LONG TERM EVOLUTION OF OBJECTS IN GSO-DISPOSAL ORBIT

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ABSTRACT

The paper presents the results of the long-term evolution of the objects in the graveyard orbit. A long-term numerical propagator has been exercised which includes the perturbation due to SRP, third body effects due luni-solar forces and Earths gravitation potential. The paper is divided into two parts; the first part of the paper examines the perigee height variation over one hundred years, for INSAT class of satellites. The initial perigee of the disposal orbit is taken to be as per the recommendations of the IADC. The results of the study have been verified with other long-term propagator and are in conformity, in general, with the proposed change suggested in the IADC guidelines related to the specification of the eccentricity of the disposal orbit to be below 0.005. For eccentricity above this, the objects pass through the protected GSO region.

In the second part of the paper, long-term growth of the perigee for objects placed with their initial eccentricity vector appropriately with respect to Sun was studied. Case studies were carried out for different initial epoch to account for the effects of lunar node on long-term propagation. Based on the study it is evident that, for a few cases, the Sun pointed eccentricity vector is advantageous and the specification for eccentricity for the disposal orbit may not be required if placed in this orientation. The perigee does not fall below the 200 km above the GEO for the initial Sun pointed eccentricity vector. As the area-to mass increases, the peak-to-peak variation of the perigee altitude is found to be high. For few of the cases, the perigee altitude of 200 km above the GEO is protected if the initial eccentricity vector is towards the Sun. The study indicates that for an epoch, there is an appropriate phasing of the eccentricity vector with respect to the Sun, which will ensure that the protected regions are not penetrated even for initial disposal orbit eccentricity more than 0.005. In view of this, the IADC guideline or the support document may indicate this strategy, which will require no strict stipulation on eccentricity for the disposal orbit of GSO objects. The sensitivity studies carried show that higher the initial eccentricity of the disposal orbit more stringent will be the requirement of achieving the appropriate phasing of the eccentricity vector with respect to sun. Details of implication of the implementation of this scheme, however, have to be addressed during the terminal operations.

1.0 **INTRODUCTION**

Most operational satellites are maintained in the "geo-stationary ring" along with other abandoned objects. The abandoned objects move freely in a well predictable manner following the laws of gravitation and less predictably, under the solar radiation pressure and third body perturbation causing probability of collision with active spacecraft. In order to reduce the occurrence of probable collision, IADC has brought out the guidelines [1] for the preservation of the GEO orbit. According to the guidelines, the GEO spacecraft that have terminated their mission in GEO should be maneuvered far away from GEO enough not to cause interference with objects in GEO. In addition, eccentricity should be controlled as small as possible in re-orbit maneuver. As per the recommendation, the minimum increase in perigee altitude taking into account all orbital perturbation is

$$235 \text{ km} + 1000 * Cr * (A/m) \qquad (1)$$

where
Cr : Solar Radiation coefficient
A/m : the area to mass ratio [sq.m /Kg]
235 km : sum of protected region for GEO (200 km) and maximum descent of re-orbited spacecraft due to luni-solar and geo-potential perturbations (35 km)
The guideline as specified above also includes effect of perturbations due to SRP.

Several studies have been carried out in the past regarding the stability of disposal orbits. Martin et al [2] found that the orbits after re-orbiting should be kept as near circular as possible. Their studies indicate violation of minimum perigee if the eccentricity of disposal orbit is increased. There is a general conclusion that the IADC

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suggested guideline of at least 235 km for the perigee of the GSO disposal is appropriate provided the initial eccentricity of the disposal orbit is not more than 0.005.

An attempt was made to assess the effect of initial eccentricity more than 0.005 on the long-term perturbation due to all the perturbative forces, particularly to understand the eccentricity vector evolution, which primarily determines the perigee radius behavior. The studies carried out, indicate that the space debris does not penetrate the protected regions if the initial eccentricity vector is pointed appropriately towards the Sun for various A/m of the abandoned spacecraft. This paper presents various aspect related to long term perigee growth for appropriately placed eccentricity vector at the end of life disposal orbit and presents the results of the studies carried out.

2.0 LONG TERM ORBIT PROPAGATION

The major perturbative forces that play a dominant role on long-term propagation are due to Geopotential, differential gravitational attraction due to third body and solar radiation pressure on.

This section describes the numerical analysis of the long-term eccentricity growth. The orbit propagation is based on special perturbations method which is also frequently referred to in literature, as Cowell's formulation / numerical method. Major advantage of the method is that the solution obtained contains all secular and periodic (short, long) variations caused by the perturbing forces. The solution is obtained by numerical integration of the perturbed equation of motion, which considers all the major perturbing forces.

$$\ddot{\mathbf{r}} = -(\mu / r^3)\,\mathbf{r} + \mathbf{a}_{pert} \qquad (2)$$

where

$$\mathbf{a}_{pert} = \mathbf{a}_{nonsph} + \mathbf{a}_{drag} + \mathbf{a}_{3body} + \mathbf{a}_{srp} + \mathbf{a}_{tides} \qquad (3)$$

The major perturbing forces considered are due to non-spherical mass distribution of Earth, atmospheric drag, differential gravitational attraction of the third body viz. Sun, moon, solar radiation pressure and ocean, solid Earth tides. The numerical integration method used is 8^{th}-order Adams-Bashforth integrator. Force models used are the 36×36 EGM-96 Earth gravity model, Jacchia-71 atmosphere, JPL DE-405 planetary ephemeredes, Earth albedo and solid Earth tides.

2.1 Gravitational Potential

The gravitational field is modeled using the standard spherical harmonic representation,
where

$\mu \equiv$ Earth's gravitational constant

$a_e \equiv$ Semimajor axis of the Earth's reference ellipsoid

$$U = \left(\frac{m}{r}\right) \sum \left(\frac{a_e}{r}\right)^n \sum P_{n,m}(\sin f) \times \left[C_{n,m} \cos m l + S_{m,n} \sin m l\right] \qquad (4)$$

r, λ, $\phi \equiv$ Satellite distance, longitude and latitude in Earth fixed system

$C_{n,m}$, $S_{n,m} \equiv$ Spherical harmonic coefficients of degree n and order m

$P_{n,m} \equiv$ Associated Legendre Functions of degree n and order m.

Acceleration due to the Earth's gravity field is obtained by taking the gradient of the potential function.

2.2 Third Body Effects

Differential gravitational attraction due to sun and moon is modelled as

$$\mathbf{f}_{3B} = \Sigma\ \mu_{3b} \left[(D_{3b} / \Delta_{3b}^3) - (\mathbf{r}_{3b} / r_{3b}^3) \right] \qquad (5)$$

where

μ_{3b} = Gravitational constant of the third body

Δ_{3b} = Position vector of the third body with respect to the satellite

r_{3b} = Position vector of the third body with respect to the Earth

Third body being Sun and Moon.

Stability of the disposal orbit was investigated by carrying out orbit prediction over 100 Years, using the in-house developed orbit prediction software which is in operational use for all the Indian satellite missions and has also been used to carry out orbit analysis like orbit resonance Gopinath et al [3].

3.0 LONG TERM EVOLUTION OF GSO SATELLITES

Objective of the study was to investigate feasibility of positioning the initial eccentricity vector in such a way that minimum perigee condition is not violated over 100

Years, even when the initial eccentricity magnitude is more than 0.005. The generalised model is capable of simulating the Earths gravitational potential, luni-solar perturbation, solar radiation pressure, etc, described earlier. The perturbation sources taken into account for this study are Earth gravity model up to order, degree 12, Gravitational attraction due to Sun & Moon and SRP. Using this long-term propagator the studies were initially carried for ISRO satellites whose area/mass ratios of ISRO satellites are depicted below:

Table 1. Area to Mass ratio for ISRO GEO spacecraft

Space craft Name	Area/Mass m^2/ Kg
GSAT-2	0.02
GSAT-3	0.021
INSAT-2E	0.027
INSAT-3B	0.021
INSAT-3A	0.025
INSAT-3C	0.021
INSAT-3E	0.025
MESTSAT	0.011

3.1 Orbit Stability Studies For E £ 0.005

Large number of case studies for satellites having A/m ranging from 0.01 sq.m /kg to 0.03 sq.m/kg with circular orbit at 42400 km (semi-major axis) but different combinations of other parameters like epoch, node etc., were carried out to study the orbit stability over 100 Years. The results obtained show eccentricity change is generally sinusoidal with approximately an 11-year period and also a 1-year period, which is due to SRP. The magnitude of perturbation results in a drop of about 35 km in perigee radius that confirms the basis used in the recommended IADC guideline of minimum perigee radius of at least 35 km above the protected region for disposal orbits.

Similar studies with eccentricity equal to 0.005 indicate that entry into protected region may occur in some cases. However, for e< 0.005, the space objects do not enter the protected region. Based on the study, it was evident that for the satellites with their initial perigee height in accordance with the IADC guideline and for e< 0.005, the satellites do not enter the protected region for any initial epoch placement of the satellite in the disposal orbit. Subsequently studies were carried out with larger initial eccentricity using other strategies, which are presented, in the following sections.

4.0 THE SUN POINTED PERIGEE STRATEGY:

It is well known phenomena that the eccentricity vector traces a near circle over a year due to the effect of SRP. The radius of the circle is directly proportional to the projected area and inversely proportional to the mass of the satellite. The center of this circle depends on the initial eccentricity vector and the direction of Sun, at epoch. In Sun-pointed perigee, the strategy is to perform re-orbiting maneuvers such that the initial orbit achieved will have the eccentricity vector point along sun. Once this is achieved the solar radiation pressure perturbation will not contribute to the growth of eccentricity magnitude, which in turn means SRP will not contribute to perigee radius lowering.

Considering third-body (luni-solar) gravitational effects and influence of solar radiation pressure, IADC has specified a guideline [Ref. 1] for the minimum perigee of geosynchronous disposal orbits for GEO satellites at end-of-life as

$$\Delta H = 235 + 1000\, C_r\ (A/m) \tag{6}$$

where ΔH is the perigee altitude of the disposal orbit above GEO.

Based on the results discussed in section 3.1, the above stated IADC guideline should be modified to include a statement that the magnitude of eccentricity shall be less than 0.005. Further, it may also be noted that, ΔH can be chosen as 235 km and magnitude of the eccentricity can be chosen as

$$e = 0.0112 \ast C_r\ (A/m) \tag{7}$$

along with sun pointed eccentricity vector, in order to minimise the delta velocity required for achieving the disposal orbit at end of life. Such a strategy would result in a saving of delta velocity, for example, about 2(m/s) for a satellite with (A/m) of 0.1 (m^2/kg).

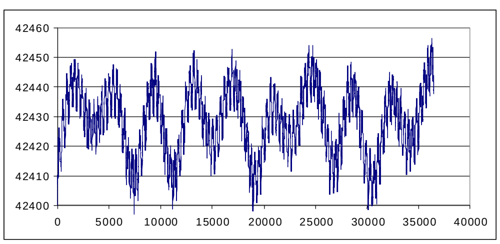
4.1 Stability Studies Of Disposal Orbits For E ³ 0.005

Earlier studies regarding the stability of disposal orbits report violation of minimum perigee if the eccentricity of diposal orbit is increased. There is a general conclusion that the IADC suggested guideline of at least 235 km for the perigee is appropriate provided the initial eccentricity of the disposal orbit is not more than 0.005.

Hence in this study, an attempt was made to assess the effect of initial eccentricity more than 0.005 on the long-term perturbation due to all the perturbative forces, particularly to understand the eccentricity vector evolution, which primarily determines the perigee radius behavior. Objective of the study was to investigate feasibility of positioning the initial eccentricity vector in such a way that minimum perigee condition is not violated over 100 Years, even when the initial eccentricity magnitude is more than 0.005. Long-term propagation of the orbit for 100 Years was done and the minimum perigee radius variation over 100 Years was obtained with the following initial conditions, for different values of initial eccentricity (and corresponding semi-major axis) for the GEO disposal cases.

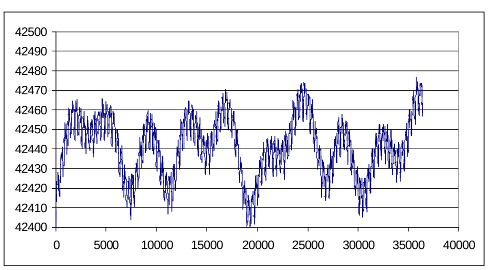
Epoch	: 2010-1-1
Inclination	: 0.04 deg
R.A of ascending node (RAAN)	: 0.0 deg
Argument of perigee (AOP)	: 281.4 deg
Area/mass ratio for SRP	: 0.01 m^2/kg

Above values for RAAN and AOP are selected to keep the eccentricity vector pointed towards sun at epoch. Figures presented below, show variation of perigee radius over one hundred years. Perigee radius in km and Time past epoch in days are shown on the Y-axis and X-axis respectively.



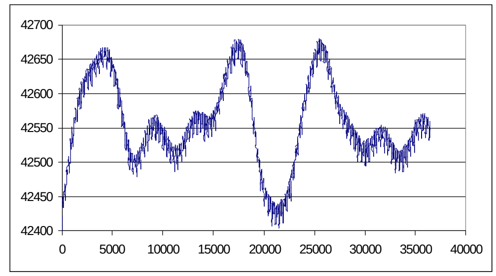
a=42613 km, e= 0.005, sw= 281.4deg. epoch: 2010-01-01

Figure 1. Perigee variation for A/m = 0.01sqm/kg



a=42828.3 km, e= 0.01, sw= 281.4deg. epoch: 2010-01-01

Figure 2. Perigee variation for A/m = 0.01 sqm/kg



a=44631.6km, e= 0.05, sw= 281.4deg. epoch: 2010-01-01

Figure 3. Perigee variation for A/m = 0.01 sqm/kg

The results obtained are summarised in the following Tab 2. As seen in the Tab 2, the choice of eccentricity vector does indeed ensure that the satellite does not enter the " Protected region", since the perigee radius remains above 42364 km, throughout 100 years.

Table 2. Effect of higher "e"

Epoch elements			Minimum perigee radius over 100 Years (km)
Semimajor axis (km)	Eccentricity	Perigee radius (km)	
42613	0.005	42400	42400
42828.3	0.01	42400	42400
44631.6	0.05	42400	42400

In fact as seen from the Tab-2, eccentricity more than 0.005, even 0.05, will also ensure the disposal orbit requirement of no entry to the protected region, if the eccentricity vector is sun pointed at the time of re-orbiting, for the epoch considered viz., 2010 Jan 1.

4.2 Dependence On Orientation Of Eccentricity Vector For A Given Epoch

Case 1

Another epoch (2010-3-1) and eccentricity of 0.01 with perigee radius of 42400 km was chosen for analysing the effect of orientation of eccentricity vector at epoch, on the evolution of perigee radius and eccentricity magnitude over 100 Years. The study was carried out with the following initial conditions:

Epoch	: 2010-3-1
Inclination	: 0.04 deg
R.A of ascending node	: 0.0 deg
Semi-major axis	: 42828.3 km
Eccentricity	: 0.01
Perigee radius	: 42400 km
A/m	: 0.01 m^2/kg

The results obtained for various cases of 'e' vector orientation are shown in the following Tab. 3

Table 3. Effect of phasing for the epoch 2010-3 -1

Epoch elements		Minimum perigee radius over 100 Years (km)	No Entry into protected region
Argument of perigee (deg)	Phasing w.r.t. Sun (deg)		
341.8	0	42355	NOT MET
71.8	90	42370	MET
161.8	180	42325	NOT MET
251.8	-90	42381	MET
281.8	-60	42389	MET

The result shown above clearly brings out the strong dependence of long-term orbit evolution on the orientation of "e" vector at epoch. In the above case, positioning of epoch "e" vector 60 deg behind Sun, gives best results, by ensuring minimum perigee radius over 100 Years, well above the Protected Region.

Case 2

Results for another epoch (2028-1-1) and eccentricity of 0.0075 with perigee radius of 42400 km is presented below to demonstrate the effect of orientation of eccentricity vector at epoch, on the evolution of perigee radius over 100 Years. This study was carried out with the following initial conditions:

Epoch	: 2028-1-1
Inclination	: 0.04 deg
R.A of ascending node	: 0.0 deg
Semimajor axis	: 42720.4 km
Eccentricity	: 0.0075
Perigee radius	: 42400 km

The results obtained for various cases of 'e' vector orientation with initial conditions given above, are shown in the following Tab 4
.

Table 4. Effect of phasing for the epoch 2028-1 -1

Epoch elements		Minimum perigee radius over 100 Years (km)	No Entry into protected region
Argument of perigee (deg)	Phasing w.r.t. Sun (deg)		
101.4	+180	42376	MET
220	-61	42357	NOT MET
260	-21	42396	MET
281.4	0	42393	MET
300	+19	42382	MET
320	+39	42372	MET

Results presented in the above Tab 4, for e of 0.0075, shows that for the epoch of 2028-Jan-1, phasing of 21 deg behind sun maximises the minimum perigee over 100 years ensuring no entry into protected region.

Results for another epoch, 2010-jan-1, with high eccentricity of 0.05, for the same other initial conditions are given below to illustrate the sensitivity of minimum perigee over 100 years with respect to the phase angle of eccentricity vector.

Table 5. Sensitivity of phasing angle for 2010-1-1

Epoch elements		Minimum perigee radius over 100 Years (km)	No Entry into protected region
Argument of perigee (deg)	Phasing w.r.t. Sun (deg)		
241.4	-40	42369	MET
261.4	-20	42400	MET
281.4	0	42400	MET
301.4	+20	42289	NOT MET

Hence, it is clear that with appropriate positioning of the initial eccentricity vector with respect to Sun, the abandoned spacecraft will not enter into the protected region, even if the eccentricity of disposal orbit is more than 0.005.

Further typical results presented in the above three tables (Tab. 3,4 and 5) for eccentricity of 0.0075, 0.01 and 0.05 shows the existence of an arc length for ‘e’ vector orientation which can ensure no entry into protected region. Higher the eccentricity, lesser is the arc length.

5.0 CONCLUSIONS

Based on the studies carried out following conclusions are presented.

- The IADC guideline for perigee of the disposal orbits as a minimum of 235 km + allowance due to SRP above GEO, does indeed ensure no entry into the protected region, if the eccentricity magnitude is less than 0.005.
- Delta velocity required for achieving disposal orbit can be minimised, if sun pointed perigee with magnitude of ‘e’ as $0.0112 * C_r$ (A/m) and ΔH of 235 km are achieved at the end of re-orbiting maneuvers.
- Orientation of the initial eccentricity vector plays a significant role on the long-term evolution of eccentricity and hence on the perigee radius.
- Initial eccentricity magnitude more than 0.005, even 0.05, will also ensure no entry into the protected region, provided proper orientation of eccentricity vector is achieved after re-orbiting. The orientation of e vector to be achieved depends on the epoch.

6.0 REFERENCES

[1] IADC Space Debris Mitigation Guideline

[2] Lewis, H.G., Swinerd, G.G., Martin, C.E., and Campbell, W.S., The Stability of Disposal Orbits at Super-Synchronous Altitudes, Paper no. IAC-03-IAA.5.2.06

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