ECCENTRICITY MANAGEMENT FOR GEOSTATIONARY SATELLITES DURING END OF LIFE OPERATIONS

N. Delong [1] , C. Frémeaux [2]

Centre National d'Etudes Spatiales, Toulouse, France
[1] Nicolas.Delong@cnes.fr
[2] Claire.fremeaux@cnes.fr,

ABSTRACT

Inter-Agency Space Debris Coordination Committee (IADC) recommends mitigation measures such as preventing on-orbit break-ups, removing spacecrafts and orbital stages at the end of their mission operations from the useful densely populated orbit regions and limiting the objects released during normal operations. Among them, at the end of their life, geostationary satellites must:

- be removed from geostationary region.
- Be passivated, in order to reduce explosion risk: the pressure must be lowered as much as possible. This operation has an impact on the orbit.

The aim of this paper is to present some strategy to assume this protection with the lowest propellant consumption.

1- ECCENTRICITY AND PERIGEE ALTITUDE

As an introduction to this paper, let us remind that eccentricity, semi-major axis and altitude of perigee over the geostationary orbit are linked by the following relation :

$$Z_p = a\,(1 - e\,) - a_{geo} \qquad (1)$$

Where e is the eccentricity, a is the semi-major axis and a_{geo} the geostationary orbit radius, equal to 42165 km. So, for a given semi-major axis, the greater the eccentricity gets, the lower the altitude of perigee is.

For a geostationary orbit, since the inclination and the eccentricity are very low, the right ascension of the ascending node Ω, and the perigee argument ω are undetermined. The eccentricity and the position of perigee are therefore represented by a vector $\vec{e}$, whose norm is equal to eccentricity and which is directed towards the perigee of orbit :

$$\vec{e}\left|\begin{array}{l} ex = e\cos(\Omega + w) \\ ey = e\sin(\Omega + w) \end{array}\right. \qquad (2)$$

The eccentricity vector evolves naturally under the effects of perturbations. This evolution can be divided into two parts :

- The effect of the solar radiation pressure makes the eccentricity vector turn with a one year period. Fig.1 shows in dotted line this annual effect on $\vec{e}$. The radius *Re* of the circle is equal to $1.15\ 10^{-2}.Cr.\ A/M$, where A is the aspect area toward the Sun, M is the satellite mass and Cr is the solar radiation pressure coefficient. Cr A/M is expressed in m^2/kg.

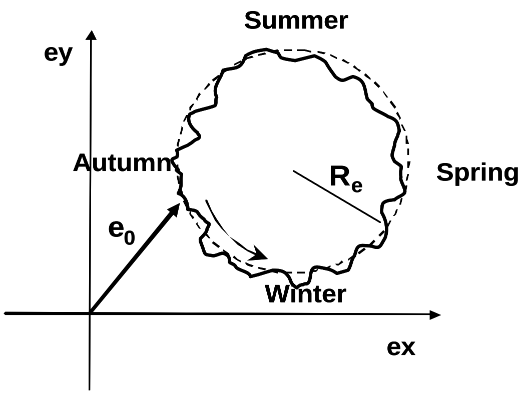

Figure 1 : eccentricity evolution due to global effects

- The combined gravitational perturbations from Earth, Sun and Moon has a lower effect on $\vec{e}$, and induce small periodic variations around previous circle.

The global effect over one year period is represented on figure 1.

We can see that the value of eccentricity, the norm of the vector $\vec{e}$, can change a lot within one year. The consequence is a natural variation of the perigee altitude over the year, independently from any semi major axis variation

2- ECCENTRICITY MANAGEMENT DURING REORBITATION

2.1- IADC recommendations applied to geostationary satellites

Concerning geostationary satellites, the region of space to be protected is defined by the following :

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- an altitude upper than geostationary altitude minus 200 km and lower than geostationary altitude plus 200 km
- a latitude between -15 degrees and +15 degrees.

This region includes the operational region as well as the region used for satellite relocations.

To avoid collision risk, IADC recommends to remove satellites above this region at the end of their mission, far enough to prevent it from re-entering in this area with orbital perturbations. The minimal increase in perigee altitude is :

$$\Delta H = 235 \text{ km} + 1000.Cr.A/M \qquad (1)$$

This minimal altitude takes into account 35 km margin for the combined gravitational perturbations from Earth, Sun and Moon, while the term "1000.Cr.A/M" represents the maximum extent of solar radiation pressure effect on orbit eccentricity and thus on altitude of perigee.

2.2- Lower cost re-orbitation with IADC recommendations

Any satellite control centre tries to minimize the propellant consumption for manoeuvres, during the mission phase and also during the reorbitation phase. Indeed, the lowest the consumption of re-orbitation is, the higher the satellite can be re-orbited, or, from another point of view, the longer its lifetime is, the quantity of propellant reserved for reorbitation being smaller.
With respect to the IADC recommendation expressed by the formula 1, the lowest cost re-orbitation will consist in increasing the altitude of perigee to the ΔH given value, but also the apogee to the same value, which means a null eccentricity to be targeted at the end of the re-orbitation. The circular orbit is in fact the lowest-cost orbit to target with respect to the IADC recommendations since the altitude is constant and equal to the minimum value. Applied to a Eurostar 2000 satellite for example, with a Cr A/M equal to about 0.05 m^2/kg, the minimal altitude increase is 285 km, which means total cost of 10.4 m/s velocity increase.
The question is now : how will the satellite altitude change during the months following reorbitation ?

2.3- The natural evolution of orbit eccentricity after re-orbitation

As explain before, the evolution of the eccentricity will have an annual periodicity. The evolution can be represented on figure 2.

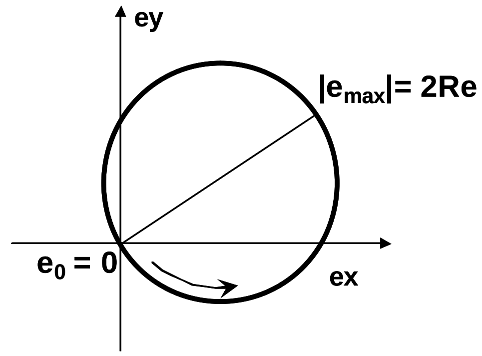

Fig. 2 : natural evolution from a null eccentricity

So, six months after the re-orbitation, the eccentricity will reach a maximum value of *2Re*, and therefore the perigee altitude of the orbit will reach a minimum value *Zp (6months) = a(1- e_{max})-Re*.
With the previous numerical value, the perigee altitude 6 months after the re-orbitation will have decreased down to 235 km, as shown on fig. 3.

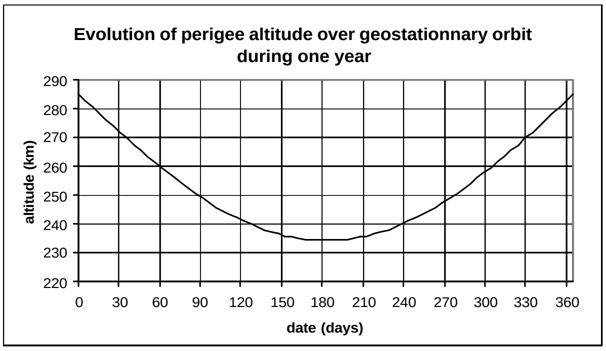

Fig.3 : Evolution of perigee altitude during one year

So, we can see that the IADC recommendation on the perigee altitude as been performed successfully : The satellite will remain 200 km above the geostationary orbit, if we consider that the satellite altitude can decrease of 35 km under the action of combined gravitational perturbations from Earth, Sun and Moon.

Then, the IADC formula gives a sufficient condition to respect this altitude constraint.
But in fact, this condition is not a necessary one : we can imagine indeed that if we could keep the evolution of perigee altitude constant all over the year, equal to 235 km (including the 35 km margin), the geostationary region would be also protected, but with a lower cost in terms of propellant.
This solution is given hereafter.

2.4- Geostationary region protection with the lowest cost of propellant

The solution is to re-orbit the satellite while targeting its natural eccentricity, which means a perigee toward the Sun and an eccentricity equal to Re. The natural evolution will follow a circle centred on 0, then the eccentricity will remain constant, and so will the altitude of perigee. This can be summarized with the two figures 4 and 5 :

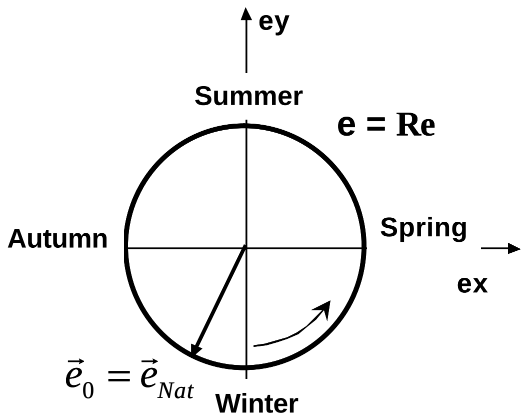

Figure 4 : natural evolution from natural eccentricity

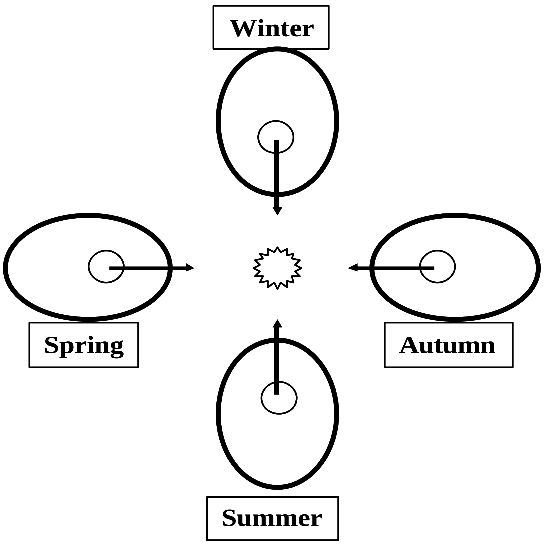

Figure 5: orbit evolution from natural eccentricity

This solution ensures that the influence of the solar pressure radiation will have no more effect on the altitude of perigee. To protect the geostationary region, taking into account the combined gravitational effect of Earth, Sun and Moon, the increase of perigee altitude have to be at least of Zp = 235 km.

The semi-major axis to target is then given by formula 3

$$a = \frac{Zp + a_{geo}}{1 - R_e} \qquad (3)$$

Applied to the previous satellite, the cost to reach this value is about 9.4 m/s of velocity increase. So for the same effect with regard to the protection of the geostationary region, the economy is about 1 m/s velocity increase. This can represent several months of lifetime economy if the inclination of the satellite at the end of its mission is not controlled.

2.5- Evolution of aspect area toward the Sun

An other parameter which is relevant is the aspect area towards the Sun. Since the satellite attitude is no more controlled after its extinction, the solar arrays are no more Sun pointed, but their direction can be considered as hazardous and variable under the effects of attitude perturbations. The aspect area towards the Sun is then far lower than the on station one, and so is the natural eccentricity Re.

2.6- Real case application

CNES has recently performed recently the reorbitation of a geostationary satellite, following the strategy described here above an, targeting the natural eccentricity with a sun pointed perigee.

Fig. 6 and 7 show the natural evolution of eccentricity and perigee altitude with regard to the semi-major axis during one year after the réorbitation :
The thick and continuous line (curve n°2) shows the real evolution, restored from the NORAD measurements (curve n°3). The altitude varies from minus 20 km up to minus 10 km under the semi-major axis. The restituated aspect area towards the Sun is roughly 40% lower than the on station value. The curve n°1 shows the simulated evolution with this on station value, with a perigee altitude varying down to minus 30 km under the semi-major axis.
The last curve (n°4) represents the simulated evolution as if a null eccentricity would have been targeted during the réorbitation. The lowest perigee value in this case is about 50 km under the semi-major axis.

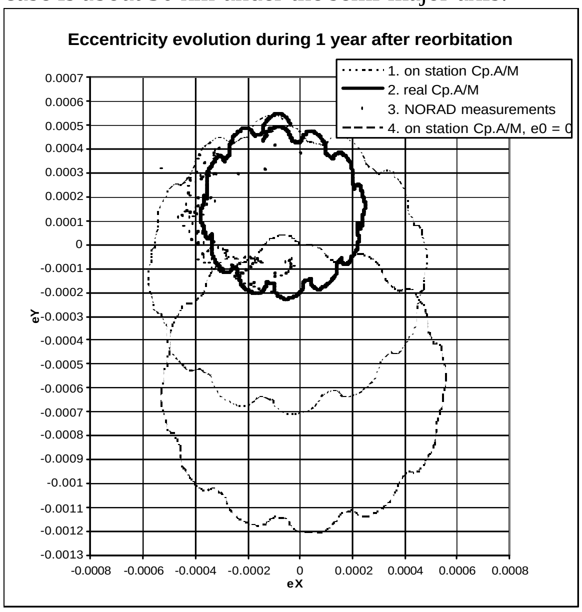

Figure 6 : eccentricity evolution during 1 year after re-orbitation

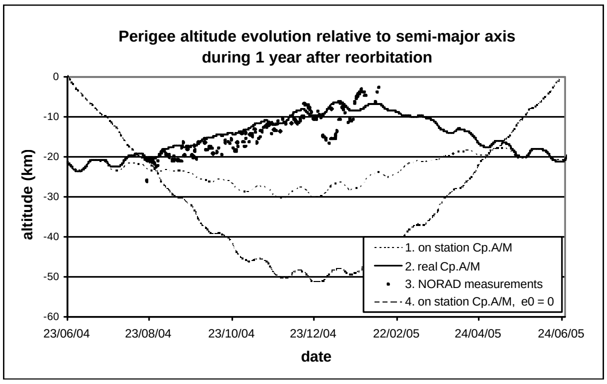

Fig. 7: Perigee altitude with regard to semi-major axis during 1 year after re-orbitation

In conclusion, the eccentricity is a key orbital element to manage during reorbitation, even though the semi major axis remains the main one. But the end of re-orbitation does not mean the end of satellite life. The aim of the following part is to show how fluidic passivation can also slightly impact the altitude of perigee, and to present how it is possible to influence these impacts in order to keep the benefits of reorbitation.

3- ECCENTRICITY MANAGEMENT DURING FLUIDIC PASSIVATION

3.1- An experience of passivation in CNES

In 1999, CNES has successfully achieved complete passivation for the GEO satellite TDF2, after its re-orbitation, but a perigee decrease occurred, which proved to be caused by the very low forces involved during passivation process. TDF2 was a telecommunication satellite with ASP Spacebus platform. The propulsion subsystem was based on helium pressurised bipropellant system
For the fluidic passivation, the spacecraft was first put to Sun Acquisition Mode (in order to improve stability) using the tanks with some remaining propellants.
Then the emptying operations could began, using 2 thrusters (5A and 5B) simultaneously which were chosen because they were on opposite walls, in the roll axis direction, so they should create no torque and opposite thrusts. The result should be a minimum actuation of other thrusters for attitude control, and no modification of spacecraft orbit.
For this reasons, the TDF2 passivation was performed without other preliminary studies. It consisted in opening all the tanks and performing 14 firing phases for a total open duration of about 11 hours. At the end of the passivation, residual tanks pressure was then lower than two bars and satellite transmitter was finally turned off.
The last orbit restitution showed that the fluidic passivation had no impact on North-South parameters, no impact on semi-major axis, but a slightly impact on eccentricity, which increased from 4.10^{-3} to $4.7\ 10^{-3}$, and that caused a 27 km perigee decrease.

Several possible causes were studied :

- Other thrusters activity for attitude transition to Sun Acquisition Mode and attitude control during firing
- Thrust dissymmetry between thrusters 5A and 5B, during propellant run-out, during depressurization, and also due to partial freezing of the thruster on the shadowed wall of the satellite.

Among these potential causes, going to Sun Acquisition Mode and attitude control during firing phases were proved to cause little perturbation, while thrust dissymetries due to different thrusters characteristics and possible partial freezing could have significant impact and explain the perigee decrease.

As a conclusion of this passivation experience, it was important for the future satellite passivation to try to understand and to minimize the perigee decrease, and even to have an increase of it if possible. These studies were then performed for a Eurostar 2000 platform, in anticipation of future end of life operations of CNES geostationary controlled satellites.

3.2- Eurostar 2000 platform case

The Astrium Eurostar 2000 platform propulsion subsystem is also composed of 4 propellant tanks, with the same oxidiser and fuel as TDF2, but it has only 12 thrusters located on 6 different points on the satellite : thrusters i A and i B are located one beside the other. Pairs of thrusters 1, 2 and 3 are located on North satellite wall (-Y) and are used in station-keeping for South manoeuvres, thrusters 4 are one West wall (-X), used for East manoeuvres, and thrusters 5 and 6 on East wall (+X), used for West manoeuvres.

The idea of getting a robust attitude in case of gas expelling causing attitude disturbance seems a good one. Thus the end-of-life global sequence could be as follows :
First a set of important East manoeuvres in station-keeping mode, in order to reach the IADC recommended altitude above geostationary orbit. The remaining propellant estimation (including uncertainty) should be sufficient to allow these first manoeuvres to be performed with no particular care for bubbles,
then additional manoeuvres could be carefully made, still in station-keeping mode, maybe with particular conditions (special pulsed mode …) allowing to monitor efficiently each manoeuvre (attitude, thrusters temperature, tank pressure …) in order to stop the manoeuvre as soon as gas arrival is detected
Last, a Sun pointed attitude could be acquired and firing sequences could be performed with chosen thrusters and

chosen duty-cycles (cold thrusts while liquid propellant still present) until the propellant tanks pressure gets low enough (TDF2 case) or the attitude is lost (in the worst case).
Because Eurostar 2000 platforms do not have symmetrical thrusters, it is not possible to try to get a null resultant force during the sun-pointed passivation phase. Thrusters 1, 2 and 3 are perpendicular to the spin axis, which is not in favour of stability. It was chosen for the simulations that thruster 4 should be activated : its direction is aligned with the spacecraft mass centre, and residual torques could be compensated with thrusters 5, 1 and 2.

3.3- Basic simulations

These first simulations were made in order to get information about the expected orbit disturbance in some simple cases :

- typical end-of-life state parameters for a Eurostar 2000 platform
- several orbit shapes : circular or elliptic with 4 different perigee orientations : towards sun, 90°, 180° and 270° from sun direction.
- one single combustion phase performed at different hours : 0:00, 6:00, 12:00 or 18:00 GMT

Thruster 4 being one the Sun facing wall (in order to avoid freezing), the main thrust component is in the opposite direction. So the simulated single thrusts are equivalent to radial manoeuvres at 0:00 (local satellite time) and 12:00, East ones at 18:00 and West ones at 6:00 (if such manoeuvres were performed in nominal station-keeping earth-pointed attitude).
The simulations give interesting results :

- a 6:00 thrust induces a semi-major axis decrease, which also means a perigee altitude decrease except when perigee is at 6:00 (perigee thrust)
- symmetrically, a 18:00 thrust induces a semi-major axis increase, and consequently a perigee altitude increase, except when perigee is at 18:00 (perigee thrust)
- a 0:00 or 12:00 thrust has no significant effect on perigee altitude when apogee is at 0:00 or 12:00

The Tab. 1 summarizes the positive / negative effect on perigee altitude after each kind of thrust : minus signs point out unfavourable situations that should be avoided. Plus signs show favourable cases in which a re-orbitation can be improved if necessary. Equal signs correspond to cases with negligible impact on perigee altitude.

Apogee time / Orbit shape / Thrust time	(orbit = circle)	0 h	6 h	12 h	18h
0 h	=	=	-	=	+
6 h	-	-	-	-	=
12 h	=	=	-	=	+
18 h	+	+	=	+	+

Table 1 : effect of thrust on perigee altitude

With respect to perigee altitude, this table clearly shows that :

- whatever the initial orbit shape is, 6 h thrusts should be avoided and 18h thrusts are recommended.
- for any thrust time, if the initial orbit has its apogee at 6h the perigee altitude can only decrease, if the apogee is at 18h it can only increase.
- 0h and 12h thrust times, or apogee times, are medium cases : the impact can be good or bad depending on the other parameter

Plus and minus signs show cases where the impact on perigee altitude is significant : between 15 km and 35 km. For the two extremes situations : (6h apogee+ 6h thrust) and (18h apogee + 18h thrust) the impact is higher than 60 km.
As a conclusion, these basic simulations have proved that thrust hours as well as initial orbit shape can have a strong influence on the perigee evolution during passivation operations.
Deciding about thrust hours is a matter that can be managed during passivation stage, but the initial orbit shape cannot : it must be taken into account in advance, while computing and performing re-orbitation manoeuvres.

3.4- Complete scenarii simulations

Another set of simulations has been made with a more complete and realistic scenario.
Satellite re-orbitation has been made as follows: the target orbit was circular, 300 km higher than geostationary orbit. Several manoeuvres were performed but gas bubbles appeared during the last one and manoeuvre was stopped before its nominal end. Thus the orbit apogee altitude is geo + 300 km but perigee altitude is only geo + 250 km at the end of re-orbitation.
Spacecraft tanks are then swapped, and spinned –X sun pointed attitude can be acquired.
In order to take advantage from passivation as far as possible, passivation thrusts are scheduled around 18h (local satellite time) which is the most favourable time.

Final combustion (with a 10% pulsed firing mode) lasts half an hour centred around 18h the first day, then four hour depressurisation phases take place in steady state mode with very low (and decreasing) forces, starting the second day at 16h, 18h and 20h, and the third day at 16h, until tank pressure gets very low.
Following charts show the spacecraft altitude evolution during passivation in several cases :

- if initial apogee was at 18h, passivation thrust can raise perigee up to initial targeted value of geo + 300 km and thus achieve complete re-orbitation with respect to IADC recommendations, as shown on Fig. 8.

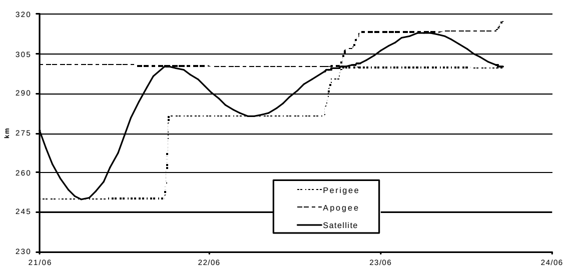

Figure 8 : initial apogee at 18h
X=km 230-320 / Y=21/06, 22/06,23/06/24/06

- if initial apogee was at 0h, the orbit is slightly raised and final perigee = geo + 265 km : Fig.9

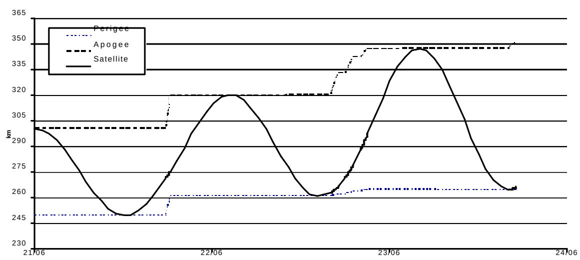

Figure 9 : initial apogee at 0h
X=km 230-365 / Y=21/06, 22/06,23/06/24/06

- if initial apogee was at 6h, the apogee altitude increases, but perigee remains unchanged :

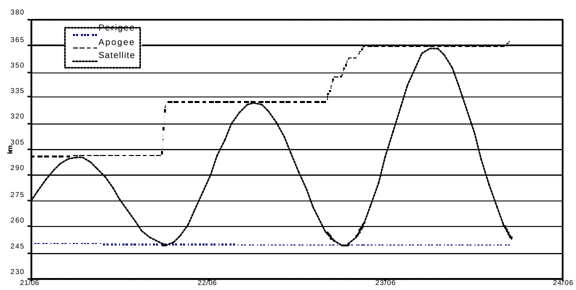

Figure 10 : initial apogee at 6h
X=km 230-380 / Y=21/06, 22/06,23/06/24/06

- with the same initial apogee time (6h), if the whole passivation thrust were made around 6h instead of 18h, the perigee altitude would decrease down to 185 km; a significant part of re-orbitation effort would thus be lost!

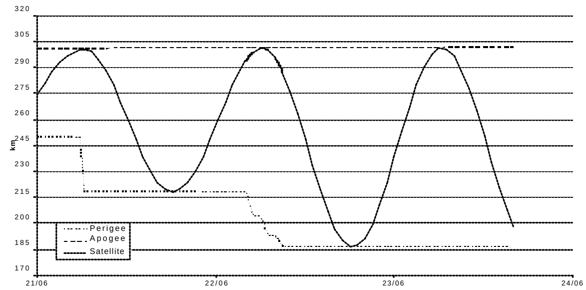

Figure 11 : initial apogee at 6h and passivation at 6h
X=km 170-320 / Y=21/06, 22/06,23/06/24/06

It can be noticed that all previous results are obtained with exactly the same propellant mass !
A 18h apogee is the best initial situation. Re-orbitation manoeuvres should be scheduled in order to be in this situation whenever manoeuvres have to be stopped. For example, first East manoeuvre could be made at apogee time (0h LST, most of the time), with an amplitude that makes the orbit circular. Second manoeuvre could be made at 6h, giving an elliptic orbit with 6h perigee and 18h apogee. Third one at 18h with same amplitude as previous one would get back to circular orbit, and so on.

4- CONCLUSION

The aim of this paper was to show the importance of eccentricity management during end of life of geostationary satellites. For the re-orbitation, targeting the natural eccentricity allows to ensure that the perigee altitude reached at the end of the operations will remain stable on long-term, and is not an instantaneous value. This strategy ensures the protection of the geostationary region with the lowest propellant consumption.
Passivation operations that follow the re-orbitation can have important impacts on the eccentricity, and then on the perigee altitude, even if very low forces are involved. In the case of a Sun pointed attitude, there are some favourable and unfavourable times to perform the passivation thrusts.

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