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MULTI-TETHERED MANEUVERS TO CHANGE THE INCLINATION OF THE ORBIT OF A SPACECRAFT

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The study of applications of tethers in space is growing in the last years. One possibility is the use of tethers to help to maneuver a spacecraft. The main idea is to replace maneuvers based in fuel consumption delivered by propulsion systems by maneuvers using tethers. Several types of tethers have been considered with this goal, such as electrodynamics tethers, to increase or decrease the orbit of a spacecraft travelling around the Earth; momentum exchange tethers, which usually consists in a tether linking two or more satellites that is romped at a certain point; tethered sling shot maneuver, which uses a celestial body to fix a tether that rotates the spacecraft to give or remove energy from the spacecraft, similar to a gravity assisted maneuver. The present paper makes an initial study to extend the concept of the tethered sling shot maneuver to the use of more than one tether attached to the moon to change the inclination of a spacecraft passing near the moon. The advantage is to reduce the size of the tethers and also to increase the total effects of the maneuver by splitting the maneuver in several tethers. It means that an initial study is made in the idea of using multiple tethers fixed in one body to perform orbital maneuvers, verifying the effects of the maneuver as well as the tension in the cables..

Keywords Astrodynamics; Orbital Maneuvers; Tethers; Tethered Slingshot Maneuvers.

I. INTRODUCTION

In general, the idea of using tethers in space started in 1895, when Konstantin Tsiolkovsky considered the idea of an orbital tower, inspired by the Eiffel Tower (Yi Chen et alli, 2013, Burov et alli 2014a, 2014b, 2015).

Developments in technologies can make possible more applications related to the use of tethers in space activities in several fields. New applications for tethers include "Tethered Space Robot", "Tethered Space Net" and "Tethered Spacecraft Formation." There are also proposals for orbital maintenance, such as supply service, orbital maneuvers, and capture/removal of space debris.

Williams et al. (2004) proposed a planetary capture maneuver. The spacecraft is considered to be in an hyperbolic orbit and it is captured by the tether. After a predefined rotation the spacecraft is released into an elliptical orbit around the target planet. In this study, the mass of the tether is optimized to ensure that the cable does not break during the maneuver.

In general, a "Tethered Sling Shot Maneuvers" (TSSM) is a maneuver where a cable is anchored on a celestial body (planet, moon or asteroid) to attach a spacecraft during a certain time. This configuration makes the spacecraft to rotate around the celestial body for a given angle, so changing its trajectory, in a form similar to a "Gravity Assisted Maneuver". The main advantage of the maneuver based on tethers is that the gains of energy are much larger when compared to the equivalent "Gravity Assisted Maneuvers". More details about the pure gravity close approach can be found in references like Minovitch (1961), Hollister et alli (1966), Flandro (1966), D'Amario et alli (1981, 1982), Byrnes and D'Amario (1982), Broucke (1988), Striepe and Braun (1991), Dowling et alli (1990, 1991), Sukhanov (1999), Longuski and Strange (2002), McConaghy et alli (2003), Prado (2007), Ross and Scheeres (2007), Gomes and Prado (2008, 2010), Ferraz et alli (2013, 2015).

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This is particularly true when smaller bodies, like an asteroid or a small moon of a planet, is considered. Those small bodies have very weak gravity fields, which produce small rotation angles for a gravity assisted maneuver, which results in small variations of energy. In that sense, the use of tethers can give larger rotation angles, providing stronger variations of energy to the spacecraft. Those large variations can result in captures and escapes of the spacecraft with respect to the main body of the system, as shown in Prado (2015). The moon is not so small, but it is not too large to give strong effects in a gravity assisted maneuver.

There are also other ideas related to this maneuver in the literature. Lanoix and Misra (2000) studied problems related to the construction of an anchor device that can be used to fix a tether in an asteroid, taking into account the mass of the tether and the tension on the cable. After the fixation of the tether, the velocity vector of the spacecraft relative to the asteroid is rotated, thus modifying its heliocentric velocity. The link to the anchoring device can then be removed, and the spacecraft leave the system. Penzo and Mayer (1986) also used this idea in asteroids, studying the same type of problem. Lanoix (2000) considered similar problems and Puig-Suari et al. (1995) studied the use of tethers to explore the moon and the planets of the Solar System with the use of a solar-powered electric engine to give velocity to the tether. Thomson and Stern (1995) considered the materials used to build the cable. Later, Prado (2015) used small moons of the larger planets to make captures around the planets using tethers fixed in their moons. Prado et al. (2018) proposed a device to be fixed in an asteroid to give energy to a spacecraft making a passage near the asteroid. Oliveira and Prado (2017) evaluated the use of tethers with uniform cross-sectional area and subjected to a uniform mechanical stress in orbital inclination change maneuvers. Nascimento et alli (2017) proposed the use of a tether fixed in the moon to change the orbit of a spacecraft.

The objective of the present work is to study the feasibility of orbital maneuvers using multiple tethers fixed in the moon to help to reduce the cost of a mission that needs an inclination change, as an extension of Nascimento et alli (2017). A system of tethers like that can be fixed in the surface of the Moon to perform an Orbital Plane Change Maneuver (OPCM) of an approaching spacecraft. The system is composed by several cables (tethers) that are anchored in the surface of the Moon, each of them providing a turning angle for the spacecraft. The main advantage of using several tethers is that it is possible to reduce the size of each tether and still getting larger variations of inclinations. This concept increases the difficult, in terms of requiring more tethers, but reduces the problem of using large tethers. Independent of these practical problems, the focus of the present paper is to show a possibility given by the astrodynamics, not going in the details of the practical construction of the device.

This type of maneuver is particularly interesting for a spacecraft leaving the Earth for an interplanetary trajectory in a plane different from the one that it is launched. In this case, it is considered a cable that is fixed in the surface of the Moon, which is used to rotate the velocity vector of a spacecraft that is passing nearby. This rotation changes the energy, velocity and angular momentum of the spacecraft with respect to the Earth. The new spacecraft trajectory will have different values for the Keplerian elements with respect to the Earth, including a variation in inclination, which is a costly maneuver if made based in propulsion systems. Several geometries will be considered, including planar maneuvers, which maximize energy variation and three-dimensional maneuvers, which is able to change the inclination of the trajectory.

The present study includes the consideration of the size of the cable, as well as its tension, for each maneuver. These points are very important to give an idea of the requirements that will be necessary to implement such a device. It is a generic and preliminary study of the problem. Simplifying hypothesis are made and a detailed studies need to be made to verify in more details this type of maneuver. In that sense, a first analysis can be made using the “patched-conics model”, which divides the motion of a spacecraft in a series of two-body problems.

ANALYTICAL EQUATIONS FOR THIS MANEUVER

The first step is to recall the pure gravity Swing-by maneuver. This type of maneuver is used when standard maneuvers based in fuel consumption, impulsive (Lang, 1979; Gaias and D‘Amico, 2015) or using low thrust (Sukhanov and Prado, 2001; Cao et alli, 2014)), do not give the energy required for the spacecraft in a given mission. The main idea is to use the rotation made by the gravity field in a trajectory of a spacecraft to give or remove energy from a spacecraft that passes near the celestial body. It is a usual maneuver in astrodynamics and very efficient in missions going to the outer planets of the Solar System. Figure 1 describes the maneuver and shows some of the most important variables involved.

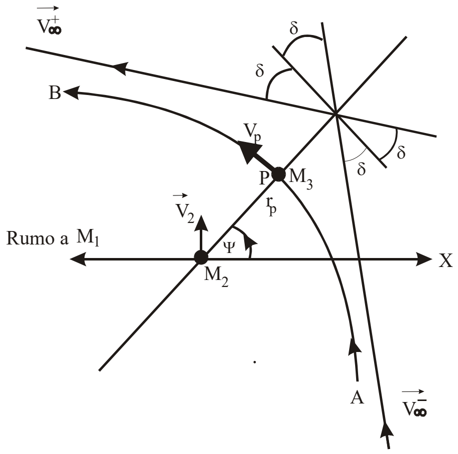

Fig 1 - The swing-by maneuver and some of its variables. [Prado (2001)]

The variables are:

$\vec{V}_2$ = the velocity of M_2 with respect to M_1;

$\vec{V}_{\infty}^{-}, \vec{V}_{\infty}^{+}$ = velocity vectors of the spacecraft with respect to M_2, before and after the encounter, respectively;

$\vec{V}_i, \vec{V}_o$ = velocity vectors of the spacecraft with respect to M_1, before and after the encounter, in an inertial frame, respectively;

δ = half of the angle of curvature (the angle between $\vec{V}_{\infty}^{-}$ and $\vec{V}_{\infty}^{+}$);

r_p = the minimum approach distance during the encounter (point P) between M_2 and M_3;

Ψ = the angle between the periapsis line (line connecting M_2 to P) and the line M_1 - M_2.

For this study, during the passage of the spacecraft near the Moon, it is assumed that the motion is governed by the classical problem of two bodies. The maneuver is defined by the plane formed by the vectors $\vec{r}_p$ and $\vec{V}_p$. Then, the vectors $\vec{V}_{\infty}^{-}$ and $\vec{V}_{\infty}^{+}$, are the vectors before and after the conventional swing-by, respectively, relative to M_2, which can be written as a linear combination of the versors associated with $\vec{r}_p$ and $\vec{V}_p$. Using $\vec{V}_{\infty}$ to represent both $\vec{V}_{\infty}^{-}$ and $\vec{V}_{\infty}^{+}$, provided that the conditions are the same for both, a double solution will provide the values for $\vec{V}_{\infty}^{-}$ and $\vec{V}_{\infty}^{+}$, that is:

$$\vec{V}_{\infty} = A\frac{\vec{r}_p}{r_p} + B\frac{\vec{V}_p}{V_p} \tag{1}$$

2. ORBITAL MANEUVERS USING "TETHERS"

The are several papers in the literature considering tethers in space, like Crouch et alli (1984), Lorenzini (1987), Cartmell and McKenzie (2008), Shi et alli (2016), Yang et alli (2016), Sun et alli (2017), Ferraz et alli (2017).

For the present study, it is supposed that a spacecraft is launched from Earth, approaches the Moon, reaches the tether, and then rotates by an angle 2δ, according to the geometry presented in Fig. 2. This is the situation where there is only one tether in the maneuver.

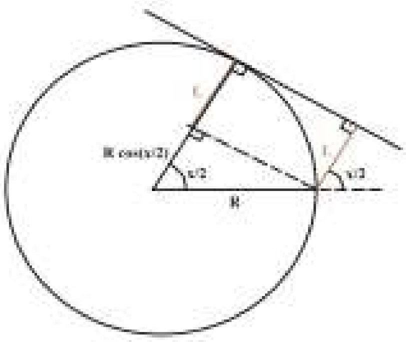

Fig. 2 - Geometry of the tether system and around the Moon. [Nascimento et alli (2017)]

A first step is to obtain an equation for the rotation angle of the spacecraft is now developed. It is assumed that the limit of the rotation is reached when the spacecraft passes tangent to the moon after it is released from the tether. Equation (2) shows the result.

$$\delta = 2\arccos\,[(R-L)/R] \tag{2}$$

where R is the radius of the Moon, L is the length of the tether and "2δ" is the rotation angle of the spacecraft.

Note that the gravitational interactions during the time that the tether is acting is neglected in the present model. It means that the rotation has a constant velocity, which is the velocity of approach of the spacecraft. It is designated by V_{inf}. The orbital velocity (V_2) of the Moon around the Earth is assumed to be V_2 = 1.022 km/s. The amount of energy δE gained by the tethered maneuver is given by Prado (2015).

$$\delta E = -2 * V_2 * V_{inf} * \sin(\delta) * \sin(\psi), \tag{3}$$

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The velocity variation δV is given by Prado (2000) and Prado and Felipe (2007).

$$\delta V = 2 * V_{inf} * \sin[\delta] \tag{4}$$

For the numerical simulations made in the present paper, the approach angle ψ used is given by $\psi = 3*\pi/_2$, because it is the geometry of maximum gains of energy. To compare the gains of energy obtained from a gravity assist with the moon and the one obtained from a single tethered maneuver, it is necessary to obtain the maximum angle of rotation that can be obtained by the gravity maneuver around the Moon. It is given by Eq. (5).

$$max_{angle} = \arcsin[1/(1+R*V_{inf}*V_{inf}/\mu_{moon})], \tag{5}$$

where μ_{moon} = 4904 km^3/s^2. The maximum energy variation obtained by the Moon's gravity is then given by Eq. (7).

$$\delta E_{maxlu} = -2*V_2*V_{inf}*\sin(max_{angle})*\sin(\psi), \tag{6}$$

The maximum velocity variation obtained by the Moon's gravity is given by

$$\delta V_{maxlu} = 2*V_2*\sin(max_{angle}) \tag{7}$$

The angle Ψ is defined as the angle formed between the periapsis line of the spacecraft relative to the Moon and the imaginary Earth-Moon line.

Figure 1 shows the concept of swing-by. The present study considers M_1 as being the Earth, M_2 the Moon and M_3 a spacecraft in an orbit coming from the Earth and approaching the Moon. Prado (2000) studied the change of the inclination of the spacecraft, which is given by equations (8) to (13) (Prado, 2000).

$$\left|\vec{C}_i\right| = dV_\infty\left(\left(\sin\beta\sin\delta + \cos\beta\cos\delta\sin\gamma\right)^2 + \left(\frac{V_2}{V_\infty} + \cos\alpha\cos\delta\cos\gamma + \cos\beta\sin\alpha\sin\delta - \cos\delta\sin\alpha\sin\beta\sin\gamma\right)^2\right)^{1/2} \tag{8}$$

$$C_{iZ} = dV_\infty\left(\frac{V_2}{V_\infty} + \cos\alpha\cos\delta\cos\gamma + \cos\beta\sin\alpha\sin\delta - \cos\delta\sin\alpha\sin\beta\sin\gamma\right) \tag{9}$$

So,

$$\mathrm{Cos}(i_i) = \frac{C_{iZ}}{\left|\vec{C}_i\right|} = \frac{1}{\sqrt{1 + \left(\dfrac{\sin\beta\sin\delta + \cos\beta\cos\delta\sin\gamma}{\dfrac{V_2}{V_\infty} + \cos\alpha\cos\delta\cos\gamma + \cos\beta\sin\alpha\sin\delta - \cos\delta\sin\alpha\sin\beta\sin\gamma}\right)^2}} \tag{10}$$

$$\left|\vec{C}_o\right| = dV_\infty\left(\left(\sin\beta\sin\delta - \cos\beta\cos\delta\sin\gamma\right)^2 + \left(\frac{V_2}{V_\infty} + \cos\alpha\cos\delta\cos\gamma - \cos\beta\sin\alpha\sin\delta - \cos\delta\sin\alpha\sin\beta\sin\gamma\right)^2\right)^{1/2} \tag{11}$$

$$C_{oZ} = dV_\infty\left(\frac{V_2}{V_\infty} + \cos\alpha\cos\delta\cos\gamma - \cos\beta\sin\alpha\sin\delta - \cos\delta\sin\alpha\sin\beta\sin\gamma\right) \tag{12}$$

What results in

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$$\mathrm{Cos}(i_o) = \frac{C_{oZ}}{|\vec{C}_o|} = \frac{1}{\sqrt{1+\left(\dfrac{\sin\beta\sin\delta-\cos\beta\cos\delta\sin\gamma}{\dfrac{V_2}{V_\infty}+\cos\alpha\cos\delta\cos\gamma-\cos\beta\sin\alpha\sin\delta-\cos\delta\sin\alpha\sin\beta\sin\gamma}\right)^2}} \qquad 13)$$

Where $\vec{C}_i$ and $\vec{C}_o$ represent the initial and final angular momentum of the spacecraft, respectively, and ii and i represent the initial and final inclination of the spacecraft.

Figure 3 shows the "Capture Portal", detailed in Prado et alli (2018). The idea of the present paper is similar to that one, just using a series of tethers to divide the maneuver in several steps.

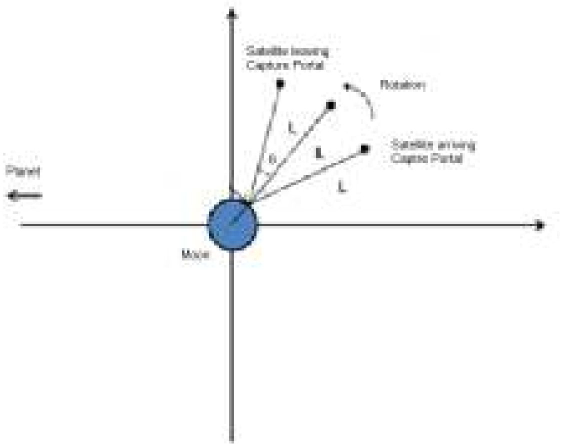

Fig.3 – Capture Portal. [Prado et alli (2018)]

The main results are presented in the next figures, showing the effects of the maneuvers in the trajectory of the spacecraft.

The maximum energy variation given by the gravitational interaction with the Moon was studied by Nascimento et alli (2017). There is a maximum with respect to the velocity of approach, near 2 km/s. The energy variation given by the tether is shown in Fig. 4, as a function of the approach velocity and the parameter L, the length of the tether. Comparing the results from that study and this figure, we can observe that using only the gravitational assistance, the maximum energy gained by the spacecraft is about 1.7 km^2/s^2, while using the tether this amount can be much greater, up to 10 km^2/s^2, as shown in Fig. 4. Note that the maximum value reached is about 1.7 km^2/s^2 in that study, while using the tether this same energy gained increases with no limitation as the approach velocity increases. This is very important, because the velocity of approach depends of several constraints of the mission, and is not a free parameter. The variation of the cable length is also a determinant factor in the amount of energy gained. The large the length of the tether, the larger is the amount of energy gained. The reason is that larger tethers can generate larger rotation angles for the spacecraft.

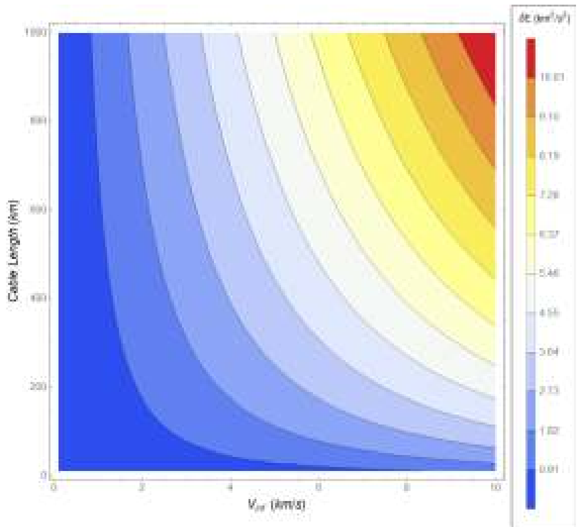

Fig.4 – The energy variation given by the tether.

The benefits, in terms of energy gains, provided by the tether-assisted maneuver are easily noticed. Large regions of the plots transformed closed elliptical orbits into open hyperbolic orbits, due to the extra energy obtained from the maneuver.

Note that the approach velocity is usually a condition given by the mission. In the case of the use of tethers, the inclination of the orbit of the spacecraft may be associated not only with the velocity of approach, but also with the length of the tether.

The variations of velocity of the tethered and the gravitational maneuvers was studied by Nascimento et alli (2017), that showed that the variation of velocity increases with the approach velocity and also with the length of the tether. It decreases with the approach velocity, which means that the effects is opposite to the ones coming from the tethered maneuver. That study showed that the amount of δV can be up to 10 km/s while in the case of only gravity maneuver with the Moon, this amount is limited to about 2 km/s.

After that, the variations in the inclination of the spacecraft are obtained. The variation of the inclination of the orbital plane is shown in Fig. 5, as a function of the cable length and the approach velocity. This inclination can reach values higher than 46 degrees using the tether, while in the case of the gravity maneuver the variation of the orbit inclination is limited to about 30 degrees in the case where the approach velocity is about 1.7 km/s, which is a very particular value that does not occur in all missions. This variation is also obtained as a function of the approach velocity, in the case of the pure gravity maneuver. Since the approach velocity is generally a condition given by the mission, in the case of the gravity maneuver this inclination is limited by this condition. In the case of the tethered maneuver, the inclination of the orbit of the spacecraft can also be associated with the cable length. Using one more parameter (the length of the tether) gives much more options to reach the desired objective. Note that the maximum value of the difference of the orbit inclination is reached in a specific region for the given range of parameters, where this maximum is reached when the cable length is within 800 and 1000 km and the approach velocity is within 4 and 5 km/s.

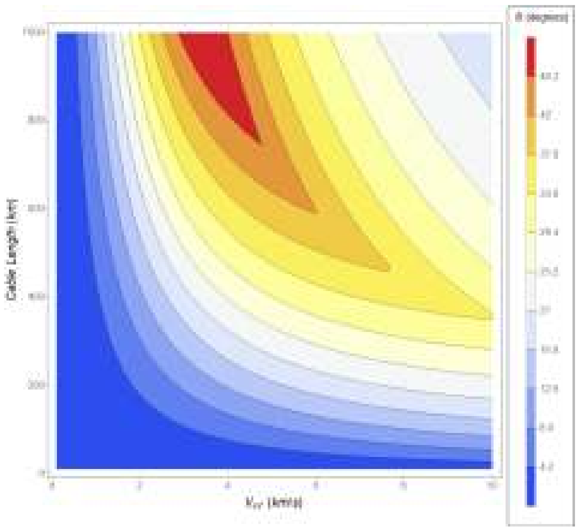

Fig.5 - The variation of the inclination of the orbital plane using the TSSM.

The next step in this research is the study of the tethered maneuver using a multi-tethered solution, fixing several tethers in the surface of the Moon to change orbit of the spacecraft that is passing by the moon.

2.1 ESTIMATING THE NUMBER OF TETHERS

The idea now is to estimate the number of tethers required to make a given inclination change. We consider the length of the tether to be given by L and the radius of the moon is denoted by R. The number of tethers required to change the inclination of the spacecraft, due to the tethered maneuver, is shown in Figs. 6-9, for different ranges of the variables involved. This quantity is a function of the tether length L and the inclination required by the maneuver. Of course, when smaller tethers are used, a large number is required. Figure 6 shows the number of tethers as a function of the desired rotation angle for an interval from 30º to 90º and the tether length in the interval from 1 to 10 km. Analysing this figure we can see that, using a tether with L = 10 km, it is necessary to use 50 tethers to rotate the spacecraft by 90º. Figure 7 shows the same maneuver with tether rotation angles from 90º to 180º, with the same parameter but with a maximum for the length of the tether given by L = 10 km. In this case it is needed 140 cables to rotate the spacecraft. Considering the same simulation, Figures 8 and 9 show the maneuver with tether rotation angles from 30º to 90º, but now with a maximum length for the tether of L = 100 km. The variation of the length of the cable is also a determinant factor in the amount of rotation made by the maneuver.

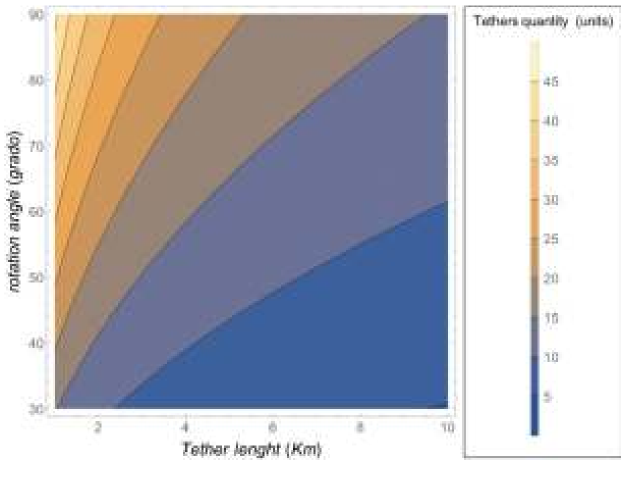

Fig.6 – Number of tethers as a function of the desired rotation angle (from 30º to 90º) and the tether length in the region from 1 to 10 km.

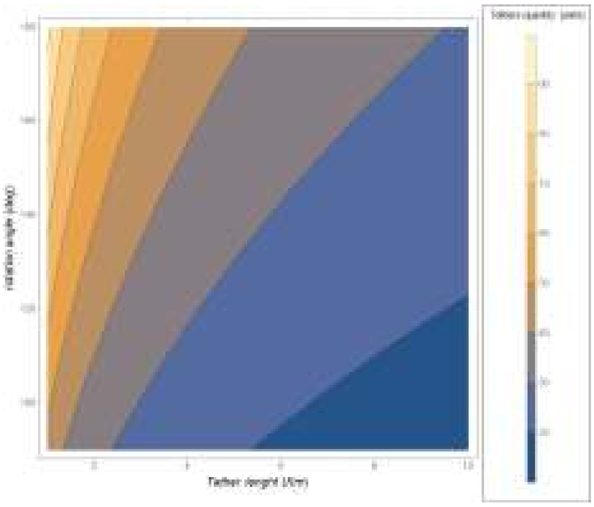

Fig. 7 – Number of tethers as a function of the desired rotation angle (from 90º to 180º) and the tether length in the region from 1 to 10 km.

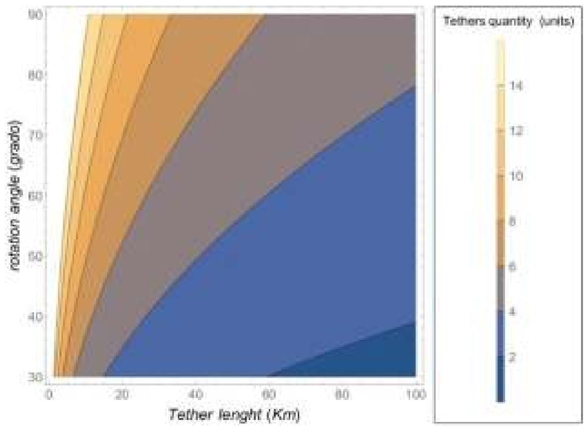

Fig. 8 – Number of tethers as a function of the desired rotation angle (from 30º to 90º) and the tether length in the region from 1 to 100 km.

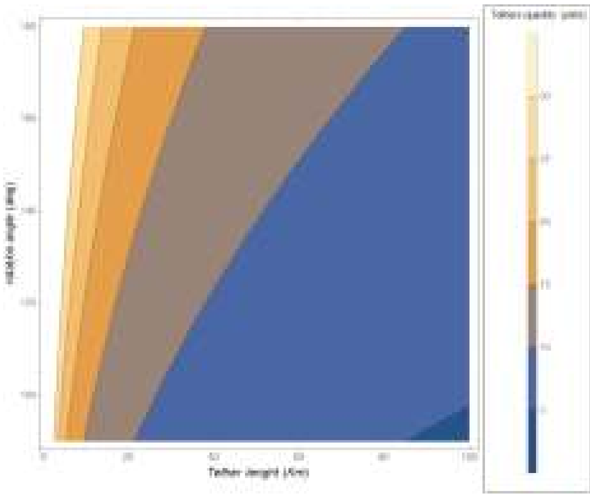

Fig. 9 – Number of tethers as a function of the desired rotation angle (from 90º to 180º) and the tether length in the region from 1 to 100 km.

2.2 TETHER TENSION EVALUATION

In this section the tension that the tether is subjected during the maneuver is calculated, based on the variations of the mass and velocity of the spacecraft and the tether size. The idea is to make a first calculation of the values involved.

It is possible to determine the tether strength requirements as a simple function of the velocity of approach (V_{inf}). It is assumed, in a first approximation, that the gravity of the moon is not strong enough to change this velocity too much during the maneuver. It means that the rotation is assumed to be done with a constant velocity. To perform this study, the cable is assumed to be inextensible and with no mass. Izzo and Wittig (2016) showed that the total mechanical stress in a given point of the tether is given by the force exerted by the spacecraft on the tether, which means that the internal forces due to the rotation of the cable is not considered. The force at a point $r$of the tether, assuming an uniform linear density μ and total length r, rotating around the origin with angular velocity ω is given by

$$F_t(\tilde{r}) = \int_{\tilde{r}}^{r} \mu r \omega^2 \, dr = \frac{1}{2}\mu(r^2 - \tilde{r}^2)\mu^2 \qquad (8)$$

A spacecraft with mass m is connected to the tether at a distance r to the surface of the Moon. Then, the radial force exerted by the tether on the spacecraft is modeled by Izzo and Wittig (2016) as

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$$F = \begin{cases} -F & \dot{r} > 0 \\ \min(-F, -m\frac{v^2}{r}) & r \doteq 0 \\ 0 & \dot{r} < 0 \end{cases} \quad (9)$$

The Centripetal Force is given by

$$F_c = m r \omega^2 \quad (10)$$

The tension (T) is given by:

$$T = \frac{mv^2}{r} \quad (11)$$

From Equation 11, the simulation of the tension on the tether caused by the spacecraft with 500 kg of mass and approaching the Moon with a maximum velocity of 10 km/s is shown in Fig. 10. This quantity is directly proportional to the mass of the spacecraft and its velocity and inversely proportional to the length of the tether. Figure 11 shows the simulation of the tension on the tether caused by the spacecraft with 1000 kg of mass and approaching the Moon with the same maximum velocity of 10 km/s.

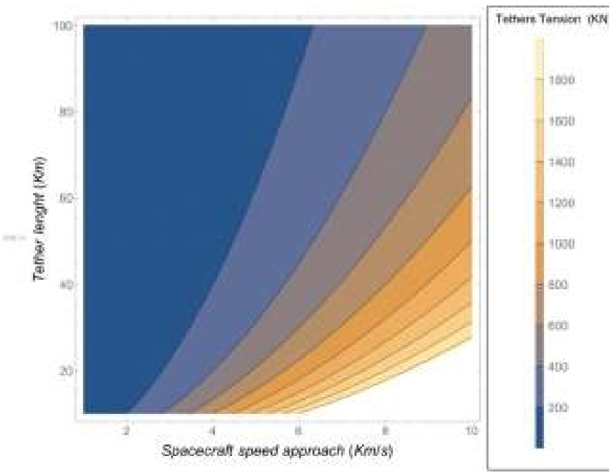

Fig 10. – Tension in the tether for a spacecraft with a mass of 500 kg.

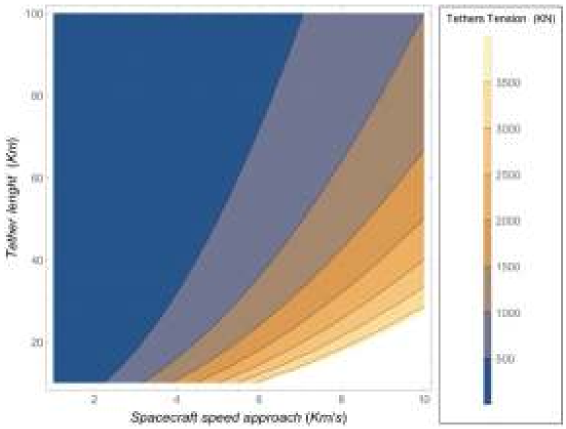

Fig.11 – Tension in the tether for a spacecraft with a mass of 1000 kg.

2.3 MULTI-TETHER TENSION EVALUATION

The design of the multi-tethered solution to make a change in the inclination of the orbit of a spacecraft is based on the idea of fixing tethers in the surface of the Moon. The sizes and arranged according to a geometry of interest for the mission. In a very trivial way, the performance of multi-tethers can be seen in Figure 12.

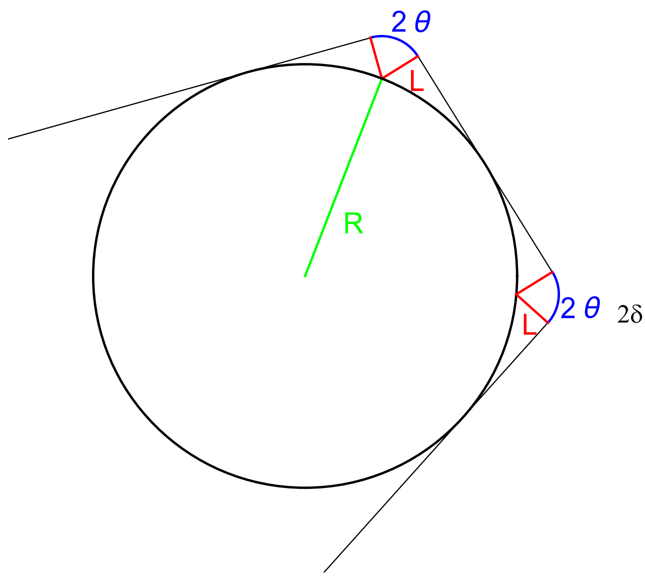

Fig. 12– Geometry of the multi-tether design.

Defining "r" as the distance between the spacecraft and the center of the moon, it is possible to find the next relations.

$$(L\cos(\delta) + R)^2 + (L\sin(\delta))^2 = r^2 \quad (12)$$

Solving, we have:

$$L^2 \cos^2(\delta) + 2RL\cos(\delta) + R^2\ L^2 \sin^2(\delta) = r^2, \quad (13)$$

What results in

$$r = \sqrt{L^2 + 2RLcos(\delta) + R^2} \quad (14)$$

Based on the concept of Energy Conservation, we have that:

$$\frac{mV^2}{2} - \frac{\mu m}{r} = \epsilon \quad (15)$$

And

$$\epsilon = \frac{mV_{inf}^2}{2} \quad (16)$$

The tension in the tether as a function of the angle of rotation (δ), is given by

$$\frac{mV^2}{R} = T + \frac{\mu m}{r^2} \quad (17)$$

$$T = -\frac{\mu m}{r^2} + \frac{2\epsilon}{R} + \frac{2\mu m}{Rr} \quad (18)$$

So, the tension in the tether per unit of mass (T_t) is given by:

$$\frac{T}{m} = -\frac{\mu}{r^2} + \frac{V_{inf}^2}{R} + \frac{2\mu}{Rr} = T_t \quad (19)$$

where

"R" is the radius of the moon and "μ" is the gravitational parameter of the Moon.

Now, solving for "r", we will have

$$r = \left(\frac{\frac{-2\mu}{R} \mp \sqrt{\left(\frac{2\mu}{R}\right)^2 - \frac{4V_{inf}^2}{R}\mu}}{\frac{2V_{inf}^2}{R}} \right) \quad (20)$$

Finally, we substitute Equation (20) in Equation (19), to make simulations of the tension in the tether using a numerical solution.

The simulations were made initially for V_{inf} = 1 km/s, with the rotation in the interval from 0 to π/2 and with the tether length assuming the values 1 km, 10 km, 100 km and 1000 km.

Then we change V_{inf} to 3 km/s and 10 km/s, keeping the other variables as described in the previous paragraph.

Finally, after several simulations we could conclude that the tension is almost constant, with little variation due to the effect of the gravitational attraction of the Moon.

CONCLUSIONS

This study described the possibility of making a TSSM using the moon to make a variation of energy and inclination in a spacecraft. The results showed the advantages of the maneuver using the tethers It can give larger variations of energy and inclination.

It also shows the benefits of using several tethers, and not just one. The sizes and tensions are different compared to the solutions using just one tether.

The maximum stress of the material was shown, which is a limiting factor for the maneuvers.

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