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Space systems — Disposal of satellites operating at geosynchronous altitude

Systèmes spatiaux — Élimination des satellites opérant à une altitude géostionnaire

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Contents

Page

Foreword

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Introduction

This International Standard prescribes requirements for planning and executing manoeuvres and operations to remove an operating satellite from geosynchronous orbit at the end of its mission and place it in an orbit for final disposal where it will not pose a future hazard to satellites operating in the geosynchronous ring.

This International Standard includes requirements related to the following:

— when the disposal action needs to be initiated,

— selecting the final disposal orbit,

— executing the disposal action successfully, and

— depleting all energy sources to prevent explosions after disposal.

End-of-mission disposal of an Earth-orbiting satellite broadly means the following:

a) removing the satellite from the region of space where other satellites are operating, so as not to interfere or collide with these other users of space in the future, and

b) ensuring that the disposed object is left in an inert state and is incapable of generating an explosive event that could release debris which might threaten operating satellites[1].

For satellites operating in the geosynchronous belt, the most effective means of disposal is first to re-orbit the satellite to a super-synchronous orbit above the region of operating spacecraft and the manoeuvre corridor used for relocating operating satellites to new longitudinal slots, and then to discharge batteries and vent propellants and take other actions to preclude a debris-producing event.

1) Further information will be provided in the future International Standard, ISO 16127.

Space systems — Disposal of satellites operating at geosynchronous altitude

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1 Scope

This International Standard specifies requirements for the following:

— planning for disposal of satellites operating at geosynchronous altitude to ensure that final disposal is sufficiently characterized and that adequate propellant will be reserved for the manoeuvre;

— selecting final disposal orbits where the satellite will not re-enter the operational region within the next 100 years;

— executing the disposal manoeuvre successfully;

— depleting all energy sources on board the vehicle before the end of its life to minimize the possibility of an event that can produce debris.

This International Standard provides techniques for planning and executing the disposal of space hardware that reflect current internationally accepted guidelines and consider current operational procedures and best practices.

2 Normative references

The following referenced documents are indispensable for the application 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 24113:2010, *Space systems — Space debris mitigation requirements*

3 Terms and definitions

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

3.1
inclination excursion region
region in space occupied either by a non-operational geostationary satellite or by an operational geosynchronous satellite without inclination station-keeping

3.2
re-orbit manoeuvre
action of moving a satellite to a new orbit

3.3
satellite
manufactured object or vehicle intended to orbit the earth, the moon or another celestial body

4 Symbols and abbreviated terms

4.1 Symbols

a semi-major axis

C_R solar radiation pressure coefficient of the spacecraft[2] ($0 < C_R < 2$)

e eccentricity

h_p perigee altitude

i inclination

I_{sp} specific impulse

L_S solar longitude

M mean anomaly

p semilatus rectum or semi-parameter [$p = a(1 - e^2)$]

r radius of orbit

v true anomaly

μ Earth gravitational constant

σ standard deviation or the positive root of the variance, which measures the dispersion of the data

Ω right ascension of ascending node (RAAN)

ω argument of perigee

A/m effective area-to-mass ratio: projected area of the spacecraft perpendicular to the sun's ray divided by the mass of the spacecraft

ΔH change in altitude

ΔV delta velocity or total velocity change

4.2 Abbreviated terms

EGM Earth gravitational model

EOMDP end-of-mission disposal plan

GEO geosynchronous (geostationary) Earth orbit

RAAN right ascension of ascending node

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2) In some references, the C_R is defined as the index of surface reflection.

5 Geosynchronous region

The geosynchronous region is a circular ring around the Earth in the equatorial plane. Within this region, an object in space moves along the ring at a mean angular rate that is equal or very close to the Earth's rotation, meaning that the satellite appears to be positioned over a fixed location on the ground.

Without so-called north-south station-keeping, the inclination of a GEO satellite will gradually cycle between 0° (equatorial orbit) and a maximum of approximately 14,6° and back again. In addition to maintaining the accuracy of its inclination, a GEO satellite must execute station-keeping manoeuvres to maintain longitudinal accuracy, so as to prevent a naturally occurring drift to the east or to the west caused by asymmetries in the Earth's gravitational field, unless the satellite is located at one of the two “gravity wells” on the geostationary arc.

Figure 1 shows a three-dimensional view of the geosynchronous ring with a cross-section defining the approximate size of the ring. Figure 2 gives the dimensions of three regions of the cross-section. The cross-section is defined by two axes: the latitudinal axis and radial axis. This plane of the cross-section is perpendicular to the Earth's equatorial plane.

The three concentric boxes shown in Figure 2 give the approximate boundaries for three types of orbits. The smallest box represents the region where a geostationary satellite will be confined under station-keeping, and the next larger box approximates the region where a geosynchronous satellite may be located when its inclination is not controlled but it remains under a mission-specified value. For example, the upper value for some specific geosynchronous missions may range from 3° to 5° depending on the ground user's antenna design. The largest box represents the inclination excursion region for a non-operational GEO satellite and the ±200 km protected region. For most communication satellites, the longitude station-keeping limit is ±0,1°.

6 Protected region

The GEO protected region, defined by ISO 24113 and indicated by Region B in Figure 1, includes the rectangular toroid centred on geostationary altitude, with an extent 200 km above and below this altitude and with inclination limits of +15° to −15°. While operations are usually conducted within about 75 km of geostationary altitude, the GEO protected region is extended in altitude to create a manoeuvre corridor for relocating spacecraft. Passivation of the disposed spacecraft is necessary to ensure that accidental explosions from on-board energy sources do not create debris that could re-enter the protected region.

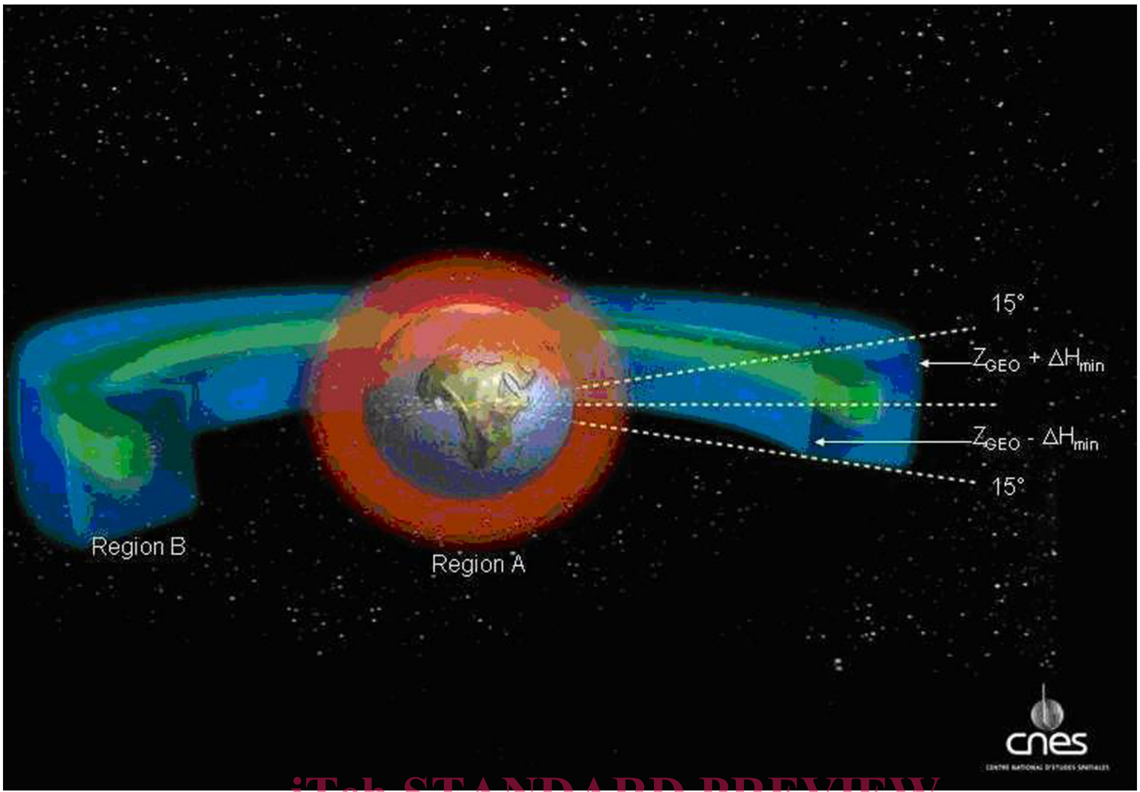

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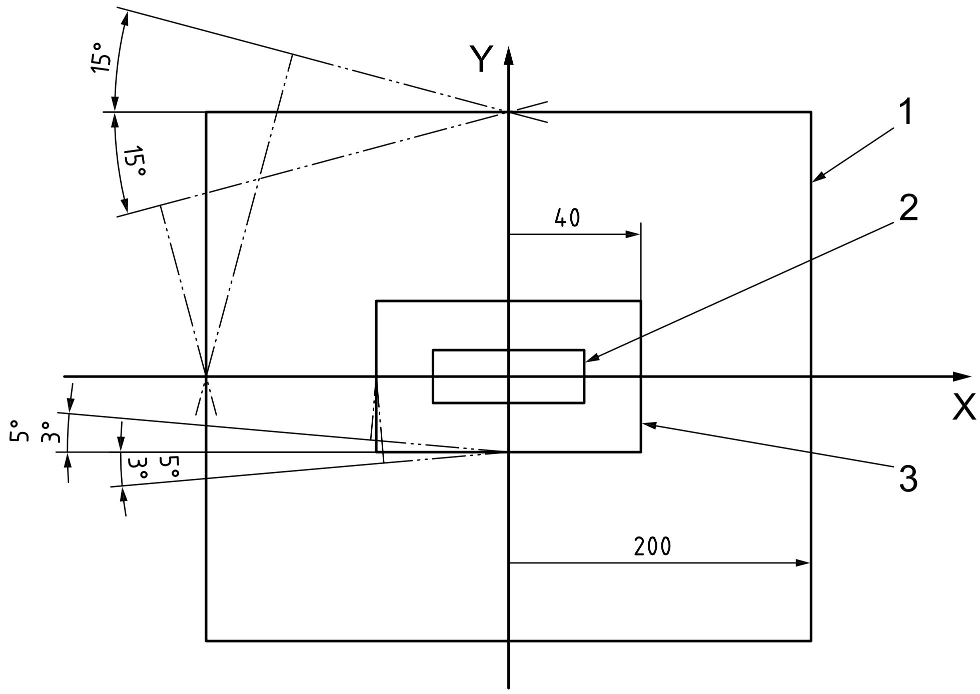
Figure 1 — Three-dimensional view of geosynchronous ring

Dimensions in kilometres



Key

X radial (away from Earth)

Y latitude (north)

1 protected region

2 geostationary control box (±37,5 km × ±37,5 km)

3 geosynchronous control box (±40 km radial; ±3° to ±5° in inclination)

NOTE The dimensions in the figure are not to scale.

Figure 2 — Cross-section of the geosynchronous ring

7 Primary requirements

7.1 Disposal manoeuvre planning

An EOMDP shall be developed, maintained and updated in all phases of mission and spacecraft design and operation. The EOMDP shall be an integral part of the space debris mitigation plan specified by ISO 24113. The EOMDP shall include the following:

a) details of the nominal mission orbit;

b) details of the targeted disposal orbit;

c) estimates of the propellant required for the disposal action;

d) identity of systems and capabilities required for successful completion of the disposal action;

e) criteria that, when met, shall dictate initiation of the disposal action;

f) identities of energy sources required to be depleted before end of life;

g) timeline for initiating and executing the disposal action;

h) timeline for depleting the remaining energy sources;

i) those individuals or entities, or individuals and entities to be notified of the end of mission and disposal and a timeline for notification.

7.2 Probability of successful disposal

In accordance with the requirements of ISO 24113:2010, 6.3.1, a spacecraft shall be designed such that the joint probability of having sufficient energy (propellant) remaining to achieve the final disposal orbit and successfully executing commands to deplete energy sources equals or exceeds 0,9 at the time disposal is executed. ISO 24113:2010, 6.3.1, also requires that the disposal success probability shall be evaluated as conditional probability (weighted on the mission success). Annex E provides a discussion of the conditional probability. Details of the design that provide the basis for the probability estimate shall be included in the EOMDP.

7.3 Criteria for executing disposal action

Specific criteria for initiating the disposal action shall be developed, included in the EOMDP and monitored throughout the mission life.

EXAMPLE Propellant amount remaining; redundancy remaining; status of electrical power; status of systems critical to a successful disposal action; time required to execute disposal action.

Projections of mission life based on these criteria shall be made as a regular part of mission status reviews.

7.4 Contingency planning

Independent of the success or failure of other aspects of a disposal action, a contingency plan shall be developed to deplete all energy sources and secure the vehicle before the final demise of the spacecraft. The objective shall be to ensure that actions necessary to secure the vehicle are taken before end of life. The contingency plan shall include criteria that define when the securing actions are to be taken, the rationale for each criterion, and a schedule for securing actions. The contingency plan shall be included in the EOMDP.

8 Disposal planning requirements

8.1 General

Planning activities for end-of-mission disposal shall start in the mission design phase. Planning for the actual disposal action should begin at least six months before the date of re-orbit manoeuvres. The steps described in 8.2 to 8.8 shall be followed in all mission phases and shall be documented in the EOMDP.

8.2 Estimating propellant reserves

The amount of fuel necessary to perform spacecraft disposal shall be estimated from the design phase, in accordance with the needed accuracy level, and reserved for the disposal phase. The minimum ΔV capability ($3-\sigma$) to reach the targeted disposal orbit shall be determined and specified in the EOMDP. The fuel required to provide this ΔV shall be maintained for end-of-life disposal[3].

3) Further information will be provided in the future International Standard, ISO 23339.

8.3 Computing the initial perigee increase

In accordance with the requirements of ISO 24113:2010, 6.3.2, a spacecraft operating within the GEO protected region must, after completion of its GEO disposal manoeuvres, have an orbital state that satisfies at least one of the two conditions outlined below.

a) The orbit has an initial eccentricity of less than 0,003, and a minimum perigee altitude, ΔH, expressed in kilometres, above the geostationary altitude (35 786 km) calculated according to Equation (1):

$$\Delta H = 235 + (1\,000 \times C_R \times A/m) \tag{1}$$

The minimum value of C_R for computing the initial perigee increase shall be no less than 1,5 (a conservative estimate for C_R, so as to adequately predict the solar radiation pressure effect). Justification shall be provided for using a value less than 1,5. Equation (1) was derived to ensure that the long-term perturbations will not cause the GEO debris to re-enter a protected zone of GEO plus 200 km.

b) The orbit has a perigee altitude sufficiently above the geostationary altitude that the spacecraft will not enter the GEO protected region within 100 years, irrespective of long-term perturbation forces.

8.4 Developing basic manoeuvre requirements for a stable disposal orbit

A stable disposal orbit shall be established by one of the two options described below.

a) Use Equation (1) and the eccentricity constraint to determine initial disposal orbit conditions.

b) Perform long-term (100-year) numerical integrations of the selected disposal orbit. The predicted minimum perigee altitude shall be greater than the 200 km protected region (see 8.5). It is recommended that the optimal eccentricity vector be determined from Tables A.1 to A.3, as a function of the date of orbital insertion and the value of $C_R \times A/m$.

The altitude stability will be improved for either method if the following apply:

— the initial disposal perigee points toward the sun (perigee is sun-pointing);

— the disposal manoeuvres are performed in the most favourable season of the year, such that the same amount of perigee altitude increase will give the largest clearance over 100 years.

NOTE 1 The true optimal direction will differ slightly from the actual sun-pointing direction as a result of lunar perturbations.

NOTE 2 See Annex A for the optimal eccentricity and argument of perigee as a function of time for various values of $C_R \times A/m$. Disposal orbits defined in accordance with Equation (1) are stable if the final eccentricity is less than 0,003. Tables A.1 to A.3 can be used to select the initial guess if option b) is used to determine the initial orbit parameters.

Should the intention be to operate the vehicle after placing it in a disposal orbit, the effects of such operation on the orbit shall be estimated, and this estimate and computations verifying that the operations will not compromise the long-term stability of the orbit (i.e. perigee shall remain above the protected region for 100 years) shall be included in the EOMDP. In all cases, the spacecraft shall be passivated (see 8.7) prior to end of life.

8.5 Developing long-term (100-year) disposal orbit characteristics

Long-term (100-year) orbit histories are needed only when the second option [see 8.4 b)] is chosen to establish a stable disposal orbit. If 8.4 b) is chosen, orbit propagation results developed by a reliable orbit propagator, either semi-analytic or numerical, shall be used to predict histories of perigee heights above GEO for a period of 100 years after initial insertion into the disposal orbit. The orbit propagator shall be of high precision and include as a minimum the perturbing forces of Earth's gravitational harmonics (up to a degree/order of 6 by 6), lunisolar attractions and solar radiation pressure. The precision of long-term propagation of the propagator shall be verified against another well-established orbit propagator. Details on the orbit propagator used, assumptions made and analysis results shall be included in the EOMDP.

8.6 Determining the manoeuvre sequence

The manoeuvre sequence shall be determined that will place the GEO satellite in the required disposal orbit, have the optimal near-sun-pointing perigee and exhaust all the propellant on board. The disposal orbit is obtained after passivation and complete tank depletion, which can have unpredictable effects on orbital parameters and altitude. See Annex B for examples. The initial conditions of the disposal orbit shall be determined using the steps outlined in 8.4 and 8.5.

8.7 Developing a vehicle securing plan

Depletion of propellant creates forces that can affect a vehicle's orbit. The vehicle securing plan shall specify the following:

a) steps to deplete on-board energy sources after the satellite has been placed into the disposal orbit;

b) the effects the depletion action will have on the final orbit of the vehicle (the goal should be either to increase altitude or at least to limit a possible decrease in altitude);

c) criteria for when the plan will be executed; and

d) a schedule to be followed.

8.8 Developing a contingency plan

If a malfunction or other circumstance makes it necessary to proceed to the disposal phase earlier than planned, a contingency plan shall be developed that includes provisions for the following:

a) selecting an alternative orbit that is the least likely to interfere with the protected area (see Annex C): the contingency plan shall include criteria and techniques for selecting this orbit;

b) manoeuvring the satellite to the alternative orbit;

c) securing the satellite after the move; and

d) securing the vehicle if specified criteria are met at any time in the mission.

Annex D provides an example in which the quantity of propellant remaining is uncertain.

Annex A
(informative)

Tabulated values of the optimal eccentricity vector

Tables A.1 to A.3 contain the optimal eccentricity vector [eccentricity and argument of perigee plus RAAN (or longitude of periapsis)] as a function of time and a function of $(C_R \times A/m)$, expressed in square metres per kilogram, that will yield the highest perigee over the next 100 years. The optimal values were calculated in a brute-force fashion using increments of $2{,}3 \times 10^{-5}$ in eccentricity and 5° in longitude of periapsis. The benefit gained from using the optimal vector over the sun-pointing strategy varied from 0 km to 20 km (the average was approximately 9 km). However, if the sun-pointing strategy is chosen for the disposed vehicle, then the longitude of periapsis should be set equal to the value of the solar longitude (depicted as L_S in Tables A.1 to A.3) with an eccentricity equal to $0{,}01 \times C_R \times A/m$. These charts can be interpolated to find the optimal vector for any particular satellite at a given time. However, the following should be noted when using these tables.

The initial conditions used to generate the data assumed a constant semi-major axis of 300 km above GEO (i.e. a constant ΔV was used in the disposal), mean anomaly of 180° (i.e. the last burn occurs at apogee, raising the perigee so that the eccentricity is equal to the tabulated value), an inclination of 7,74° (maximum at end of life if inclination drift is allowed) and an epoch of 0:00 Universal Time on the first day of each month. Additional analysis has shown that the optimal vector depends little upon these elements (the minimum perigee altitude may vary by approximately 2 km for each component), but if a high level of accuracy is required for a given disposal, the interpolated values found from the tables should be used as an initial guess so as to find the optimum for a particular disposal situation. The exception is the RAAN: in the search process, the initial RAAN was set to 62,3° and the argument of perigee was changed in 5° increments until the optimal value was found. Different RAANs were then checked and it was found that the relevant angular parameter was the argument of perigee plus RAAN; if this value is held constant, then the results will again be consistent with 1 km to 2 km, irrespective of the particular RAAN.

In addition, care should be taken if interpolating the values. In searching for optimal values in the angular argument, it was found that, at times, there was not one pure maximum, but multiple local maximums. As either the time or $C_R \times A/m$ advanced, the true maximum jumped from one peak to another. For example, consider the 2008-05-01 disposal. A $C_R \times A/m$ of 0,015 m²/kg has an optimal eccentricity of 0,000 04 and a longitude of periapsis and 262,3°, whereas the $C_R \times A/m$ of 0,03 m²/kg has optimal values of 0,000 09 and 37,3°. Linearly interpolating would imply optimal values of 0,000 057 and 307,3° for a $C_R \times A/m$ of 0,02 m²/kg. Instead, the $C_R \times A/m = 0{,}02$ optimal value was actually 0,000 015 and 47,3°. In this case, the optimal point switched from one maximum to another, and therefore the intermediate maximum would actually be close to one point or the other.

Therefore, when confronted with angular changes greater than 90°, it is recommended that interpolation not be performed. Instead, the closer value should be used either directly or as a starting point for a more refined search.

A few final comments on the general behaviour of the system are warranted. When the A/m was small ($C_R \times A/m < 0{,}01$), the optimal angle was pointed at the lunar apogee; when the A/m was large ($C_R \times A/m > 0{,}03$), the solar radiation pressure force became dominant and the optimal angle was directed toward the sun.