IAC-12-B4,2,11,x15102

ATTITUDE DYNAMICS ANALYSIS OF AALTO-1 SATELLITE DURING DE-ORBITING EXPERIMENT WITH PLASMA BRAKE

Osama Khurshid
Aalto University, Finland, osama.khurshid@aalto.fi

Pekka Janhunen,
Finnish Meteorological Institute, Finland, pekka.janhunen@fmi.fi

Matthias Buhl
Berlin Space Technologies, Germany, buhl@berlin-space-tech.com

Aarto Visala
Aalto University, Finland, aarto.visala@aalto.fi

Jaan Praks
Aalto University, Finland, jaan.praks@aalto.fi

Martti Hallikainen
Aalto University, Finland, martti.hallikainen@aalto.fi

Aalto-1 is a university CubeSat mission carried out by a consortium of Finnish universities and RD institutions. The project is lead by Aalto University's School of Electrical Engineering. The satellite is designed to have a two-phased science mission. The first six months of the mission are dedicated to remote sensing experiments, executed with an on-board spectral imager and a radiation monitor. The second mission phase is de-orbiting device test, conducted with Electrostatic Plasma Brake (EPB) instrument. This payload and de-orbiting concept is developed by the Finnish Meteorological Institute (FMI) and it is based on the principle of the electrostatic interaction with moving plasma. It uses a charged tether which will experience a drag due to Coulomb force whenever there is a relative motion between plasma and tether. The de-orbiting device experiment is divided into four sub-phases, to obtain all the needed measurements, to prove this concept. The EPB experiment phase begins with spinning up of the satellite using magnetorquers to an angular velocity of $200\ ^\circ/s$ while maintaining the satellite's rotation axis parallel to inertial Z-axis. This is needed to keep the tether stretched by centrifugal force during the deployment. The EPB tether will then be deployed to a length of 10 m, using a motor in the EPB system. An analysis has been made to determine that the tether-plasma interaction can be observed from the change in the spin rate of the satellite. The tether will be activated close to the poles, with positive and negative voltages, in order to take advantage of the orientation of Earth's local magnetic field. The third sub-phase includes further reel-out of the tether up to a 100 m length. In this last phase of the EPB experiment, the satellite's spin rate is brought down, by tether extension, to around $27\ ^\circ/s$. The charged tether will then exert a force against the direction of orbital velocity and will tend to brake the satellite into Earth's atmosphere. The collected satellite position, attitude, spin rate, EPB tether charge and satellite deceleration rate measurement data will be sent to the ground station during each pass. The dynamics, ADCS's operation and effects of the force exerted by the deployed tether have been analysed in different phases. Also, the preliminary simulations of its behaviour coupled with the satellite's ADCS have been performed.

I. INTRODUCTION

The orbital debris is a growing concern and a challenge to overcome. Future space missions might face problems if the debris would keep increasing at the current rate. At the end of mission, most of the satellites stay in the orbit adding to the debris. The LEO region of 600-1000 km needs special attention because of its use for earth observation missions. Therefore, there is a need to take effective measures that will help reduce the rate of debris increase. One of the ways to reduce the rate of increase is to bring the satellites down into the atmosphere where they will burn due to excessive atmospheric drag. However, this would require sufficient amount of fuel if rocket propulsion is used. Hence, this further increases the interest of the space community in alternate deorbiting methodologies. Since the invention of electric solar sail [1], efforts are continuously being made to utilize the concept for deorbiting the

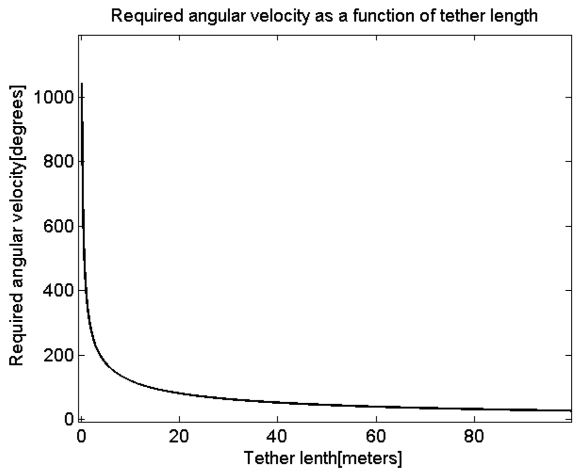

Figure 1: Tether tension as a function of angular velocity

satellites[4]. Some examples of the work done in this domain are presented in [7], [3].

The small satellites need special attention. They are easy to develop in comparison to conventional satellites and they also provide low-cost access to space. However, they have a shorter mission lifetime and would largely contribute to space debris, if not deorbited after the end of mission. CubeSats are the most important category in this regard. They should have the orbital decay lifetime of less than 25 years [2] but most of the CubeSats have the decay lifetime longer than that.

Aalto-1, a cubesat project in Aalto University, Finland will fly an electrostatic, tether-based deorbiting experiment. The experiment is designed to deorbit the satellite at the end of its mission and reduce the orbital decay lifetime in LEO environment. This paper mainly presents the attitude dynamics analysis of the experiment and the requirements, from attitude determination and control system's (ADCS) point of view.

This paper is divided into seven sections. The second section section of this paper presents an overview of the experiment. The third section presents the mechanics of spinning up the satellite and then reeling out the tether. The fourth section presents the mechanical model of the tether, using the spring and damping forces. The fifth section presents spin controller that will be used for the whole experiment phase and a few complications for using the controller. The sixth section presents the basics of charged tether phase and the last section then concludes the paper and presents the future work.

II. THE EXPERIMENT

Electrostatic Plasma Brake (EPB) [3] is a simple de-orbiting device. The idea behind this deorbiting technique is the electric solar wind sail [1]. A charged, conducting tether will experience Coulomb drag from the plasma whenever the plasma is moving with respect to the tether. This fact can be used for efficient, propellantless, interplanetary spacecraft propulsion. This is also called electric solar wind sail. This can also be used for braking down or deorbiting the LEO satellites [3]. Aalto-1, in the last phase of its mission, will deploy a 10-100 m tether to test this proposition. This will also be tested to possibly deorbit the satellite at the end of the mission using the novel 'plasma brake' mechanism. The plasma brake payload is designed to fit on one PCB of dimensions 10 x 10 x 15 cm. The module will weigh a little over 100 g.

III. SPIN-UP AND TETHER REELOUT

During the first phase, the satellite has to be spun at a high angular velocity. It is required to provide the centrifugal force that can assist in tether deployment and keep it stretched. This helps in avoiding tether's getting entangled with the satellite or with itself. The required centrifugal force is provided by the point mass attached to the end of the tether. The satellite has to be spun to an angular velocity that can keep the tether tension within certain boundary limits.

Spin rate requirements

The experiment will use a point mass of 0.5 grams attached to the end of a three-fold aluminum hoytether [5], having a linear mass density of 1.155e-5 $\frac{kg}{m}$. The tether has a base wire of 50 μm and three loop wires of 25 μm each. The point mass will be used to provide the required tension in the tether, necessary tokeep it stretched. A certain angular velocity is required to maintain the appropriate tension. The amount of angular velocity, required, depends upon the tether length. It has to be determined while considering that the tether can sustain a maximum of 50 mN, when it is stretched [6]. Using the equation 1, the current, required angular velocity can be calculated at any time instant if the tether length is known.

$$\mathbf{F_c} = \mathbf{T_{tether}}$$

$$(m_{pm}(l) + m_{tether}\tfrac{l}{2})\omega_{req}^2 = \mathbf{T_{tether}} \quad (1)$$

$$\omega_{req} = \sqrt{\frac{\mathbf{T_{tether}}}{(m_{pm}l + m_{tether}\frac{l}{2})}}$$

where, F_c is the centripetal force acting on the end point mass, m_{pm} is the point mass attached at the end of tether, ω_{req} is the required angular velocity, m_{tether} is the instantaneous mass of the tether, considered to

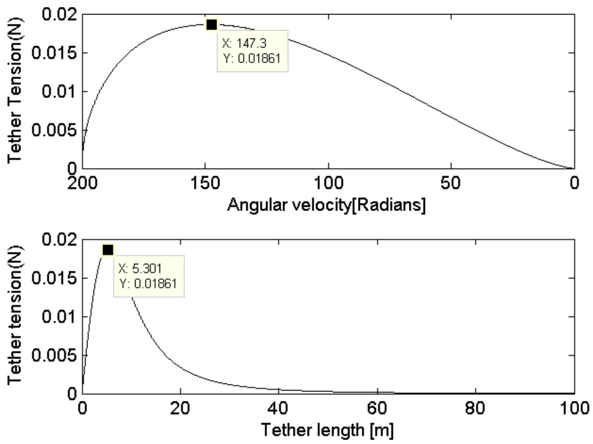

Figure 2: Tether tension as a function of angular velocity

be lying at half the tether's instantaneous length, and T_{tether} is the tension in the tether at any given instant of time. Also, $l = l_0 + 0.15$ where l_0 is the length of the tether at any instant and the factor of 0.15 shows the 15 cm displacement of the tether reel out motor, from the center of mass(CoM) of the satellite. Similarly, the maximum angular velocity can be determined at any instant by the following relation,

$$\omega_{max} < \sqrt{\frac{\mathbf{T_{ult}}}{(m_{pm} l + m_{tether} \frac{l}{2})}} \tag{2}$$

where, T_{ult} is the maximum or the ultimate tension that the tether can sustain, 50 mN in our case with certain safe margin.

A continuous tension of approximately 20-25 mN is sustainable by the tether under normal conditions. The angular velocity required to maintain the tether tension at 25 mN, with a 0.5 grams end point mass, is shown in Fig.1

The required tension can be maintained by spinning the satellite, initially, at a fictiously high angular velocity of 1045°/s. This rate is not realizable due to certain design restrictions and limitations. Therefore, present aim is to spin the satellite up to 200 °/s. Tension is maximum in the tether at the point where it is attached to the satellite. At this point, it must not exceed 50 mN. The tension in the tether changes as the tether length and hence the angular velocity change. It has been analyzed at the attachment point while assuming that the tether mass is lying at half the length of total reeled out tether, all the time. This is presented in Fig. 2.

The tether tension rises up to a maximum of 18.61 mN; which is well below the aimed mark therefore, the system needsto be spun again. Thus, whenever the angular velocity will drop below the required mark then the system would need to be respun to a higher angular velocity. The new angular velocity would depend on the tether length. The angular velocity at maximum tension point, in Fig.2, is about 147°/s. If the system is not spun again during the deployment phase then there is a possibility that tether will get entangled with itself or with the satellite. Thus, the tether will be kept stretched only if the angular velocity is maintained at a certain level.

The spin axis

The spin axis of the satellite is needed to be aligned with the spin axis of the earth. This is to keep the force vector, acting on the current carrying charged tether, within the spin plane. The reason being that the force acting on charged tether should not tilt the spin plane and only change the spin rate. The details are explained later in section six.

Tether reel-out and Change of inertia, angular velocity and momentum

Once the satellite has been spun to an initial velocity of 200 °/s, the second phase would be for tether deployment. The tether will be deployed using a special deployment mechanism. The reel mechanism consists of a motor that winds out the tether [11]. In this phase the satellite is required to maintain the angular velocity, corresponding to a particular length of the tether and the required tension in it.

This phase brings forward the requirement that the ADCS controls the nutational motion and minimizes the tumbling of the satellite. The spin motion will be about the axis of maximum inertia. In current simulations, the convention for the body coordinate frame and the satellite inertia principal axes are shown in Fig.3. Where, $\mathbf{x_p}$ is the axis of minimum, $\mathbf{y_p}$ is the axis of maximum and $\mathbf{z_p}$ is the axis of intermediate moment of inertia.

In the current satellite mechanical model, the principal y-axis is deviated from the body y-axis by an angle of 5°. This is because of the mass distribution in the satellite. This angle is important because the tether reels out along -z_b axis. Thus, if the y_p would have been perfectly aligned with the y_b axis then the tether would reel out at 90° angle to the spin axis. But, any misalignment between the two axes will cause the tether to reel out at a certain angle not equal to 90°. The maximum allowable angle deviation has been determined, considering the hardware limitations, to be 10°. Thus the allowable angle is 90±10°. The current angle is 5°.

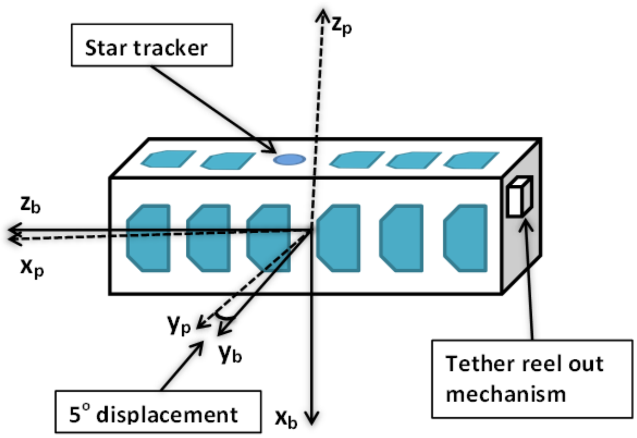

Figure 3: The body reference frame convention and principal inertia axes

This deviation does not pose any problem as it is within allowable limits.

During the tether deployment, the inertia of the satellite constantly changes. The change in inertia has to be known to be able to compute the required torque from the magnetorquers. The spin controller maintains the satellite's angular velocity and momentum at a required mark. It is important for the controller to know the changes in angular velocity and hence the momentum. Therefore, an accurate model of changing system inertia is required. The change in the principal inertia components can, roughly, be modelled by using the following relation,

$$\begin{aligned} \mathbf{I}_{total} &= \mathbf{I}_{sat} + \mathbf{I}_{thether} + \mathbf{I}_{pointmass} \\ &= \tfrac{1}{12} m_{sat}(h^2 + d^2) + \tfrac{l \cdot \lambda r^2}{3} + 2 m_{pm} r_{pm}^2 \end{aligned} \tag{3}$$

where, m_{sat} is the mass of the satellite, l is the length of the tether at time t, λ is the linear mass density of the tether, m_{pm} is the point mass attached to the end of the tether and r_{pm} is the distance, of the point mass, from the system's CoM.

The inertia is determined while assuming the tether to be a rigid rod of infinitely small radius. However, the equation 3 provides only the change in principal inertia components and not the products. The inertia products are also very important in order to know the change in the principal axes and their relation with the body axes. At this stage of developement, the detailed model of satellite mass distribution is not available. Therefore, only the simplified model is used to estimate the inertia tensor for continously changing tether length. The change in principal inertia components is presented in Fig.4. The inertia about the y_p and z_p axes increases as the tether is reeled out.

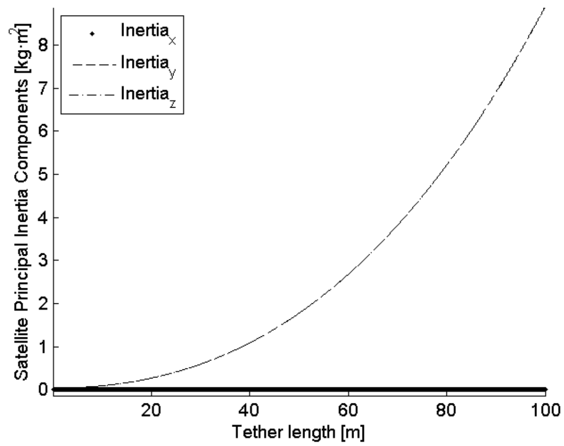

Figure 4: The change in inertia with changing tether length

As the tether reels out, it also changes the CoM of the rotating satellite-tether system. This also causes a change in the inertia products. The initial CoM of the satellite is located at [-0.22, -0.05, 0.93] cm from the geometrical center of the satellite. The final CoM, when the tether is completely reeled out, is at [-0.22, -0.05, 4.095] cm from the geometrical center of the satellite. Thus, the total displacement of about 3 cm is observed. It is assumed, during this calculation, that the tether is completely aligned with Z_{body} axis.

The satellite inertia changes however, as the system is unforced therefore, the total angular momentum is conserved i.e,

$$\begin{aligned} \mathbf{L}(t) &= \mathbf{L}(t + \Delta t) \\ I_{total}(t)\omega(t) &= I_{total}(t + \Delta t)\omega(t + \Delta t) \end{aligned} \tag{4}$$

The $\omega(t + \Delta t)$ is therefore reduced with the reeling out of the tether. The reel out rate of 1.2 $\frac{mm}{s}$ is used in currently designed system. The initial conditions for the simulations presented in this section are:
$\omega_p(0) = \left[0.01, 3.49, 0.01\right] radians/s$,
$I_p(0) = \left[0.008919, 0.041274, 0.040903\right] kg \cdot m^2$ and,
$L_p(0) = \left[0.000089, 0.144047, 0.000409\right] kg{\cdot}m^2 radians/s$

Fig.5 presents the change in the angular velocity as a function of tether length. However, it only shows how the angular velocity would change if the tether length is changing in an unforced system. The system would need to be spun to a higher angular velocity whenever the angular velocity drops below the threshold, given in Fig.1 for respective tether length. Therefore, the angular velocity will not be allowed by the controller to fall below the required level. The total angular momentum is conserved in inertial space. The principal components of the angular momentum are shown in Fig.6.

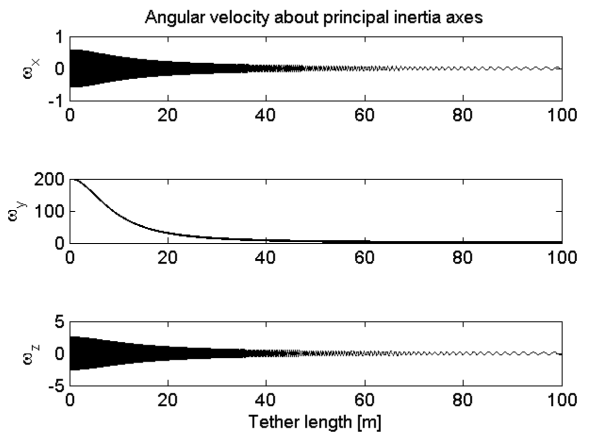

Figure 5: The change in angular velocity during reeling out of tether

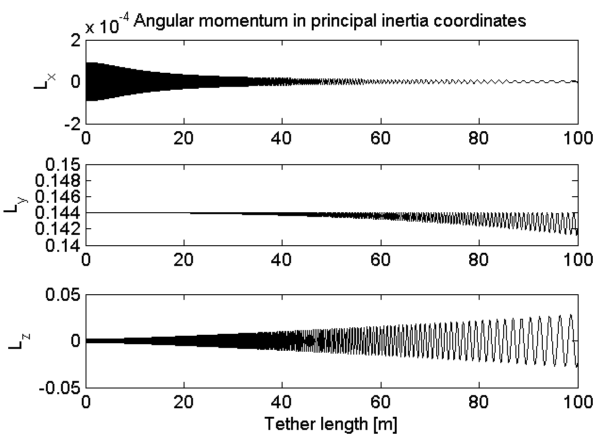

Figure 6: The change in angular momentum during reeling out of tether

The change in angular velocity changes the nutation angle of the satellite spinning at an initial angular velocity of 200 $^\circ/s$. The satellite, at this stage, acts as a rotating prolate body. The prolate body rotates in the manner that momentum vector rotates at a smaller angle than the velocity vector, about the rotation axis [4], [8]. This is also shown by Fig.7. It can be observed from the figure that the angular momentum unit vector stays closer to spin axis, y-axis, and the angular velocity unit vector rotates about it. The reason for the continously increasing nutation angle between angluar momentum, L, and angular velocity, ω, is the change in satellite inertia.

The nutation angle increases with time. This can be observed from Fig.8.

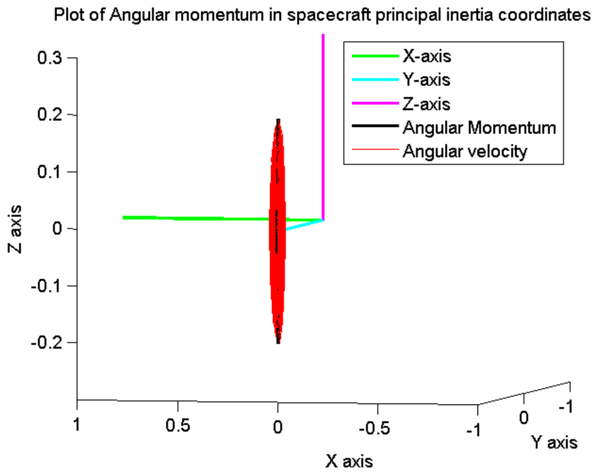

Figure 7: The change in inertia with changing tether length

IV.TETHER MODEL

The tether is modelled as a spring-damper system. The effects of spring damper systemcan be analysed by using using the spring and the damping forces' model. This gives the estimation of the tether charecteristics and torques generating from its deployment. These characteristics are important to consider while analysing the affects during the complete experiment.

The spring force is given by the Hooke's law,

$$\boldsymbol{F}_s = -k\boldsymbol{x} \tag{5}$$

where, $\boldsymbol{F_s}$ is the force between the point mass and the satellite body, k is the spring constant and $\boldsymbol{x}$ is the displacement of the tether from its equilibrium position. The damping force is modelled in two ways. First, by using the simple damping force which is,

$$\boldsymbol{F}_{d1} = -c\boldsymbol{v_n} \tag{6}$$

and secondly, by using a more realistic version of the damping model for elastic materials. This model is shown in the following equation,

$$\boldsymbol{F}_{d2} = -\tau E_l k\boldsymbol{x}\boldsymbol{v} \tag{7}$$

where,

$$\tau = \frac{|r_{avg}|}{|v_{avg}|} \tag{8}$$

and E_l is the relative loss modulus.

The total torque acting on the spinning satellite, due to the deployed tether, is presented in Fig.9.

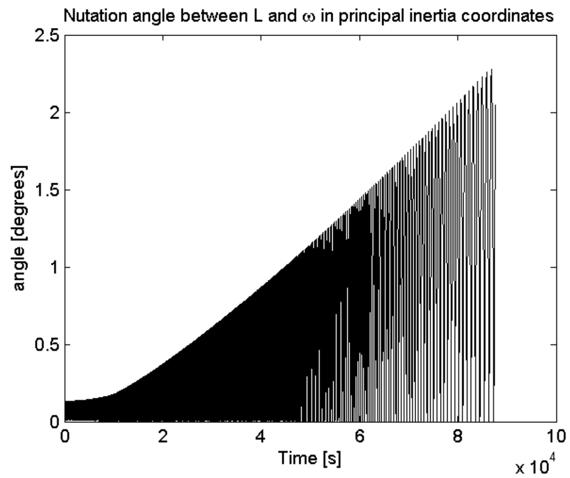

Figure 8: The change in angular velocity during reeling out of tether

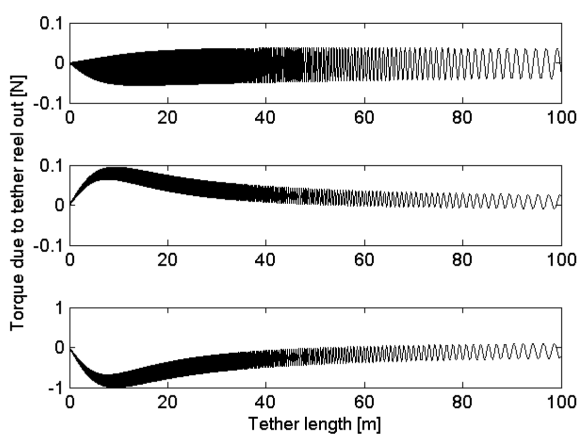

Figure 9: The torque acting on the satellite as a result of spring motion of tether

V.CONTROL SYSTEM REQUIREMENTS AND IMPLEMENTATION

The spinning satellite, with the changing inertia tensor as explained in previous section, poses the requirement of controlling the spin motion. For such a spin motion of the satellite, there are three factors that need to be considered. These are spin control, nutation control and the precession control [9].

Spin Controller

The most feasible means of spinning up the satellite to an initial rate of $200°/s$, in LEO, is by using magnetorquers. This is because of the known fact that magnetic field is very strong in LEO environment and could easily be used for attitude control. Therefore, it is designed to obtain the desired angular velocity by using only the magnetic actuation. Thus, spinning up the satellite to such a high rate needs continuous actuation of magnetorquers in order to be able to generate enough torque to control spin and nutational motion. For this purpose, the controller that will be used is presented in [10] and the simplified control equation is,

$$\mathbf{m} = \frac{-k}{||B||^2}[\underline{B} \times (\underline{\tilde{h}} + k_1 \underline{e_h})] \tag{9}$$

where, $\mathbf{m}$ is the magnetorquer dipole moment vector in principal inertia coordinates, $\underline{\mathrm{B}}$ is the earth magnetic field vector expressed in body principal coordinates, $\underline{\tilde{h}}$ is the satellite angular momentum error vector in principal inertia coordinates and $\underline{e_h}$ is the angular momentum error vector about principal body axis about which final momentum is required.

The controller operation needs several inputs for determining the duration of actuation of magnetorquers and the instants when to actuate them. These inputs are current angular velocity, tether length, current inertia of the system, current attitude of the system and required angular velocity or the momentum. The spin controller has been, prelimineraly tested to determine its usability. One of the cases has been presented here. The point where the tension starts to drop, in Fig.2, has been considered. At this point the following initial conditions have been used for the spin controller:

current angular velocity $\left[-0.0083, 2.5988, 0.0099\right]$
tether length $= 5.301$ m
current inertia $= \left[0.008921, 0.05544, 0.05507\right]$
current attitude$= \left[0.0421, 0.4618, 0.0099, -0.8859\right]$
required angular velocity $= \left[0.0, 3.0124, 0.0\right]$
required angular momentum $= 0.1670$ about y_p

The resulting change in the angular momentum is shown in Fig.10. It can be observed from the figure that the angular momentum about the x_p and z_p reduces with time and the momentum about y_p is brought to the required mark. This restores the required tension in the tether. Thus, loop iterations are required for running the controller. The motor, that reels out the tether, contains an encoder. Thus, the rate and the total length of tether, at any instant, will be obtained from the encoder outputs. The angular velocity will be determined by the gyroscopes, incorporated in the ADCS. Using the current tether length and the angular velocity, at any instant, provides the tension in the string. This tension would be used as the variable that controls the turning on or off of the spin controller. Once the string tension goes below the required mark

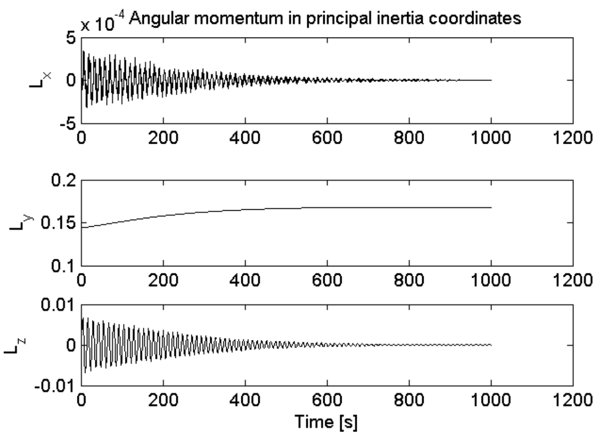

Figure 10: The change in angular momentum during spinning up the satellite to required velocity

and the controller is commanded to turn on, it will calculate the current inertia of the system, obtain the attitude information, calculate the amount by which the angular velocity has to be increased and then it will provide the necessary magnetic torque through magnetorquers.

The important factor in using magnetic control is the size and torque capabilities of magnetorquers. The current mgnetorquer design provides a dipole moment of 0.2 Am^2. The magnetorquers, with this dipole moment, would provide an average torque of about 4 μNm. In the example case, presented above, it will take about 95 minutes to provide the required change in angular momentum. Thus, the tether deployment would be halted at the time of actuation and once the required momentum is achieved then the deployment will start again.

VI.CHARGED TETHER PHASE

When the tether is charged then the force acting on the tether is given by the following relation [1]

$$\boldsymbol{F}_{plasma} = 1.72 P_{dyn} l_t \left[\frac{\epsilon_0 V_0}{e n_0}\right]^{1/2} exp\left[\frac{-m_i v_0^2}{2eV_0}\right] \qquad (10)$$

$$V_0 = r_w E_0$$

Where, E_0 is the electric field on the surface of the tether, r_w is the radius of tether wire and V_0 is the tether potential relative to plasma.The electric field on the surface of charged tether, E_0, is obtained by integrating Gauss law around the wire. Thus, a relation between E_0 and charge per unit length is obtained. The charge per unit length can be related to the tether voltage once the width of the electron sheath, around the charged tether, could be approximated. Above relation defines the force acting on the whole tether length l_t.

This force is always directed in the direction of orbital ram flow and perpendicular to the tether length. The ram flow is in the direction opposite to that of the orbital velocity. When the spin plane and the orbital plane are the same, the ram flow will always be in the spin plane. Therefore, the spin plane will not change because of the acting force. In Aalto-1 specific case, the satellite orbital plane will not be the same as spin plane all the time. Therefore, the spin plane will have disturbances that will tend to tilt it.

The experiment is currently designed so that the plasma brake experiment will be performed near the poles. The reason being that the ram velocity vector of the ionospheric plasma (with respect to the satellite) lies in the tether spin plane, so the flow can change the spin rate maximally. If there is a minimal factor of the force in the transverse plane then the change in the spin rate will be directly translated to the total force generated due to tether-plasma interactions. This will provide the scientific data for future developments.

VII.CONCLUSION

This paper has presented the first attitude dynamics analysis of deorbiting experiment with electrostatic plasma brake. The important aspects of the presented analysis reveal that it is important to know the orientation of spin axis with respect to the body axes because, this is directly linked with the tension in the deployed tether. In order to avoid the entangling of the tether, the system needs to maintain the angular velocity at a particular mark. This required angular velocity depends upon the required tether length and tension. It was shown that the spin motion needs to be controlled. Magnetic actuation is the most feasible to control motion at high angular rates, like 200 $°/s$. Finally it was shown that the angular momentum deviation, in body principal frame and in inertial frame, from the desired respective axes can be controlled using active magnetic actuation. Future work will include the improvement of the mathematical model for variation in inertia tensor, detailed analysis of spin controller and its actuation conditions and further analysis of the charged tether phase.

ACKNOWLEDGEMENTS

This work has been supported by Aalto University's Multidisciplinary Institute of Digitalisation and Energy (MIDE) and Tekes - Technology and Innovation.

References

[1] P. Janhunen, Electric sail for spacecraft propulsion, J. Prop. Power, 20, 763-764, 2004.

[2] Cal Poly, Cubesat Design Specification Rev.12, The CubeSat Program, Cal Poly SLO

[3] P. Janhunen, Electrostatic Plasma Brake for Deorbiting a Satellite, Journal of propulsion and power, Vol.26, No.2, March-April 2010,

[4] M. J. Sidi, Spacecraft Dynamics and Control - A Practical Engineering Approach pp.[97-100]

[5] R. P. Hoyt, R. L. Forward U.S. Patent No. 6286788, 2001

[6] H. Seppnen, S. Kiprich, R. Kurppa, P. Janhunen and E. Haeggstrm, Wire- to-wire bonding of um-diameter aluminum wires for the Electric Solar Wind Sail, Microelectronic Engineering, 88, 3267-3269, 2011

[7] E. Ahedo and J. R. Sanmartin, Analysis of Bare Tether Systems for Deorbiting Low-Earth-Orbit Satellites, Journal of Spacecraft and Rockets, Vol.39, No.2, March-April 2002

[8] E. Butikov, Inertial rotation of a rigid body, 2006 European Journal of Physics 27, 913 doi:10.1088/0143-0807/27/4/022

[9] R. Reynolds, G. Creamer, Global Lyapunov Control of Spin Stabilized Spacecraft 2001 Flight Mechanics Symposium; 01 June 2001. pp. 285-293. 2001

[10] Anton de Ruiter, A fault-tolerant magnetic spin stabilizing controller for the JC2Sat-FF mission, Acta Astronautica, Volume 68, Issues 12, JanuaryFebruary 2011, Pages 160-171

[11] O. Krmer, R. Rosta, L. Richter and P. Janhunen, SAMUEL (Space Applied Mechanism for Unreeling ELectric conductive tethers), EPSC Abstracts Vol.5, EPSC2010-229, 2010 European Planetary Science Congress 2010