ORBITAL DEBRIS REMOVAL BASED ON A HYBRID APPROACH COMBINING OPTIMAL CONTROL AND ENVIRONMENTAL PERTURBATIONS

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ABSTRACT

Ever since the launch of Sputnik 1 in 1957, the exploitation of the space environment has been growing every year and the number of orbital debris has been naturally expanding. Beyond a given threshold, it is predicted that orbital debris density will create an uncontrolled chain reaction of collisions: the Kessler effect. Low-Earth Orbit (LEO) debris removal could avoid this reaction. In this study, debris removal consists in moving debris from their initial orbit to a lower terminal orbit leading to their disintegration in Earth's atmosphere. Using orbital perturbations, it is possible to reduce the quantity of propellant necessary to realize the de-orbit manoeuvers. The objective of the present study is to develop an autonomous control strategy which optimizes the propellant and duration to modify the orbit of debris using environmental perturbations.

The greatest density of debris can be found in the LEO region, on sun-synchronous near-circular orbits of about 800 km altitude. The method chosen for the debris removal is the use of a "space cleaner" satellite equipped with electrical propulsion. The study focuses on the de-orbiting phase; the rendezvous and capture between the "space cleaner" and the debris are not considered. The de-orbit trajectory which optimizes fuel and time is computed using a hybrid approach combining optimal control including the J2 perturbation and strategies that use the atmospheric drag, in particular the diurnal bulge. Depending on the position of the terminal orbit's ascending node, the spacecraft could encounter up to two times the drag force if it is situated in the bulge. This study analyses and compares two types of terminal orbits: circular and eccentric. The eccentric orbit manoeuver consists of lowering the perigee in the bulge while maintaining the apogee at 800 km altitude. Simulations are run and validated using Matlab/Simulink. The atmosphere model is based on the Jacchia 1977 model.

As a first step, using the NASA Debris Assessment Software (DAS), the predicted orbital lifetime for different terminal orbits has been computed. It has been observed that a circular orbit has an equivalent elliptical orbit in terms of final de-orbit time. With an initial circular orbit at 800 km altitude, an elliptical orbit with its apogee at 800 km altitude should have its perigee about 120 km lower than the equivalent circular orbit altitude to have the same remaining orbital life. For instance, a circular orbit 400 x 400 km (apogee altitude x perigee altitude) is equivalent, in terms of orbital lifetime, to an elliptical orbit 800 x 280 km. This equivalence correlates directly to the energy loss per orbit, as can be seen in Figure 1 where the specific energy loss per orbit is illustrated as a function of the mean local time of ascending node (MLTAN). The advantage of this strategy is that the achievement of the eccentric orbit costs less propellant than getting to the equivalent circular orbit. This result is computed using optimal control including

the J2 perturbation. The time to complete the manoeuver is fixed and is computed such that the thrust magnitude does not exceed 70 mN, based on the SMART-1 satellite. Table 1 shows examples of time and propellant cost for different manoeuvers. Evidently, there is a compromise to be made between the fuel expense and the time of the manoeuver.

In the next step, the manoeuvers computed above are fed into a dynamics simulator that includes the atmospheric drag with the diurnal bulge. Different test cases are presented in which the exospheric temperature (T_∞) and the area to mass ratio (AtoM) of the satellite are varied. The exospheric temperature depends on the 11-year solar activity cycle and the area to mass ratio can be modified by an inflatable device. As can be seen from Figure 1, the control of the perigee generates an energy loss per orbit for the eccentric orbit always greater than the circular orbit.

- Test 1: T_∞ = 1000 K, AtoM = 0.04 m^2/kg
- Test 2: T_∞ = 1500 K, AtoM = 0.04 m^2/kg
- Test 3: T_∞ = 1000 K, AtoM = 0.4 m^2/kg
- Test 4: T_∞ = 1500 K, AtoM = 0.4 m^2/kg

Figure 1. Delta energy/orbit for circular versus eccentric orbits

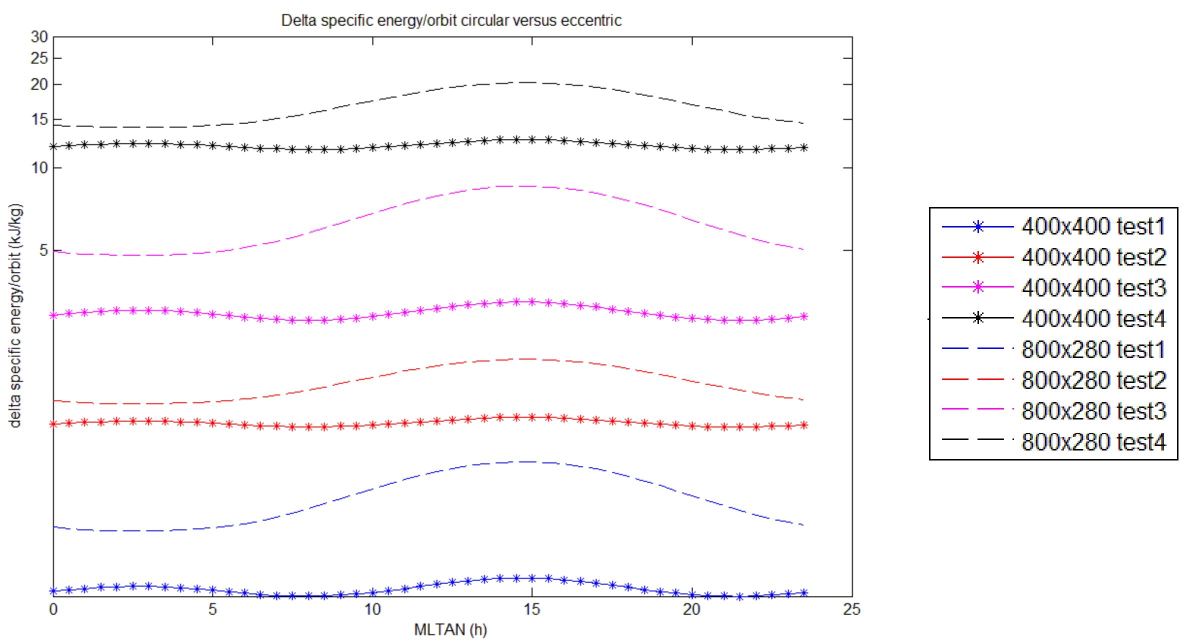

Table 1. Manoeuver time and propellant cost for different final orbits from initial circular orbit 800 km altitude

Final orbit	Test 1 (with J_2 + drag + diurnal bulge)			Test 4 (with J_2 + drag + diurnal bulge)		
	Manoeuver time (days)	Propellant mass cost (kg)	Propellant savings (%)	Manoeuver time (days)	Propellant mass cost (kg)	Propellant savings (%)
800 x 280 km	55.6	6.1135	13.31	35.8	3.9988	13.66
400 x 400 km	38.3	7.0518		24.7	4.6317	
800 x 230 km	60.2	6.7408	14.44	37.4	4.2491	11.06
350 x 350 km	42.6	7.8787		25.6	4.7775	

Table 1 shows the results of the de-orbit manoeuvers for test 1 and test 4 only. These results demonstrate that the strategic use of atmospheric drag can reduce the propellant consumption to achieve equivalent terminal orbits. The article will show the results for the other test cases as well as for many other final orbits.