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Debris Mitigation in Geostationary Earth Orbit

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Abstract

The Inter-Agency Debris Committee recommendation for the reorbiting of geostationary satellites at the end of life involves an altitude increase of no more than 300km for most operational satellites. Although this reduces the collision probability in the geostationary ring itself, it does not remove the possibility of fragmentation debris produced in the reorbital region from passing into the geostationary ring. Unless the satellites are reorbited to a higher altitude than the current recommendations, the debris problem will continue to escalate to an unmanageable level. Due to mass and fuel budgets there are a limited number of available propulsive options, which can achieve the necessary reorbit.

The research focus selected has been solar sails and this paper describes ongoing research for the reorbit of geostationary satellites using this method.

Introduction

Since the first satellite was launched in 1957 mankind has continued to litter the immediate orbital vicinity of space. The amount of debris has grown to such a point that current and future missions have to take into account the very real collision threat imposed by this debris. If this danger is ignored in the future to the extent it is now, there is a serious risk the amount of free space remaining will become limited and a very valuable resource will be lost.

There are no real legal responsibilities for satellite operators and owners in any of the orbital regions of space. The UN Space treaty of 1967 did include a number of points which deemed the states from which the operator originates to be responsible for the satellites and any resulting debris but as most satellites nowadays are joint ventures between companies and countries these are not easily followed. Some operators insure their satellites at end of life for third party liability. There are also problems with detection limitations. In Low Earth Orbit items smaller than 10cm cannot be detected. This figure increases to 1m for objects in Geostationary orbit.

In the late 1980's some of the space-faring nations actually realised that the future debris problem was going to be an issue not just in geostationary, but in the whole near-earth orbit space. This prompted the main space agencies have developed orbital or space debris mitigation handbooks or guidelines [1-3].

Two regions of space were designated as protected zones. The first was LEO and included any area of orbital space to an altitude of 2000km. The recommendation for satellites at the end of life in this region was to deorbit them to an altitude where the decaying effect of the Earth's atmosphere would ensure the eventual re-entry of the satellite in no more than 25 years. The second protected region was that of Geostationary Earth Orbit altitude, 35786km +/- 235km.

The obvious recommendation for GEO was to reorbit the satellites at the end of life to a safe distance above the geostationary ring. This prompted the Inter Agency Space Debris council to produce a reorbiting formula [4] to be followed by all GEO users at the end of life:

$$\Delta H = 235 + 1000\ C_r\ A/m$$

ΔH = Change in altitude
C_r = Solar radiation pressure coefficient
A = Average cross sectional area
m = Mass of satellite

Initially satellites had area to mass ratios of approximately 0.03 which meant a relatively low reorbit of 300km. This value is steadily increasing as the size of satellites increases. It is more likely to be in the region of 0.05 or even 0.1 which indicates the necessity of a much larger reorbit value. However, even when the reorbit value was lower, this did not guarantee that such a manoeuvre would take place. In fact it was only in the eighties that satellite operators started to reorbit the satellites and not even then by the recommended amount.

Today the fraction of satellites that are reorbited according to recommendations is less than a third [5]. Most satellite operators will move the satellite out of the way just enough to place their new satellite in their licensed longitude location.

Debris fragmentation

The current debris situation is not exactly known although there are estimates. In terms of actual operating satellites the number is approximately 300. Of these the majority are grouped in the high density regions such as above America and Europe. Clearly this is due to the high demand for communications in those regions. Future plans for geostationary slots include satellite clusters which are collocated. SES Astra already operates 6 or 7 satellites in the same longitude slot. Obviously these are controlled more tightly as they are under the same operator. However if there were to be a control loss for just one of these satellites the dangers are clear.

There are two types of fragmentation which occur. The first is through explosion which is the most dangerous as it can generate a large amount of debris. The second is fragmentation caused by collision. These collisions can be between operational satellites, non-operational satellites, debris generated by other fragmentations or debris generated by operational workings.

Although explosions have been the most hazardous generator of debris to date, the risk of collision is intricately linked to the generation of debris through explosion. Another problem with collision is that the debris generated in these events will produce a debris cloud which is capable of spreading over thousands of kilometres[6] and so, even with the reorbit of satellites to 275km above GEO, this can produce a substantial threat to the active satellites in GEO and thus increase the probability of collisions.

Basic estimates for collisions in geostationary during a certain time period can be calculated using the population density and the collision cross section area. Typical values for worst case scenarios can be anything up to 1.5×10^{-6} collisions/year/m^2

Using debris propagation tools such as OrbitVis, collision probabilities can be calculated for any number of satellites. For a primary satellite in a geostationary orbit with approximately 1300 US Spacecom two line elements of all objects in and around the geostationary region, the probability of collision reached 1.6×10^{-8} collisions/year (Figure 1). With a close approach range of only 30km there were 29 close approaches.

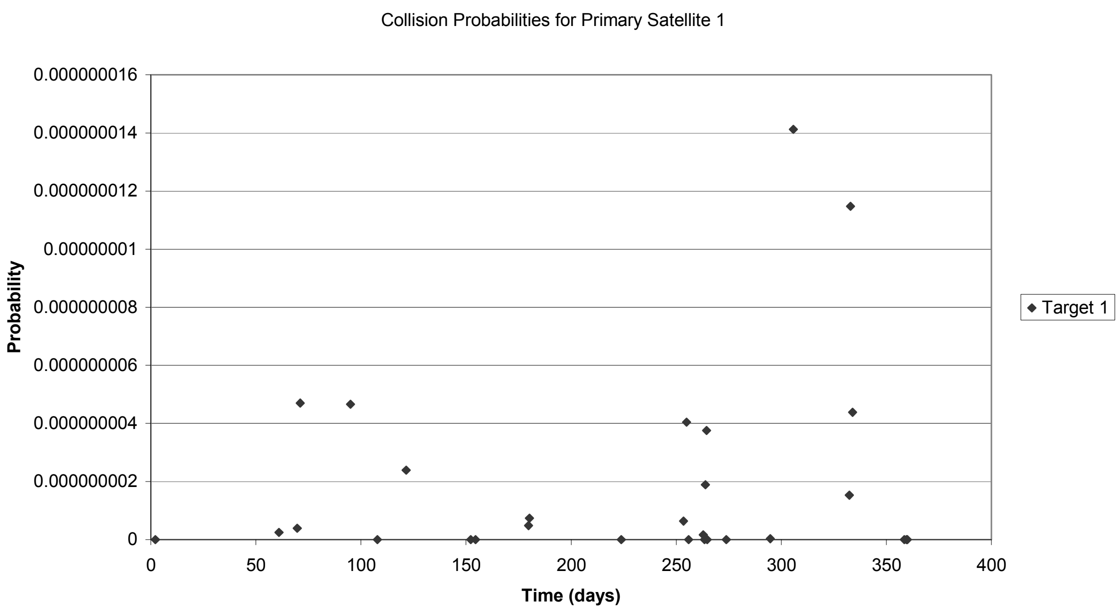

Figure 1: Collision probabilities for the primary geostationary satellite during 1 year.

The collision probability for satellites in the geostationary belt is very small and this is mainly due to the fact the satellites are moving in the same direction or are oscillating around one of the two stable longitude points at a relatively low velocity. However when the satellites are reorbited to a higher altitude, their drift rate increases substantially. During a ten year period a satellite with an orbit radius of 42464km will drift west at an increasing rate and change in inclination (Figure 2).

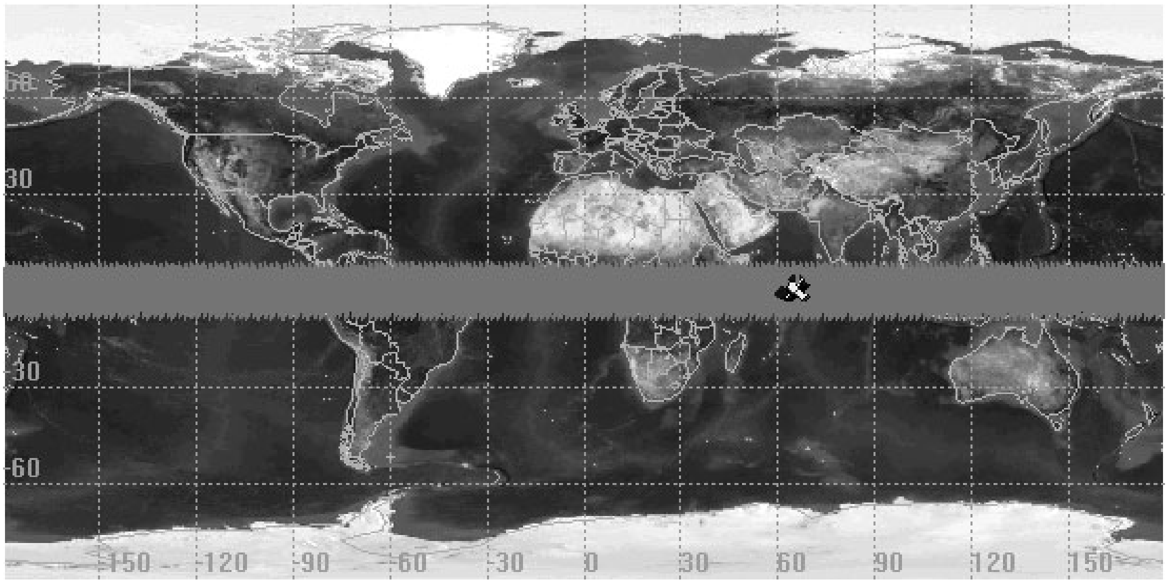

Figure 2: Ground trace of a geostationary satellite 300km above GEO over ten years.

Another problem with the small collision probability estimates is the current method of tracking and modelling. Detection limitations produce unrealistic figures. To date there have been three recognised fragmentations in GEO. It is unsure if there have been more than these, as is unknown exactly how many pieces were generated in those particular fragmentations.

Solar Sails

Solar sailing is one of the simplest propulsion options involving the transfer of momentum from solar photons to the solar sail. The solar radiation pressure force for perfectly reflected photons is double that of photons absorbed into the sail.

The defining measurement for solar sailing is the characteristic acceleration, a measurement of the acceleration achieved by a sail at a distance of 1 AU from the sun. The control of the solar sail is achieved by controlling the angle at which the photons or sunlight hits the sail. This will either increase or decrease the energy of the orbit therefore raising or lowering the orbit of the satellite.

For a perfectly reflecting sail the magnitude of the force is written as:

$$F = 2\frac{S_0}{c}A\left(\frac{R_0}{R}\right)^2 \cos^2 \alpha \qquad (1)$$

$S_0 = 1368\ \mathrm{W/m^2}$

Since R_0 is the average distance from the Earth to the Sun and R is the actual distance from the sun, for solar sails at Geostationary this value approximates to 1 and can be neglected.

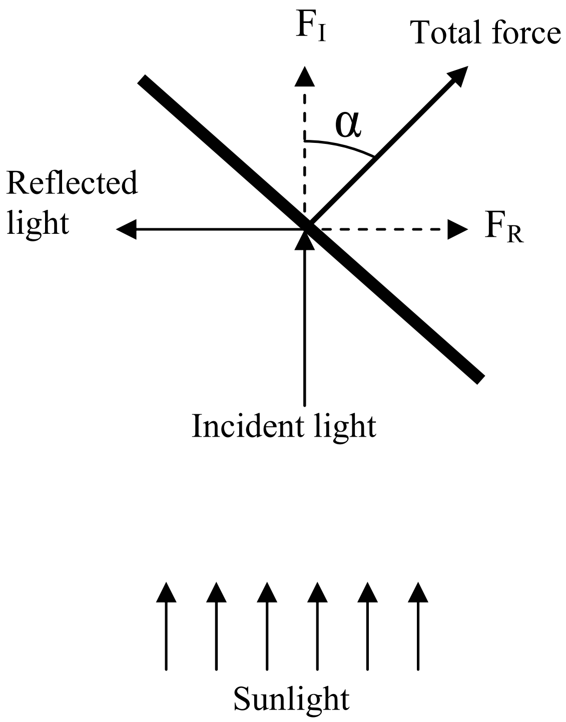

Figure 3: Force vectors of a perfectly reflecting sail

When divided by the total mass of the satellite, m_{sc}, the acceleration is:

$$a = 2\frac{S_0 A}{cm_{sc}}\cos^2\alpha \tag{2}$$

However, taking into account the reflectivity, absorptivity, and transmission of the sail, an efficiency factor, η, is included in the equation. This allows for less than perfect reflection of the sail and also any warping of the sail material.

$$a = 2\eta\frac{S_0 A}{cm_{sc}}\cos^2\alpha \tag{3}$$

The biggest advantage of Solar sailing is that it has an unlimited fuel resource. This appealing characteristic makes it ideal for missions anywhere in the solar system. In terms of time periods, those for planetary missions would be small in comparison with standard chemical propulsion due to the accelerative nature of sails. For short trips it is a less obvious candidate due to the length of time taken to accelerate to a useful velocity. However, at the end of a satellite life, the time taken to reorbit it is only important in terms of the area time product and thus the possibility of collision with other satellites in the region.

As mentioned previously, the current difficulty with end of life reorbit for satellites in GEO is that the necessary fuel to meet the minimum guideline of 275km above geostationary is equivalent to an extra few months of operational revenue. There are some satellite operators who successfully follow the recommended guidelines. Unfortunately the majority will only perform a partial reorbit, enough to make way for the next generation of their satellites, with a large number who perform no reorbit at all. Using an autonomous solar sail as a reorbit mechanism would remove the temptation of extending the useful life of the satellite by those extra few months. It would also be the solution if the ability to command or control the satellite.

Even if the minimum reorbit criteria are met, this works out as 5kg of fuel per 1000kg of satellite. Current designs for solar sails work out as 40kg for an area of $400m^2$. However, if the debris possibilities are taken into account and satellites were to be reorbited to a much safer altitude of at least 37786km, the fuel mass budget increases to approximately 35kg per 1000kg of satellite. Using current solar sail technologies this is a realistic mass for a reorbit device.

Using a solar sail of only $100m^2$, which is comparable to solar panel area, and an efficiency of only 85%, it is theoretically possible to reorbit a 3000kg satellite to an altitude 400km above geostationary in only 649 days. Although this sounds like a long time it is important to remember that at this point the satellite is non-functional and no revenue is being wasted due to the lengthy reorbit time.

$$\Delta V \approx a_0 T \qquad\qquad a_0 = \frac{S_0 2\eta}{\sigma} = 2.58\times10^{-7} \mathrm{m/s^2} \tag{4}$$

ΔV for 400km reorbit = 14.49m/s

$\therefore$ T = 56075851s = 649 days

This reorbit calculation, even at a less than perfect efficiency, is based on the reorbit of the satellite with a constant increase in the energy of the orbit, i.e. the sail will be at the optimum angle for propulsion at all times during the orbit. To control the sail to such a high degree not only increases the weight and sophistication of the solar sail device but also the cost, thus defeating the initial objective.

The main difficulty with solar sails is reaching a level of autonomy that would be reliable, whilst keeping the mass low. To have a fully autonomous solar sail is easily attainable but would require repetition of heavy control systems that would make it an unrealistic choice. Some of the alternative options available are:

- One sail deployed from side of satellite
- Thermally adaptive sails
- Multiple sail configuration
- Single sail with attitude control vanes

All of these options rely on having a maximum effective reflective area during the positive half of the orbit (travelling away from the sun) and a minimum during the negative (travelling towards the sun). Although this will increase the energy of the orbit, it will be less than the orbit described above.

The simplest solution is to use one sail that is deployed from the optimum face of the satellite. There are a number of assumptions made for this option including the satellite is not spinning at the time of deployment. The solar sail itself would be constructed of highly reflective material on one side of the sail and an absorptive material on the other side (Figure 4).

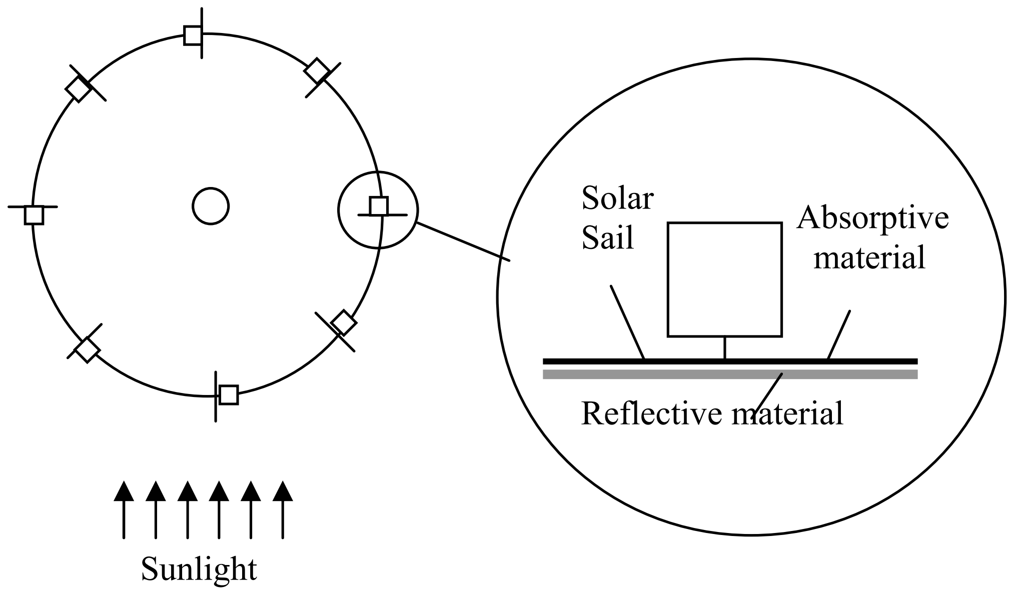

Figure 4: Deployment configuration for a basic solar sail from an Earth pointing satellite.

The immediate problem with this basic design is that the energy increase of the orbit will be reduced by substantial amount. Depending on how reflective and absorptive the materials of the sail are, during the negative half of the orbit the photons absorbed by the solar sail will impart a maximum deceleration force of $4.56\times10^{-6}Nm^{-2}$ when they impact the sail. This will significantly increase the time period for a reorbit manoeuvre. The immediate concern would be how quickly the satellite could be removed from the orbital slot to make way for the next generation satellite.

By using Satellite Tool Kit analysis to model the sail described above, it can be demonstrated that even with poor efficiency, poor reflectivity and a small cross sectional area, the orbit can be raised by 10km in three months (Figure 5). This would be enough for the slot operator to safely place their next satellite in orbit.

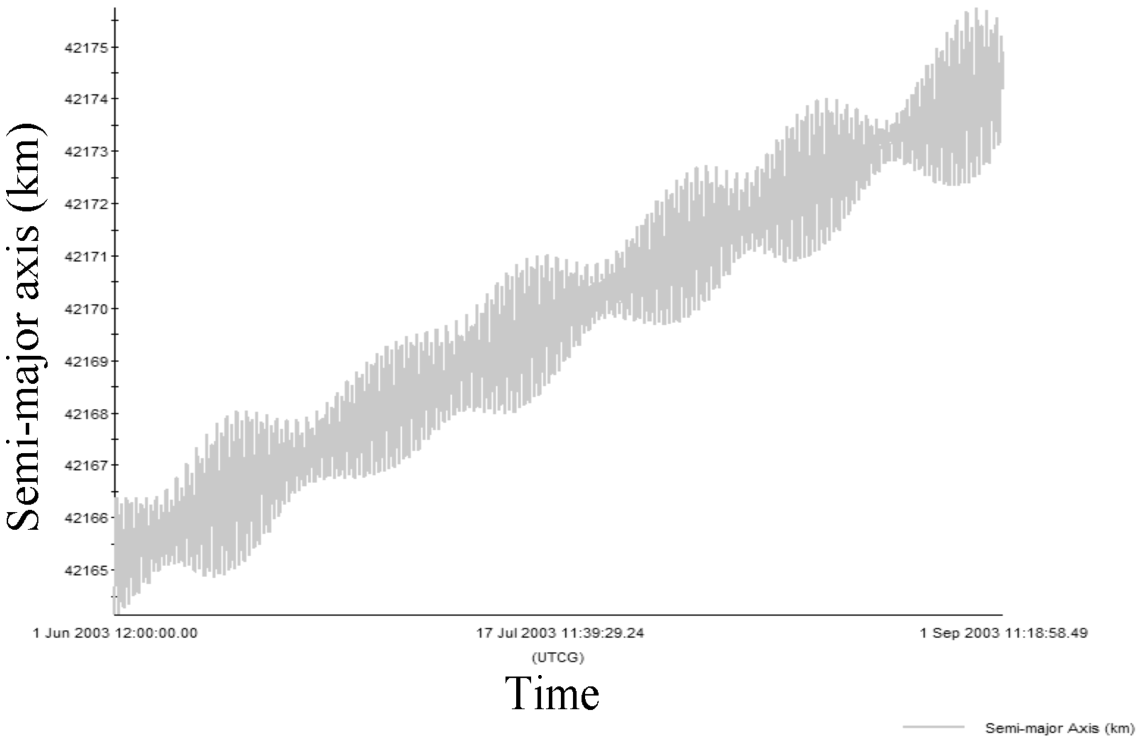

Figure 5: Increase in semi-major axis in the first three months of reorbit

Ongoing work

The thermal characteristics of the solar sail are important in terms of the sail design and reorbit capability. Current work includes thermal modelling of solar sails using I-DEAS/TMG. This program allows comprehensive modelling of radiative heat transfer of satellites in Geostationary orbit.

This is being used simultaneously with Matlab to model the attitude behaviour of the satellite/solar sail. The ultimate goal of this research is to produce a solar sail device that is capable of being deployed in any random attitude, i.e. the satellite is spinning, but still able to achieve a reorbit altitude of at least 36086km.

Conclusion

Geostationary Earth Orbit is incredibly valuable and finite. The current mitigation measures in place to limit the amount of debris generated are short term and unenforced. The reorbit of satellites in Geostationary to a safe altitude is both necessary and responsible.

Reorbiting satellites to an altitude 300km above GEO is easily achievable using fully controlled solar sails. The ability to reorbit using low mass autonomous sails requires a non-trivial geometrical solution.

References

[1] The ESA Space Debris Mitigation Handbook.
[2] NASA Orbital Debris Mitigation Standard, NASA Safety Standard NSS 1740.14 .
[3] NASDA Space Debris Mitigation Standard.
[4] Loftus, J. P., The Management of the Geostationary Environment, *Acta Astronautica*, Vol. 44, Nos. 7-12, pp 645-648, 1999.
[5] Jehn, R. and Hernandez, C., International Practices to Protect the Geostationary Ring, *Space Debris*, Vol. 1, pp 221-223, 2001.

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