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6U CUBESATS AND ELECTROSPRAY PROPULSION SYSTEMS TO LUNAR ORBITS

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Abstract: This paper presents a feasibility study of electrospray propulsion for station keeping for a 6U CubeSat in lunar orbit. First the study of the long-term stability of lunar orbits is performed; then a simplified station keeping control is considered for two values of thrust and specific impulse that are feasible with constraints of power, mass and volume of an electrospray propulsion system of a 6U CubeSat. The manoeuvre performed allows to stabilise orbits at the Moon. The results enable to estimate the propellant mass consumption for station keeping at the Moon and demonstrate the potential of electrospray for CubeSat propulsion for mission beyond the Earth.

Keywords: Orbit perturbations, CubeSat, Moon, Electrospray, Orbit design

1. INTRODUCTION

Small satellites are an exciting and emerging field for performing useful missions in space. Their low development and launch cost allow new access to space which can be exploited by universities, government agencies and private companies. For one of the smallest satellite types, a CubeSat, this has resulted in satellites built to achieve mission goals without any form of active propulsion. A propulsion system is then desired to extend the range of applications of CubeSats. Current missions for these CubeSats have already been extended to the exploration of the Moon, with the necessity to couple them with a micro-propulsion system. A propulsion system is in fact needed to reach the Moon in case they are not injected on a Moon orbit by a carrier spacecraft and, possibly, to perform station keeping due to the high level of perturbation that characterises lunar orbits. Electrospray propulsion systems have the potential to provide small thrust values but at high specific impulse within the constraints of a CubeSat [1]; for this reason it was selected as main propulsion option for this work. At the same time the exploration of the Moon is of interest because the presence of water at the lunar poles and as it represents an optimal test-bed for CubeSats, to validate new technologies [2]. In addition, recently the European

Space Agency (ESA) has initiated LUnar Cubesats for Exploration (LUCE), to develop a number of CubeSat and more general nano-satellite missions and system concepts that can support ESA's lunar exploration objectives.

2. LUNAR ORBITS AND POLAR MAPS

2.1. The High-Fidelity Orbital Dynamics suite

The two-body equations of relative motion between two objects is based on the assumption that there are only two objects in space, and that they spherically symmetric gravitational fields are the only source of interactions between them. If this ideal case is considered, keplerian orbits are then the solution of the two-body equations. This is though an ideal scenario; for real space other forces can influence the dynamics of the two-body problem. Orbit perturbations should be included in the dynamics of the two-body if an accurate prediction of the satellite's dynamics is required. The High-Fidelity Orbital Dynamics (HiFiODyn) suite, developed at Politecnico di Milano, can predict the orbital evolution of a satellite, with high-fidelity dynamics. The HiFiODyn suite was developed by C. Colombo within the FP7 EU framework in the Marie Sklódowska-Curie Actions [3]. It was originally designed together with a semi-analytical propagator PlanODyn for the long-term propagation of highly elliptical orbits and the design of their disposals by enhancing the effect of natural perturbations [4]. HiFiODyn has been later extended to treat also medium Earth orbits, low Earth orbits [5], heliocentric orbits and Libration Point Orbits [6].

With this tool, maps representative of the behaviour of low lunar orbits (approximately lower than 125 km altitude) in polar region were created. These maps are useful to understand the strong instability of such orbits, looking especially at the maximum variation of orbital parameters over a reference period.

HiFiODyn, for a given initial orbit (a set of initial keplerian elements) and a given initial epoch, integrates the two-body equation of motion. Perturbations are considered in the dynamics of the satellite using to describe Gauss-planetary equations [7].

$$
\begin{aligned}
\frac{da}{dt} &= \frac{2a^2 v}{\mu} a_t \\
\frac{d\Omega}{dt} &= \frac{r\sin(\omega+\theta)}{h\sin(i)} a_h \\
\frac{di}{dt} &= \frac{r\cos(\omega+\theta)}{h} a_h \\
\frac{de}{dt} &= \frac{2}{v}(e+\cos(\theta)a_t) - \frac{r}{a}\sin(\theta)a_n \\
\frac{dM}{dt} &= n - \frac{b}{eav}(2(1+\frac{e^2 r}{p}\sin(\theta)a_t) + \frac{r}{a}\cos(\theta)a_n) \\
\frac{d\omega}{dt} &= \frac{1}{ev}(2\sin(\omega)a_t + (2e+\frac{r}{a}\cos(\theta))a_n) - \frac{r\sin(\omega+\theta)\cos(i)}{h\sin(i)} a_h
\end{aligned}
\tag{1}
$$

This set of equation is written for the t-n-h body-fixed (fixed with the satellite) reference frame where the t-axis is directed as the tangent to the motion of the satellite, the h-axis is in the direction of the angular momentum, and the n-axis is directed in the orbit plane, normal to the t-axis (inward); a_t, a_h, and a_n are respectively the perturbing acceleration acting on the t-axis, the h-axis, and the n-axis. In Eq. (1) a is the semi-major axis, Ω is the right ascnsion of the ascending node, i is the inclination, e is the eccentricity, M is the mean anomaly, and ω is the argument of perigee. For the discussion here presented, the main perturbations acting on a body that orbits around the Moon were considered; they are due due to the non-uniform gravity field of the Moon and the third body effect of both the Earth and the Sun [8, 9].

Having the orbit evolution of these six keplerian elements, maps representative of the stability of the orbit can be created. This will be done in the next section.

2.2. Polar obits around the Moon

For orbits in the vicinity of the Moon, with altitude lower than 750 km approximately, the lunar gravity field has the strongest contribution in the dynamics of the satellite [7]. So for altitudes lower than this value, a high order-high degree potential gravity model should be used to obtain a correct description of the motion; the lower the altitude, the more detailed the gravity model has to be if a reliable orbital evolution is required. For higher altitude, instead, the contribution in the dynamics of the satellite from the gravity model starts to be negligible with respect to the one due to the Earth's third body effect; the higher the altitude, the less accurate the gravity model can be. It is then helpful to understand if the electrospray propulsion system could work for station keeping around these low lunar polar orbits (the lunar injection is not treated in this work and it's here assumed that the spacecraft is injected by another satellite or directly by the launcher). To verify the employability of an electrospray thruster, it can be useful, first, to deduce what are the maximum variations in the orbit parameters for low lunar polar orbits that are representative of the stability of the orbit. In fact, low lunar orbits are in general extremely perturbed, with variations in altitude of several kilometers considered typical [8]. A high number of orbits were simulated over a period of approximately two months (more precisely 70 days) using HiFiODyn. The evolution of these orbits were then analysed. A time window of 70 days was chosen as suitable duration of a CubeSat mission around the Moon. A period of several months are considered typical for these missions [10–14].

Among all the possible initial orbits that could be chosen for this analysis, a frozen orbit was firstly studied. Indeed, selecting these orbits can reduce or even eliminate the need of station keeping due to their nature. Low lunar quasi-frozen orbits were considered. The assessment of maximum differences in keplerian elements for both frozen orbits and these quasi-frozen orbits were calculated to meet the mission requirements. In section 3 the action of the electrospray propulsion system will be taken into account on the evolution of such orbits. This is done to understand whether the electrospray propulsion system is capable of maintaining the same small difference in orbital elements observed for polar frozen orbits for the case of the quasi-frozen orbits that will be presented in this section.

A more detailed description of the process that has driven to the simulation of both frozen and non-frozen orbit is now given. Firstly, frozen orbits were considered for the study of the long term evolution of lunar orbits. These low lunar frozen orbits are governed by the strongly non-uniform mass distribution of the Moon. Therefore, if reliable information in the study of the existence of this kind of orbit is desired a high degree-high order gravity model must be used as a gravity model to calculate perturbations. [9, 15]. Though, this is not an easy task; in fact the resulting analytical equations often consist of thousands of terms [16]. For an analytical and simplified treatment of the problem, when using a reduced model of the gravity field, there has been found the existence of lunar frozen orbit (where here a frozen orbit is an orbit for which both rate of change in eccentricity e and argument of perigee ω equal to zero) only for ω equal to 90° or 270° [8, 9, 15]. This result simplifies the system of equation that should be solved in order to find keplerian parameter values representative of the lunar frozen orbit condition. This result allows us to reduce the problem to a one dimensional search for the value of e that, for ω equal to 90° or 270°, makes null the rate of cange of the argument of the periapsis [8, 9]. Inclination-eccentricity diagrams of frozen orbits can then be created, and they are present in literature with different level of accuracy (see Ref. [8, 9, 15]). In literature, often, the equations considered are averaged to predict a long-term behaviour of the frozen orbit [9, 15]. This allows to retain long-term information of the frozen orbit, that is in general needed for a preliminary design of a mission. What is here important is that these diagrams, for a given semi-major axis and for a value of argument of perigee of 90° or 270°, give the value of eccentricity and inclination that provide the lunar frozen orbit condition. Thanks to these diagrams frozen orbits were located. Three values of semi-major axis (that are required for the frozen orbit conditions) were considered and they are listed in table 1 [15]. For these three initial conditions eccentricity-inclination diagrams indicate that polar frozen orbits exist only for $\omega = 270°$ and eccentricity that span from 0 to 0.06 as maximum. This region was then simulated for a range of inclinations that belongs to $85° < i < 95°$. The reader should keep in mind that all frozen orbits location are obtained with a reduced gravity model and the real dynamics of the problem disrupt the frozen orbit condition [8], and because of that, from now on, the frozen orbit condition will be here referred as quasi-frozen.

Considering that the Moon's equatorial radius is approximately to 1738 km, the three values of semi-major axis here considered are quite low. In fact, if a circular orbit is considered with these 3 values of semi-major axis, altitudes between 75 km and 125 km will be observed if no perturbations are considered. They are then relatively close to the Moon's surface and of interest because of this characteristic.

So far, only the choice of semi-major axis and the argument of perigee have been discussed. The other orbital parameters were taken equal to the $\Omega = 180°$, $\theta = 180°$ for a period of 70 days, starting from 01/01/2018 at 12:00 UTC Gregorian time.

Tab. 1 Set of initial conditions maintained as fixed in the assessment of typical variation of low quasi-circular lunar polar orbits.

Set name	a [km]	ω [°]
LLO1	1863	270
LLO2	1838	270
LLO3	1813	270

Accordingly to the 3 limited set of keplerian elements of table 1 and eccentrcity-inclination diagrams of Ref. [9], quasi-frozen orbits exist for quasi-polar region ($85° < i < 90°$) and eccentricity values that span between approximately 0 and 0.05 and only if ω is equal to 270° (quasi-frozen orbits exist also for ω equal to 90° but for a different range of inclinations, less than 85°).

However, the range of inclination considered in the simulation, this range was extended up to the value of 95°. This is because for a simplified problem as the one considered to find frozen orbit conditions a symmetry exists between the case of direct inclination orbits and retrograde ones.

It is useful, then, to find out if this symmetry exists also for the non-averaged problem in this range of inclinations (i.e. $90°< i < 95°$). A simulation of orbits in the region considered was then performed; in this way both frozen and non-frozen orbits were studied. Specifically, the simulation was performed with a 100-degree 100-order gravity LP165P model and both the Earth and the Sun as third body effects for a period of 70 days.

Maps representative of the evolution of the keplerian elements for the polar region considered were created. Eccentricity and inclinations were varied. More precisely, for these 3 prescribed set of initial conditions, the inclinations was varied between 85° and 95° with step of 1°. The eccentricity instead was varied from the value of 0.01 to the value of 0.045 with step 0.005 for the first two set of initial parameter in table 1.

From now on, the first set of conditions in table 1 is referred as Low Lunar Orbit 1 (LLO1), the second set Low Lunar Orbit 2 (LLO2) and the third one as Low Lunar Orbit 3 (LLO3). Results in maximum difference of the orbital parameters for LLO1, LLO2, and LLO3 in the region of inclination-eccentricity considered are now presented.

The maps show iso-lines representative of the maximum variation of the orbital parameter, over the entire simulation period considered.

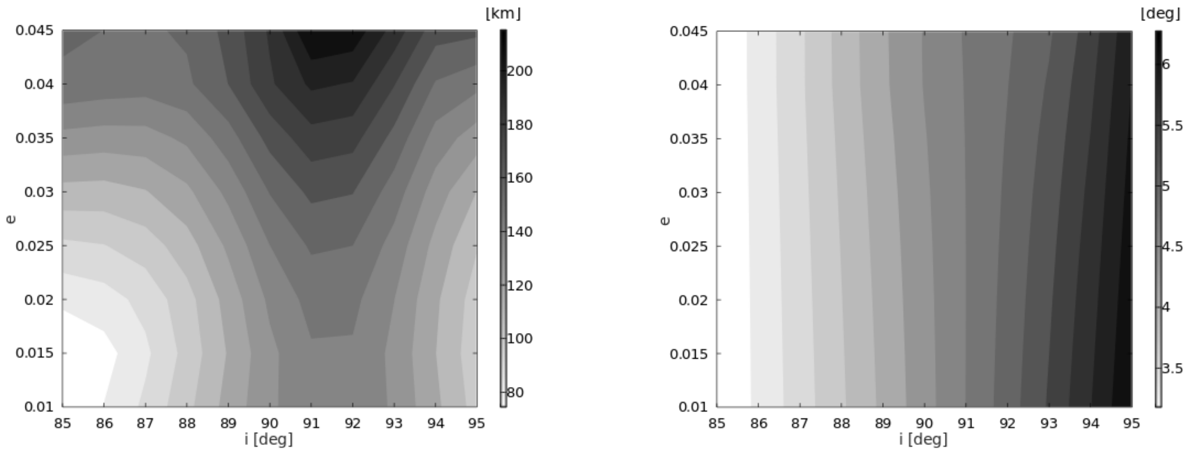

Fig. 1 LLO1: maps representative of the maximum variation over 70 days of the altitude (to the left) and of the inclination (to the right).

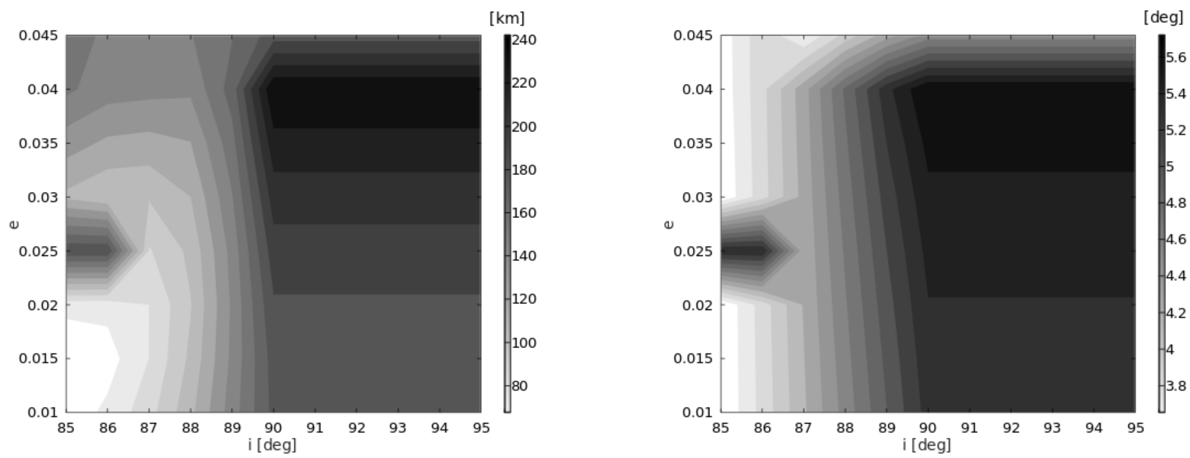

Fig. 2 LLO2: maps representative of the maximum variation over 70 days of the altitude (to the left) and of the inclination (to the right).

As it can be seen from the figures representative of the maximum difference in the altitude, a change of several km over approximately two months can be considered nominal for low lunar orbit if no station keeping control is applied. The same consideration can be extended to all the other parameters; great difference with respect to the initial parameter are always observed. This confirms that low lunar orbits are extremely perturbed and mission designs must takes this into account.

For LLO2 and LLO3, some orbits that were simulated resulted in impact conditions (i.e. altitude equals to zero km as Moon's mountains were not considered). All these impacts are observed after 55 days and orbits that have driven to these impact conditions are listed in table 2 and 3. From these tables it can be noticed that from LLO2 to LL03 the number of impact increases drastically (from 9 of LLO2 to 52 of LLO3), especially for inclinations between 89° and 93°. It can also be noticed that no symmetry exist in impact condition, with the prograde orbits seeming to be better candidates to avoid impact condition.

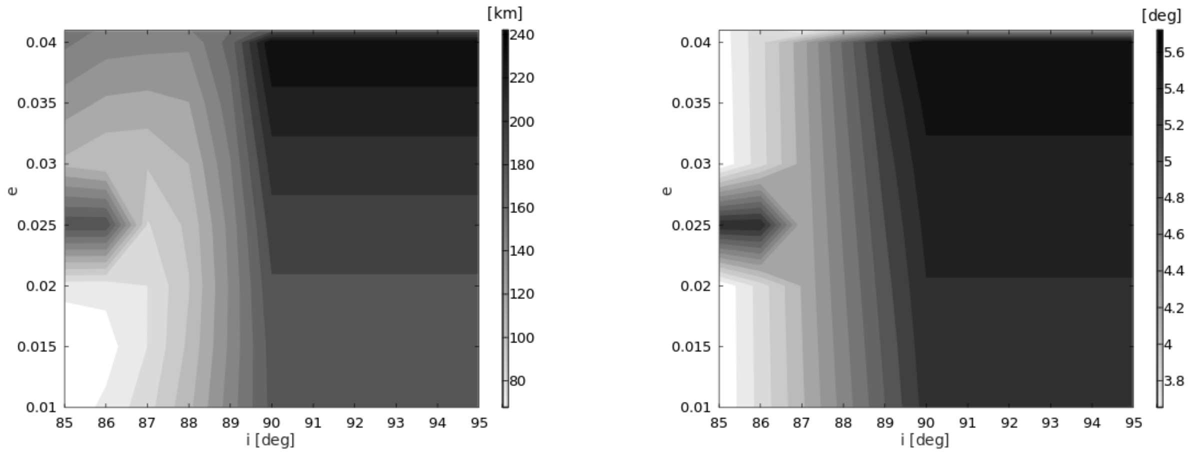

Fig. 3 LLO3: maps representative of the maximum variation over 70 days of the altitude (to the left) and of the inclination (to the right).

Tab. 2 Impact and non-impact conditions for LL02. The symbol "●" indicates that no impacts occur; the symbol "x" indicates that impact occur.

e/*i*	85°	86°	87°	88°	89°	90°	91°	92°	93°	94°	95°
0.010	●	●	●	●	●	●	●	●	●	●	●
0.015	●	●	●	●	●	●	●	●	●	●	●
0.020	●	●	●	●	●	●	●	●	●	●	●
0.025	●	●	●	●	●	●	●	●	●	●	●
0.030	●	●	●	●	●	●	●	●	●	●	●
0.035	●	●	●	●	●	●	x	x	●	●	●
0.040	●	●	●	●	●	x	x	x	●	●	●
0.045	●	●	●	●	●	x	x	x	●	●	●

Tab. 3 Impact and non-impact conditions for LL03. The symbol "●" indicates that no impacts occur; the symbol "x" indicates that impact occur.

e/*i*	85°	86°	87°	88°	89°	90°	91°	92°	93°	94°	95°
0.010	●	●	●	●	x	x	x	x	x	●	●
0.015	●	●	●	●	x	x	x	x	x	●	●
0.020	●	●	●	●	x	x	x	x	x	●	●
0.025	●	●	●	●	x	x	x	x	x	●	●
0.030	●	●	●	●	x	x	x	x	x	●	●
0.035	●	●	●	●	x	x	x	x	x	●	●
0.040	x	x	x	x	x	x	x	x	x	x	x
0.045	x	x	x	x	x	x	x	x	x	x	x

3. CONSTRAINTS FOR A 6U CUBESAT AND STATION KEEPING

3.1. Constraints of a 6U CubeSat and station keeping

In order to perform station keeping, an electrospray thruster model was created relying on experimental results available in literature. This thruster model was used to calculated thrust, specific impulse, and power consumption of an electrospray propulsion system that, depending on the applied voltage, can work or in the so-called mixed mode or in the so-called pure ionic regime. In fact, depending on the device considered, and also on the propellant, the ejected beam can be composed by both ions and droplets, representative of the mixed mode, or even by only ions in the case of a pure ionic regime. Thrust and specific impulse are greatly influenced by the relative composition of ions and droplets of the beam, allowing great flexibility in thrust performances. A detailed discussion of the thruster model can be found in Ref. [17]. What is here important is that the thruster model created can be used to assess thrust, specific impulse, and power consumption of the propulsion system, that are needed to assess the feasibility of this type of propulsion in a CubeSat.

No CubeSats have yet flown to the Moon; for this reasonthe Lunar Water Distribution (LWaDi) [18] mission was taken as reference to guess constraints of the propulsion system for a 6U CubeSat. This mission gives important information related with subsystem allocations of these 6U CubeSats to the Moon. Accordingly to this mission, as constraints of the propulsion system, it was assumed a maximum power of 40 W, a maximum wet mass of 1.5 kg and a maximum volume of 1.5 U. After having verified that the value of 0.3 mN, that is a thrust reference value for electrospray propulsion technology, is feasible with constraints of the propulsion system assumed so far, a higher value of 1 mN of thrust was considered and identified as maximum value of thrust feasible within a 6U CubeSat. These two values of the specific impulse were used for the station keeping manoeuvre, 1000 second representative of the mixed mode and 4000 second of the pure ionic regime.

For 0.3 mN of thrust the maximum power needed to obtain this thrust is of approximately 11 W and a higher thrust value is then feasible with constraint in power assumed so far. It was calculated that 1 mN of thrust is the maximum value that can withstand constraints in power, mass and volume of the propulsion system asssumed so far. So 0.3 mN and 1 mN will be used as reference values of the thrust fot the eletrospray propulsion system for a 6U CubeSat. The same maps presented in the previous section were created with the same initial parameters that characterises LLO1 of previous section but considering this time also the control acceleration due to the propulsion system itself. It will be demonstrated that, also with the small values of thrust used in the simulation (in comparison to typical values of perturbations around the Moon), the electrospray propulsion system makes a significant difference in reducing the maximum variation of the keplerian elements for the orbits considered.

3.1. Station keeping for low-lunar quasi circular orbit

It has already been explained that low lunar quasi-circular orbits are of interest and, at the same time, extremely perturbed. Therefore, some station keeping is required for these orbit if the orbit is to remain consistent.

Thrust and specific impulse of the electrospray propulsion system discussed in the previous paragpRah were used to asses the capability of such systems to change the orbital parameters for the low lunar quasi-circular orbits, even if low-thrust values are considered. A low-thrust manouver law is then needed to explore this scenario. The design of optimal transfer laws is a challenging task; nevertheless some simple optimal laws are presented in literature (see for example Ref. [19]). When low-thrust values are used to perform the manouver, the action of the thruster to the satellite's dynamics can be considered as a perturbation acting on the satellite. The action of the thruster can be then considered in the dynamics of the satellite as another perturbation in the Gauss-planetary equation, as was done for the others perturbations here considered. The acceleration exerted by the thruster was then considered in the satellite's dynamics. The LLO1 was used to compute the same maps presented in the previous chapter, but this time considering in the satellite's dynamics also the perturbation due to the action of the electropsray propulsion system. In this way the capability of the thruster to counteract perturbations for the orbits considered can be assessed. The design of the manouver used to answer this last point is now discussed.

The optimal law of table 1 in Ref. [19] was chosen to perform a manoeuvre that maximize the rate of change of the inclination. The optimal inclination manouver that was chose it's reported in Eq. 2 for convinience. This particular manouver was chosen because it is representative of one of the more expensive manouver for the propellant consumption point of view.

$$\begin{aligned} p_t &= 0 \\ p_n &= 0 \\ p_h &= \boldsymbol{p}\sin\left(sgn\left(\cos\left(\omega+\theta\right)\right)\frac{\pi}{2}\right) \end{aligned} \qquad (2)$$

In Eq. 2, $\boldsymbol{p}$ is the acceleration due to the thrust action and is considered variable in time with an initial mass of 12 kg. It is assumed that the propulsion system used to perform the manouver it is always able to thrust along the optimal direction, that changes in time. The case simulated is then an ideal case as the electropray propulsion system it is unlikely to be able to performe any instantaneous thrust vectoring control. Nevertheless, useful information related with mass consumption of the manouver and the ability of the electropsray propulsion system to deal with perturbations of the low lunar orbits will be obtained from this ideal case.

The mass is one of the main constraint of a satellite, and understand if the propellant mass consumption due to the particular manoeuvre performed is feasible with constraints of the propulsion system here assumed is essential. The new simulation consider the same polar region of LLO1 in the previous chapter.

Orbit LLO1 was chosen because no impact conditions were observed when the satellite's dynamics is free, i.e. no manouver are performed. Because the manoeuvre here considered it's related with the change of inclination and not with the change in altitude, LLO2 and LLO3 should not be chosen otherwise impacts might be present again and no useful informations of the ability of the electrospray propulsion system to change the inclination would be retrieved from the resulting impact conditions. In this analysis all the 88 cases considered for the LLO1 in the previous chapter were simulated, but with the difference that this time, in the HiFiODyn suite, also the perturbation of the thrust action able to perform an optimal inclination manouver is considered. For all the cases simulated in this new scenario the optimal manouver was performed with the goal of maintain the initial inclination fixed. Both values of thrust of 0.3 mN and 1 mN were used.

Results of this new simulation for both the case with a thrust value of 0.3 mN and the one with a thrust value of 1 mN are now shown in Fig 4. and Fig 5.

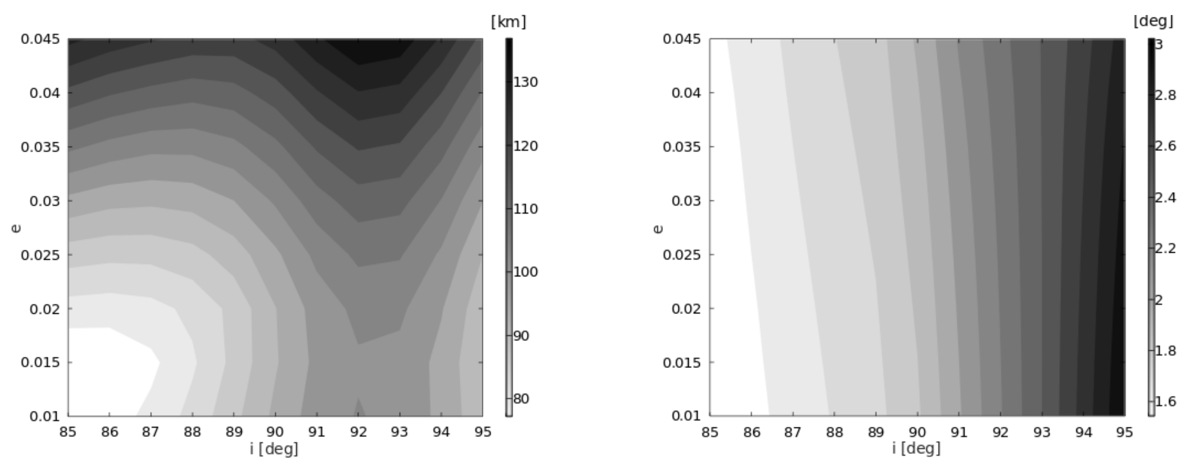

Fig. 4 LLO1: maps representative of the maximum variation over 70 days of the of the altitude (to the left) and of the inclination (to the right) for the thrust value of 0.3 mN.

From these figures it can be noticed that, even in the ideal case of an optimal manoeuvre, the propulsion it's not able to maintain the initial inclination as fixed, but it's able to reduce significantly the maximum variation in the inclination.

For the four manoeuvres considered, the propellant mass consumption was calculated. When the thrust of 0.3 mN is considered, the propellant mas consumption is of 184.954 g with a specific impulse of 1000 second and of 46.509 g when the specific impulse considered is of 4000 s. For the case of 1 mN of thrust, instead, propellant mass consumption is of 616.514 g when the value of 1000 s is considered as specific impulse while it is of 157.19 g when the specific impulse considered is of 4000 second. With a constraint in mass of the propulsion system of 1.5 kg assumed so far, the propellant mass consumption for the 4 manoeuvres considered is retained as feasible for an electrospray propulsion system.

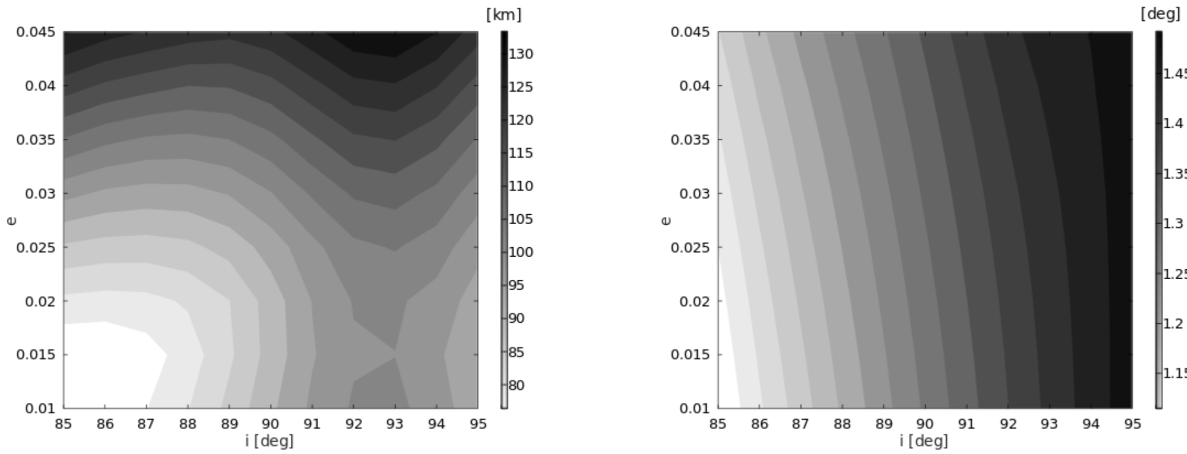

Fig. 5 LLO1: maps representative of the maximum variation over 70 days of the of the altitude (to the left) and of the inclination (to the right) for the thrust value of 1 mN.

4. CONCLUSIONS

Low lunar orbits are of interest because of the vicinity with the Moon's surface that could allow a more accurate exploration of the Moon. Therefore, in section 2 low lunar quasi-circular polar orbits were studied by means of long-term evolution maps. Without a careful choice of the initial orbit or without a propulsion system able to maintain the initial orbit in time, the satellite might impact the Moon in few months. The stability of LLO was studied by representing maps showing the total variation of orbital elements over 70 day and varying eccentricity and inclination value for a limited set of orbital parameters.

In section 3, constraints in mass, volume and power for a 6U CubeSat to the Moon were used to define two values of thrust and specific impulse needed to performed an optimal inclination manoeuvre. Comparing the case of LLO1 with and without the action of the electrospray propulsion system it was possible to understand the ability of this propulsion system in counteract Moon's perturbation. A pre-defined optimal inclination manoeuvre performed is able to reduce the maximum variation of the inclination of a significant amount. More specifically, with the optimal manoeuvre performed, initial inclination values can be maintained within a maximum of variation of 3° over 70 days when a 0.3 mN value is used an within approximately 1.5° with a 1 mN thrust value. When the inclination maneouvre is performed a reduction in the maximum variation of the altitude, that is an important parameter for the stability of the orbit, can be observed. In fact, the lower the altitude the stronger the perturbation of the Moon in the orbit. This fact suggests that the inclination is a key parameter in the control of low lunar quasi-circular orbit, with prograde orbtis that seem to be more stable than retrograde ones.

The feasibility study is then concluded with the propellant mass that is needed to perform the manoeuvres considered. Even the worst scenario is considered feasible within constraints of an electrospray propulsion system in a 6U Cubesat to the Moon. An electrospray propulsion system is a good option for allowing the extension of mission duration of a Cubesat within low lunar orbit.

5. ACKNOWLEDGMENTS

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