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USING LOW THRUST PROPULSION TO PULSED ORBIT CONTROL OF THE LUNAR POLAR SATELLITES

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Abstract. It is well known that lunar satellites in polar orbits suffer a high increase on the eccentricity due to the gravitational perturbation of the Earth. That effect is a natural consequence of the Lidov-Kozai resonance. The final fate of such satellites is the collision with the Moon. Therefore, the control of the orbital eccentricity leads to the control of the satellite's lifetime. In the present work we study this problem and introduce an approach in order to keep the orbital eccentricity of the satellite at low values. We studied different arcs of active lunar satellite propulsion, centered on the orbit apoapsis or periapsis, in order to be able to introduce a correction of the eccentricity at each cycle. The proposed method is based on an approach intended to keep the orbital eccentricity of the satellite at low values.

Keywords. Low Thrust Propulsion, Lunnar Satellites, Polar orbits

1 Introduction

Recently, the ESA's Small Mission for Advanced Research in Technology (SMART-1) completes the mission of scanning the Lunar surface from its own orbit. It was the first spacecraft which achieved the Moon's orbit employing plasma rockets (for details, see e.g., [2]), and

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therefore become a very successful start of an epoch of Moon exploration using plasma technology.

Certainly, in few years it will be build several Lunar bases, mainly in the near poles region, where the probability to found water reservoirs is larger. Because of that, the need of reconnaissance satellites in polar orbits is increasing. However, high inclination orbits aroundm the Moon are susceptible to the Lidov-Kozai resonance [3], which will limit the satellite lifetime by increasing the orbit eccentricity, until the collision between Moon's surface and the satellite.

At the Lidov-Kozai resonance scenario, eccentricity and inclination of a small object's orbit are coupled according third Delaunay momentum, therefore a perturbation would have no influence on semi-major axis. Those quantities could undergo changes and, for high inclined orbits, the eccentricity will grow without control[4].

Winter et al (2009) [6] proposed, in a previous work, a method to increase the lifetime of polar lunar satellites in near circular orbit using low thrust propulsion and the locally optimal thrust. This approach is suitable considering a costly attitude control and missions in which large fluctuations of radial distance are acceptable.

Hall-effect plasma thrusters are now a very good option for space probes primary propulsion and also for station keeping of medium and large satellites. This is because of their high specific impulse and thrust efficiencies, which is related to an economy in terms of propellant mass utilization and satellite lifetime. First Hall thrusters were developed in mid 1950's, and they were also called Closed Drift Thrusters, as become clear that the closed electron drift current inside the source channel, promoted by the crossed electric (axial) and magnetic (radial) fields configurations, is responsible for the acceleration of the plasma beam (for a historical review, see [1]). Instead of a good comprehension of the acceleration mechanism and the steady state plasma dynamics, today a lot of problems on the role of electrons transport and plasma fluctuations and instabilities still open (e.g. [5]).

In the present paper, we studied the employment of permanent magnets Hall-effect thrust to introduce a correction of the eccentricity at each cycle of lunar orbit, and therefore sustain very low values to the eccentricity of the orbit. In the next section we describe these thruster, and after that we describe the simulation methods and we show and comment our results.

2 Methodology

To perform the numerical simulations, has been considered the 3-Body problem:Moon, Earth and satellite. In that case, the coordinate system was chosen in the Earth-Moon barycenter. At each integration cycle, the classical keplerian osculating elements were computed and monitored.

During the orbit propagation, when the value of true anomaly stay between certain values of angular position, it was satisfied one condition for turning on the propulsion system. Those angular regions were defined as propulsion arcs and centered on the orbital apoapsis or periapsis.

The approach proposed is a very simple one and the idea is to introduce a correction on eccentricity by means of constant vector thrust. This thrust is activated only inside the propulsion arcs centered on apocentre or pericentre.

The procedure consists in fixing a limit value to the eccentricity emax and then activating the thrust every time the spacecraft is inside any of propulsion arcs if that limit value is exceeded.

As the value of orbital eccentricity was limited, the values of distances of apocentre or pericentre needed to be kept inside certain intervals. If the spacecraft was inside the propulsion arcs and if those distances stay beyond these intervals, it was satisfied all the conditions for turning on the propulsion system. As one can see in the Figure 1, if the spacecraft was inside the pericentre propulsion arc and its orbital apocentre distance was greater than $(1+e_{max})a$ (where a is the orbital semi-major axis) or the spacecraft was inside the apocentre propulsion arc and its orbital pericentre distance was less than $(1-e_{max})a$, the propulsion system was activated in order to maintain the value of orbital eccentricity below e_{max}.

The set of initial conditions for the simulations were listed on Table 1. We used typical values for the thrust and spacecraft mass as well we input the expected value for altitude of a polar orbit reconnaissance satellite.

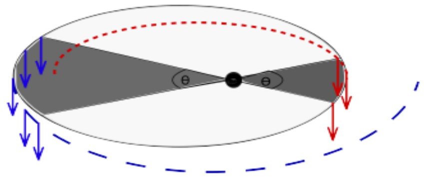

Figure 1: Schematic view of maneuver plan.

Table 1: Initial conditions used on simulations.

Parameter	Initial Value
Spacecraft mass (*kg*)	1,000
Thrust magnitude (*mN*)	100
Altitude (*km*)	500
Inclination (*deg*)	90
Angular aperture of propulsion arcs θ (*deg*)	6 to 122
Limit eccentricity (e_{max})	0.05
Initial eccentricity (e_0)	0.04

3 Results

Following, the results of our simulations. In each run has been measured the running time (R_t) needed to keep the excentricity below e_{max} during mission times between 0 and 10,000 days. From this data, was obtained an empirical expression for the uptime as function of duration of mission.

One can see in Figure 2 that the growth of eccentricity was controled during all the mission time. In that case the colision with Moon surface was properly avoided.

Figure 3 shows the uptime of PMHT as function of duration of mission. Samples of the results for θ equal to 12°, 32° , 52° and 72° are shown in both figures.

It was performed many linear fit ($R_t = A(\theta)[t_k - 339.933]$) for every tested angular apertures. The values of $A(\theta)$ were plotted in Figure 4 where can be seen that there is a minimum when θ is around 12 degrees. Figures 5 and 6 presents basically the same results as previous ones. The main diference is the set of maximum eccentricities maintained.

One can see (from the Figures 5 and 6) that the sooner the thrusters are activated the lower will be the angular coeficients $A(\theta)$ or the running time.

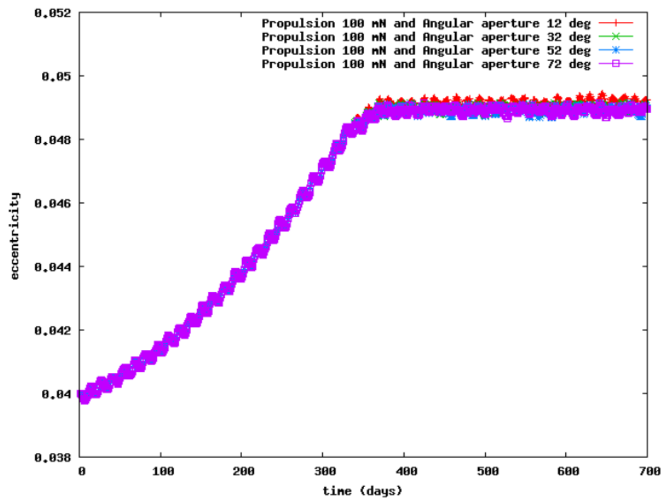

Figure 2 - Stabilization of eccentricity values with the correction providedby PMHT. Many sizes of propulsion arcs were considered.

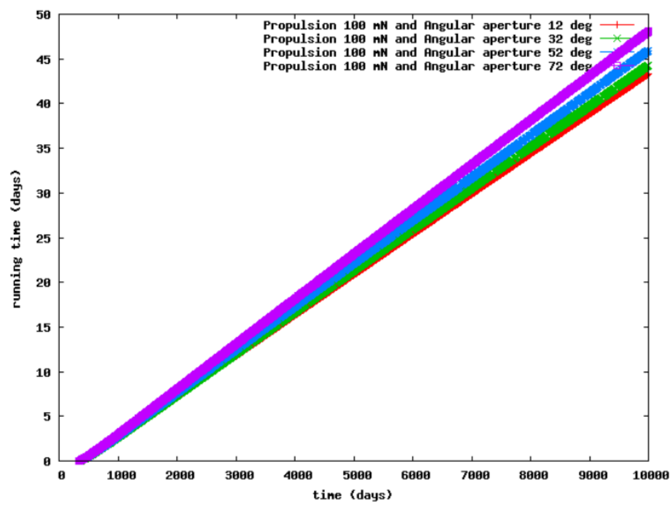

Figure 3 -The up time of thrusters as function of the total time used to the simulations

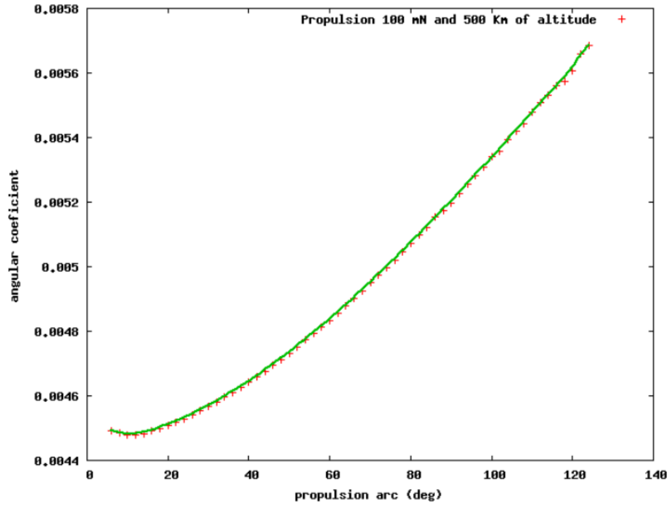

Figure 4 - Angular coeficient A(θ), resulted from linear fitting of data versus lenght of propulsion arcs.

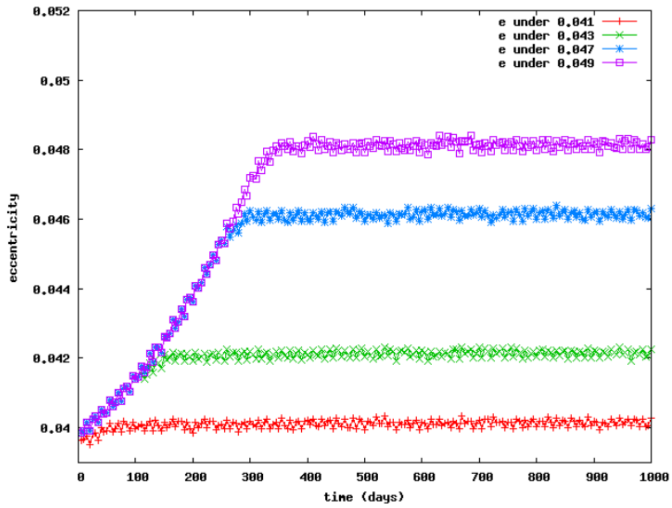

Figure - 5 Samples of eccentricties as functions of time.

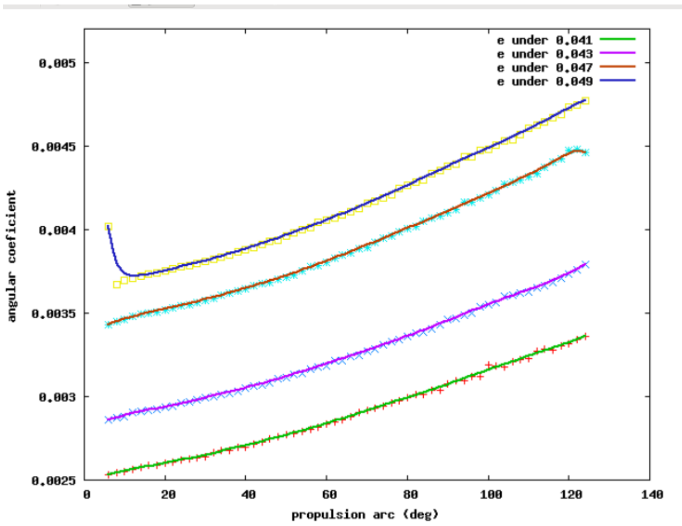

Figure 6 - Samples of angular coeficients as functions of propulsion arcs θ.

4 Conclusions

It is confirmed that the growth of eccentricity of a lunar polar satellite orbit can be controlled employing a PMHT in a pulsed manner.

There is the possibility to keep the satellite's eccentricity stable under 0.05, during a time t_k, with a running time of system given by:

$$R_t = A\left(t_k - 339.993\right) \tag{1}$$

The angular coeficient $A(\theta)$ presents a minimum value when the propulsion arc is around 12 degrees. Therefore we can assume there are an optimal lenght of these angular apertures in order to minimize the fuel consumption. The next step will be the seeking for empirical expressions of optimal θ as function of eccentricity maintained, altitude and mission time.

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