELA's GTO model, currently the most relevant for MEO orbits, can be used.