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On the use of a variable coefficient of reflectivity associated with an augmented area-to-mass ratio to de-orbit CubeSats

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Abstract. The number of missions involving CubeSats increases each year, including Brazilian missions. The simplification of the projects, aiming smaller structures and less weight, makes the missions of CubeSats cheaper compared to large satellites. Another point is that, in recent years, the concern about the increase of objects in orbit becomes important, since operational satellites and upper stage of launchers that populate the Low Earth Orbit region (LEO region) are more numerous. To try to reduce the impact of this problem, there is a consensus that objects in LEO region should re-enter in the atmosphere within 25 years. This is becoming a limit for projects of CubeSats. In this scenario, the present paper presents an alternative to this problem, which is the use of devices of area augmentation and/or variation of the coefficient of reflectivity in a CubeSat. By using these devices, it is possible, even in regions where the air density is not sufficiently large or where there is no more atmosphere, to force the decay of a CubeSat. Then, one can carry out missions of CubeSat to aim for higher altitudes than current missions, expanding the possibilities of applications.

1. Introduction

In recent years the number of missions with CubeSats has increased. The low cost of assembly and launch of CubeSats, due to the reduced weight and size, make them ideal for many different applications. For this reason CubeSats are becoming more popular in the Brazilian space program [1]. Some of these Brazilian CubeSats are already in space and others are under development. Examples of Brazilian CubeSats are ITASAT, CONASAT and NANOSATC-BR1. Along with the interest in CubeSats, not only in Brazil, but worldwide, there is a growing concern about the accumulation of objects in orbit. Among these objects are operational satellites, disabled satellites (not controlled), upper stages of launchers and fragments of collisions. To try to reduce this problem, some international organizations, such as the Inter-Agency Space Debris Coordination Committee (IADC), NASA and ESA, have issued documentation on the subject in order to regulate disposal practices for satellites at the end-of-life. Within the discussion on the subject there is a consensus that all the objects in Low Earth Orbit region (LEO region) should re-enter in the atmosphere in up to 25 years [2]. Another important concept is the de-orbit, which refers to the forced removal of any object in orbit [3]. Some of the de-orbit techniques use space tethers, thrusters, solar sails and low thrust devices. New concepts of de-orbit has



been studied in recent years, one of them related to techniques that use solar radiation pressure and atmospheric drag to force the decay of a body in orbit around the Earth [4, 5, 6, 7]. These devices are less complex than a solar sail. They have variable area (like an inflatable balloon) and variable coefficient of reflectivity that, depending on its design, eliminates the need of attitude controllers. The possible elimination of the need of attitude controllers makes these type of devices ideal for use in CubeSats. At this time there are companies developing inflatable balloon devices¹. Such devices would be inactive during the years of the mission and, at the end of the mission, a balloon is inflated in the bodies within the LEO region. The increase of the atmospheric drag force caused by the increase of the area-to-mass ratio causes the decay of the body. However, the low density of the air in high altitude limits the use of this type of device to low altitudes, or it is necessary the use of a large area increase [8]. The solution presented in this study relates the use of devices able to change its area, as the balloon mentioned above, in addition to devices with the ability to change the coefficient of reflectivity. A simple theoretical design of this device would be a balloon coated by a fabric able to change its color [9]. This type of device has a valid application in CubeSats missions and its use can increase the chances of future missions in the LEO region at higher altitudes, which could decrease the number of CubeSats in the formation of a constellation. The present paper does not address the constructive aspects of the device, but only the orbital dynamics of a CubeSat is studied. The study is done numerically, via integration of the equations of motion of a perturbed satellite. The next section shows the method used in detail.

2. Method

The study is done numerically through the integration of the equations of motion of a perturbed satellite. These equations are:

$$\ddot{\mathbf{r}} = -\frac{GM_E}{r^3}\mathbf{r} - GM_\odot \left(\frac{\mathbf{r} - \mathbf{r}_\odot}{|\mathbf{r} - \mathbf{r}_\odot|^3} - \frac{\mathbf{r}_\odot}{|\mathbf{r}_\odot|^3} \right) - GM_M \left(\frac{\mathbf{r} - \mathbf{r}_M}{|\mathbf{r} - \mathbf{r}_M|^3} - \frac{\mathbf{r}_M}{|\mathbf{r}_M|^3} \right) + \mathbf{P}_G + \mathbf{P}_D + \mathbf{P}_{SRP} \quad (1)$$

where G is the gravitational constant, M_E , M_\odot , M_M are the masses of the Earth, the Sun, and the Moon, respectively. \mathbf{r} , \mathbf{r}_\odot , \mathbf{r}_M are, in order, the position vector of the satellite, the Sun, and the Moon. The second and third terms of Equation 1 can be found in Reference [10] and are due to the perturbations of the Sun and the Moon, respectively. \mathbf{P}_G is the acceleration due to the Geopotential, \mathbf{P}_D is the acceleration due to the atmospheric drag and \mathbf{P}_{SRP} is the acceleration due to the solar radiation pressure. The numerical integration of Equation 1 is made using the RADAU integrator [11].

The model of the acceleration due to the Geopotential used here can be found in Reference [12]. However, an important problem in the Geopotential model is related to the proper choice of the order and degree to be considered. It is chosen, for the Geopotential model used here, order and degree 8. Since the decay time is less than 25 years, and we are interested in the effects of solar radiation pressure and atmospheric drag, the quantity of harmonics in the Geopotential is acceptable. The calculation of the acceleration due to the atmospheric drag is given by[10]:

$$\mathbf{P}_D = -\frac{1}{2} C_D \rho \alpha |\mathbf{v}_r|^2 \frac{\mathbf{v}}{|\mathbf{v}|} \quad (2)$$

where C_D is the aerodynamic coefficient. It is considered the value of 2.2. \mathbf{v}_r is the relative velocity between the satellite and the atmosphere. \mathbf{v} is the velocity of the satellite. α is the

¹ <http://www.gaerospace.com/aeroassist/gossamer-orbit-lowering-device-gold-for-low-risk-satellite-de-orbit/>

area-to-mass ratio. The minimum initial value (α_0) is equal to $0.007 \text{ m}^2/\text{kg}$. ρ is the atmospheric density. The paper considers the density up to 900 km altitude and used a table of medium density, which is made using the JB2008 atmospheric model. The calculation of the acceleration due to the solar radiation pressure is given by [10]:

$$\mathbf{P}_{SRP} = -P_{\odot} C_r \nu \alpha \frac{\mathbf{r}_{\odot}}{|\mathbf{r}_{\odot}|^3} AU^2 \quad (3)$$

where P_{\odot} is the solar radiation pressure, determined by the solar flux. Because the distance between the Earth and the Sun is very large, P_{\odot} is considered constant in the neighborhood of the Earth, so $P_{\odot} = 4.56 \times 10^{-6} \text{ Nm}^{-2}$. It is worth mentioning that the area-to-mass has the same value used in the calculation of the acceleration due to the atmospheric drag. C_r is the reflectivity coefficient, AU is the Earth-Sun distance and ν is the shadow function. The shadow model used is found in Reference [10], and the eclipse factor takes into account the direct solar radiation, which has value 1, when totally exposed to the Sun, umbra, with value 0 and penumbra, with value between 0 and 1.

As mentioned before, this paper considers a satellite which carries a device that increases its area and which is also able to vary its reflectivity coefficient. As Figure 1 shows, C_r has a maximum value of 2 when $\langle \mathbf{P}_{SRP}, \mathbf{v} \rangle < 0$ and the minimum value of 1, in any other situation [6].

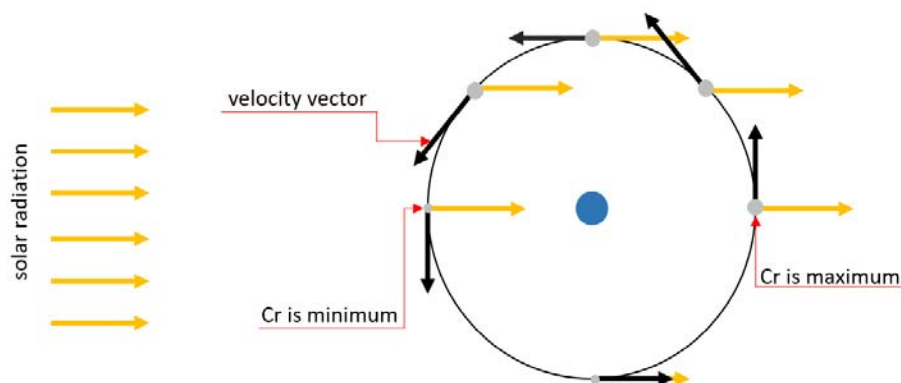


Figure 1. C_r variation control.

3. Results and Discussion

In this section, the results of the simulations are showed. In order to reach the highest altitudes possible, respecting the rule of 25 years, it is not considered the mission time to later trigger the device, i.e., the device has been active since the beginning of the mission. The initial conditions of the Sun and the satellite, common in all simulations, are presented in Table 1.

Table 1. Initial conditions for the Sun and the satellite.

	Sun	Satellite
Semi-major axis [km]	149512533.86795	-
Eccentricity	0.01617	0.005
Argument of perigee [degree]	284.245	0.0
Longitude of the ascending node [degree]	0.005	0.0
Mean anomaly [degree]	355.910	0.0
Inclination [degree]	23.437	30.0

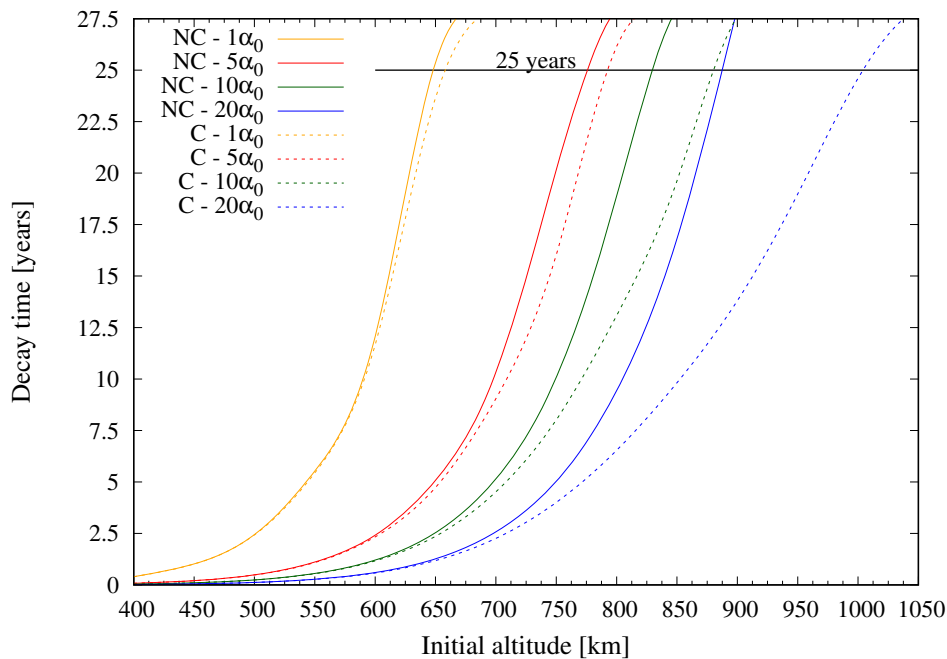


Figure 2. Cubesat decay time for the different situations.

Figure 2 presents the results for all simulations. The continuous curves (NC) show the results for the simulations where there is no presence of a control varying the coefficient of reflectivity. Its value is always equal to 1. The dashed curves (C) show the simulations where there is an active control varying the coefficient of reflectivity between 1 and 2, as previously mentioned. Simulations are done for different values of α_0 , where: $\alpha = 1\alpha_0 = 0.007 \text{ m}^2/\text{kg}$, $\alpha = 2\alpha_0 = 0.014 \text{ m}^2/\text{kg}$, $\alpha = 5\alpha_0 = 0.035 \text{ m}^2/\text{kg}$, and $\alpha = 20\alpha_0 = 0.14 \text{ m}^2/\text{kg}$. As mentioned before, the presence of the atmosphere is considered up to 900 km. However, between 900 km and 700 km, the impact of the solar radiation pressure overcomes the effect of the atmospheric drag, which becomes dominant above 700 km [6]. We can see in Figure 2 that the difference between controlled and not controlled decays increases after 700 km of altitude. Moreover, the analysis of Figure 2 shows that:

- The decay times are near the same value up to 450 km of altitude for $5\alpha_0$, $10\alpha_0$ and $20\alpha_0$, and up to 500 km altitude for $10\alpha_0$ and $20\alpha_0$;
- In all cases, the decay time decreased as α increases. Ex.: The decays times for $\alpha = 5\alpha_0$, $\alpha = 10\alpha_0$ and $\alpha = 20\alpha_0$ at the altitude of 600 km show a decrease of approximately 10 years, 11.5 years and 12.25 years with respect to $\alpha = \alpha_0$;
- The decay times decrease as the value of α increases, from certain altitudes, when we compare the curves without control (NC cases) and with active control (C cases). Ex.: The decay times at altitude of 750 km show a decrease in the time of decay, for cases C with respect to NC cases: 4.8 years for $5\alpha_0$, 2.5 years for $10\alpha_0$ and 1.25 for $20\alpha_0$;
- The yellow curves obtained for α_0 show little difference in the times of decay up to approximately 600 km of altitude. The curves NC and C, with respect to the rule of 25 years, reach approximately 650 km and 662.5 km of altitude, respectively;
- The red curves obtained for $5\alpha_0$ show little difference in the times of decay up to approximately 600 km of altitude. The curves NC and C, with respect to the rule of 25 years, reach approximately 775 km and 795 km of altitude, respectively;

- The blue curves obtained for $10\alpha_0$ show little difference in the times of decay up to approximately 650 km of altitude. The curves NC and C, with respect to the rule of 25 years, reach approximately 830 km and 880 km of altitude, respectively;
- The green curves obtained for $20\alpha_0$ show little difference in the times of decay up to approximately 650 km of altitude. The curves NC and C, with respect to the rule of 25 years, reach approximately 830 km and 880 km of altitude, respectively.

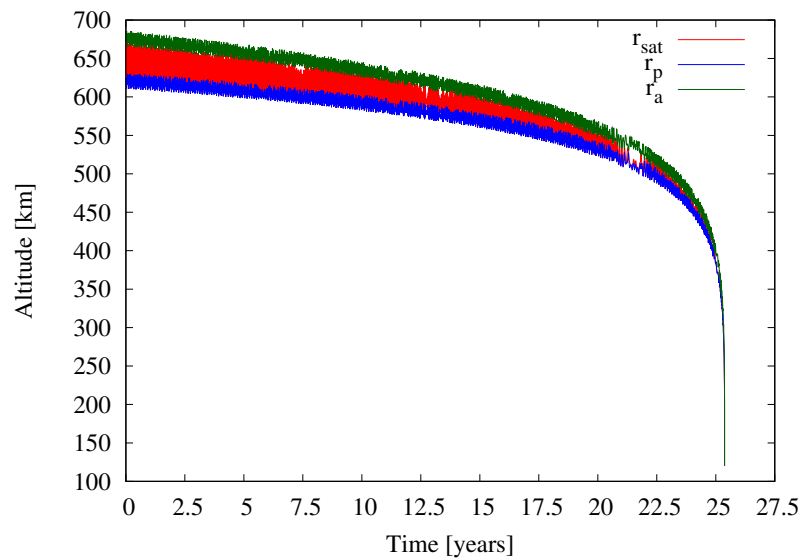


Figure 3. Decay of CubeSats from 650 km of altitude, without device and considering $\alpha=0.007$ m^2/kg .

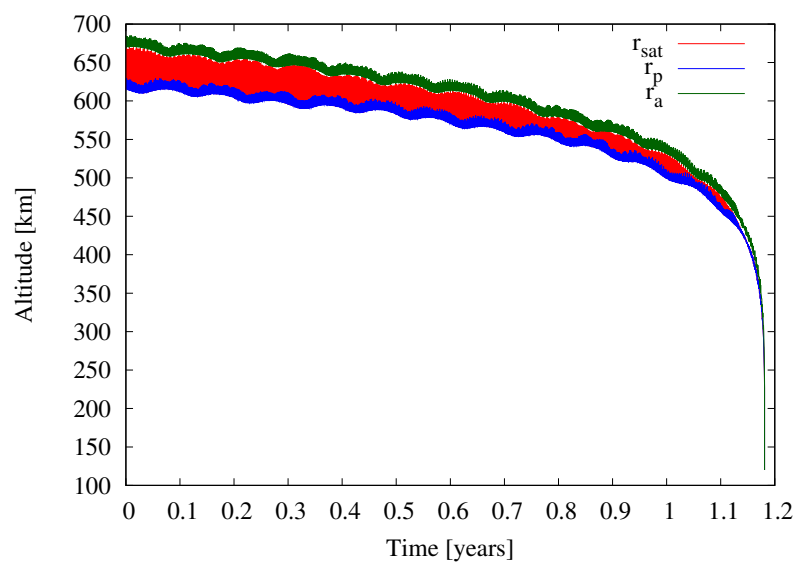


Figure 4. Decay of CubeSats from 650 km of altitude, with device and considering $\alpha=0.14$ m^2/kg .

Figures 3 and 4 show the decay of a CubeSat from 650 km, where r_{sat} is the vector of the satellite position, r_p is the perigee radius and r_a is the apogee radius. Figure 3 shows the decay for the case without the device and considering $\alpha = 0.007 \text{ m}^2/\text{kg}$. We can note that 650 km is near the initial altitude limit, so a CubeSat with $\alpha = 0.007 \text{ m}^2/\text{kg}$ respect the rule of 25 years for reentry. Figure 4 shows the decay for the case with a device to increase the area, where $\alpha = 0.14 \text{ m}^2/\text{kg}$, and able to vary the coefficient of reflectivity between 1 and 2, according to the control presented previously. When comparing Figure 3 with Figure 4, the advantage of using the proposed device is clear, where the decay time has dropped from approximately 25 years to approximately 1.18 years.

4. Conclusion

The present study shows that the area increase device and/or variation of the reflectivity coefficient increases the maximum altitude for missions where the rule of 25 years is respected and decrease the decay time for missions within the LEO region. The device of area increase and variation of the coefficient of reflectivity allows missions that respect the rule of 25 years where the initial altitude is in regions that the acceleration due to the atmosphere is negligible (above 900 km in our study), where the acceleration due to the radiation pressure being controlled is enough to cause the complete decay of the satellite. Future studies will make a broader analysis of the use of different coefficients of reflectivity and more area-to-mass settings, to try to increase even more the maximum altitude for missions of CubeSats, with respect the rule of the 25 years.

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