INTEGRATED MOMENTUM MANAGEMENT AND POWER GENERATION BY DOUBLE GIMBALED SOLAR ARRAYS FOR NANO-SATELLITES

K. auf der Heide
D-85640 Solalinden, Germany, Kolja@auf-der-Heide.de
Prof. Dr. K. Janschek,
Technische Universität Dresden, Institute of Automation, D-01062 Dresden, Germany

Abstract

The article presents a system design concept for the momentum management with double gimbaled solar arrays on nano-satellites, where the two functions momentum control and optimal power generation are integrated to save mass. Momentum unloading is proposed to be performed with a double gimbaled solar array by using the gravity gradient torque. It is demonstrated that the power generation can be significantly increased compared to single axis arrays although combining two functions on the solar array. Exploitation of this method can significantly reduce the system cost and complexity by abandoning redundant thruster systems and not depending on expendables. Adaptive structures are proposed for the realization of the gimballingmechanism to reduce mass and mechanism complexity. The momentum unloading is based on a time-optimal bang-bang control law. The feasibility of the concept is demonstrated by simulation results which have been derived for different satellite configurations and LEO parameters.

1. Introduction

In the near future, smaller and more cost efficient satellites could become an important factor in space applications due to comparably less development and launch costs as well as recent advances in micro- and nano-technology. These so called nano-satellites with a total weight of about 10 kg require new concepts for the overall system design. One important part in this concept will be the transition from the traditional subsystem structure to multifunctional structures, where equipment or parts of equipment perform more than one function at the same time to reduce the system cost, decrease mass and complexity, increase reliability, and enhance the overall performance.

One important onboard task is the momentum unloading, which is widely investigated in order to optimize space missions [1,2,3]. All techniques rely on the onboard availability of controllable external torques to counteract the angular momentum changes of the momentum wheels to keep the wheel speeds within predefined limits.

This article presents a new system design concept for nano-satellites with bias-momentum wheels for attitude control. The concept proposes to integrate the function of momentum unloading and power generation into a double gimbaled solar array (see Fig. (1)) by making use of the gravity gradient torque as controllable unloading torque.

The solar arrays on state-of-the art earth oriented satellites are pointed to the sun by single axis mechanisms only allowing tracking of the sun in the orbit plane. Due to the ecliptic, even for equatorial orbits the full capability of power generation is not realized with single axis mechanisms. For orbits with higher inclination the efficiency of solar arrays with single axis mechanisms is significantly reduced and a dual axis mechanism becomes necessary.

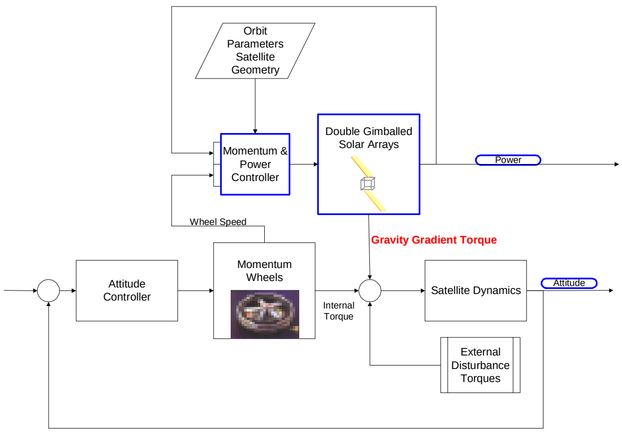


Fig. (1) Integration of optimal power generation and momentum unloading for earth oriented nano-satellites with double gimbaled solar arrays.

It will be shown that using dual axis solar array mechanisms the efficiency of the array can be increased while unloading the momentum wheels, used for attitude control. Therefore, the concept dispenses with additional torquers such as redundant thruster systems or magnetic coils leading to a reduction in mass, complexity, and cost for the overall system.

Reliability and mechanical complexity problems of a dual axis mechanism for larger satellites can be overcome for nano-satellite applications by using adaptive (smart) structures to realize the second angle rotation [4].

The paper presents the governing dynamic models and external disturbances for an earth oriented satellite with a double gimbaled solar array. External torques are derived and it is shown for the orbit altitudes under consideration, that the gravity gradient torque is dominating and therefore shall be used for unloading. System design parameters are identified to deliver general guidelines for a system design. Simulation results presented for the various parameters will show the feasibility of the proposed concept.

2. System Design

The constructive layout of the satellite system is shown in Fig. (2). The main free design parameter is the solar array (SA) length-to-width ratio, which determines fundamentally the gravity gradient torque. As the overall power demand and hence the area of the solar array are fixed by the mission requirements, it is still possible to choose the SA geometrical shape within a certain envelope (stowing and payload constraints). This free design parameter can be used to optimize the SA geometry for certain mission requirements, e.g. orbit altitude, inclination and right ascension.

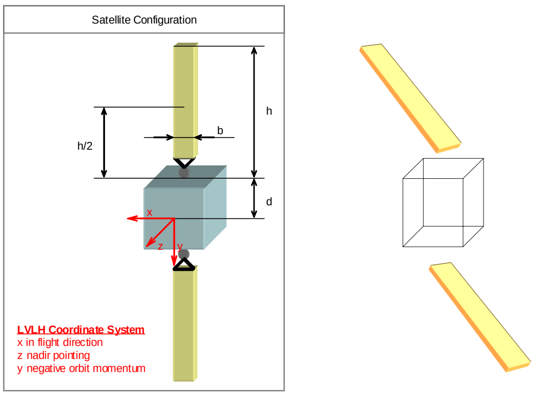


Fig. (2) General bus concept for a nano-satellite with a double gimbaled solar array. The key design parameters for the solar arrays are shown. The dual axis mechanism rotates the arrays about the pitch and a local roll axis.

The dual axis mechanism rotates the solar arrays about a LVLH-pitch and a local roll axis such that a gravity gradient torque can be generated. The solar array angles vary over a year as follows: the SA pitch axis rotates by 360° during one orbit, whereas the local roll axis varies over a year with an amplitude equal to the difference between inclination and ecliptic.

For the realization of the dual axis mechanism it is proposed to make use of adaptive structures such as piezoelectric or magnetostrictive linear motors, which allow for light weight redundant mechanisms and high reliability [4].

3. External Torques

Gravity Gradient Torque

The gravity gradient torque is defined by following formula [5]:

$$\underline{T}_{GG} = \frac{3\cdot\mu}{r^3}\cdot\left(\underline{n}_e \times [\underline{I}\cdot\underline{n}_e]\right) \; ; \; \frac{\mu}{r^3} = n^2$$

n is the orbit frequency, I the inertia tensor, and n_e a zenith pointing vector. For perfect attitude control the formula reduces to:

$$\underline{T}_{GG} = 3\cdot n^2 \cdot \begin{pmatrix} -I_{yz} \\ I_{xz} \\ 0 \end{pmatrix}$$

A gravity gradient torque only emerges around the roll and pitch axis. For the solar array in Fig. (2) the inertia tensor is calculated as a thin plate with respect to the mechanism at the bottom of the array:

$$j_{xx} = \frac{1}{3} m_{S.A.} h^2 \; ; \; j_{yy} = \frac{1}{12} m_{S.A.} b^2 \; ; \; j_{zz} = m_{S.A.}\left(\frac{1}{3}h^2 + \frac{1}{12}b^2\right)$$

$m_{S.A.}$... *solar array mass*

No products of inertia exist for an ideal gimbal.

The overall inertia tensor of the satellite is formed by the inertia of the satellite body K and the inertia of the solar array J. by rotating the array, the inertia tensor has to be determined in body coordinates. Further, for the total configuration the inertia tensor of the arrays has to be transformed into the satellite center of gravity (S).

$$I = K + 2\cdot R' \cdot J \cdot R + 2\cdot S$$

The products of inertia for the total configuration relevant for the gravity gradient torque are given as follows:

$$I_{yz} = -2\cdot\cos p\cdot\sin 2r\cdot\left(j_{zz} - j_{yy}\right) = -\cos p\cdot\sin 2r\cdot\left(\frac{1}{3}\cdot m_{sa}\cdot h^2\right)$$

$$I_{xz} = \left(\left(-\frac{1}{6}\cdot m_{sa}\cdot h^2 + \frac{1}{12}\cdot m_{sa}\cdot b^2\right) + \frac{1}{6}\cdot m_{sa}\cdot h^2\cdot\cos 2r\right)\cdot\sin 2p$$

with p the pitch angle of the solar arrays and r the roll angle. Finally, the gravity gradient torque results in:

$$T_{GG}^{X} = n^2\cdot\cos p\cdot\sin 2r\cdot m_{sa}\cdot h^2$$

$$T_{GG}^{Y} = \frac{1}{2}\cdot n^2\cdot\left(m_{sa}\cdot h^2\cdot(\cos 2r - 1) + \frac{1}{2}\cdot m_{sa}\cdot b^2\right)\cdot\sin 2p$$

Magnetic Torque

A simple model of the earth magnetic field in orbit coordinates is [5]:

$$\underline{b}^0 = \left(\frac{r_e}{r}\right)^3\cdot g\cdot\begin{pmatrix} \cos\eta\cdot\sin i \\ -\cos i \\ 2\cdot\sin\eta\cdot\sin i \end{pmatrix}$$

with i the orbital inclination and η the orbit angle defined from the ascending node of the orbit, r_e is the earth radius, and r the orbit radius. The magnetic field constant g is 30055,7 nT. To calculate the magnetic torque, the following equation is used:

$$\underline{T}_{MM} = \underline{m}\times\underline{b}^0 = \begin{pmatrix} m_y b_z - m_z b_y \\ m_z b_x - m_x b_z \\ m_x b_y - m_y b_x \end{pmatrix} = \begin{pmatrix} 2m_y \sin\eta\sin i + m_z \cos i \\ m_z\cos\eta\sin i - 2m_x\sin\eta\sin i \\ -m_x\cos i - m_y\cos\eta\sin i \end{pmatrix}\cdot g\cdot\left(\frac{r_e}{r}\right)^3 \quad (0.1)$$

with m is the magnetic dipole moment of the satellite.

Solar Pressure Torque

The solar pressure torque is dominated by large solar array areas. Generally, the torque is caused by a difference between geometrical center and center of mass. For the solar arrays the solar pressure torque is calculated as:

$$T_{SP} = 2 \cdot p \cdot A \cdot \begin{pmatrix} \Delta_y^{cog} \cdot \cos(n \cdot t) \\ \sqrt{\Delta_x^{cog^2} + \Delta_z^{cog^2}} \cdot \sin(n \cdot t) \\ \Delta_y^{cog} \cdot \sin(n \cdot t) \end{pmatrix}$$

A is the array area, *p* the solar pressure, and Δ^{cog} the difference between center of mass and center of gravity. The solar pressure can be modelled to be constant over all orbit altitudes.

Aerodynamic Torque

An aerodynamic torque is generated in low earth orbits due to atmospheric drag. The aerodynamic torque is computed as follows [5]:

$$T_{AD} = 2 \cdot \rho_0 \cdot e^{-f \cdot h} \cdot \frac{\mu}{r} \cdot A \cdot \begin{pmatrix} \Delta_y^{cog} \cdot \cos(n \cdot t) \\ \sqrt{\Delta_x^{cog^2} + \Delta_z^{cog^2}} \cdot \sin(n \cdot t) \\ \Delta_y^{cog} \cdot \sin(n \cdot t) \end{pmatrix}$$

with ρ_0 is the atmosphere's density at a reference altitude and *v* the velocity of the satellite compared to the atmosphere. The stagnation point pressure for hypersonic flows is about two times the dynamic pressure. The stagnation point pressure is assumed to be constant over the solar array area: *A* is the solar array area, μ is the gravitational constant of the earth, and Δ_{cog} is the difference between geometric center and center of gravity. A simple density model correlates the density exponentially to the orbit altitude.

Comparison of Torques

A comparison of torques shows, that for orbit altitudes higher than 600 km the aerodynamic torque can be neglected. The gravity gradient torque dominates the torques up to orbit altitude of about 15.000 km. For an orbit altitude higher than 10.000 km the solar pressure torque starts to dominate the magnetic torque.
The gravity gradient torque is shown to be dominating all other torques for low earth orbits. Therefore, the gravity gradient torque is a good candidate to unload the momentum wheels with the double gimballed solar arrays.

4. Momentum Dynamics

For the dynamics of the satellite, the principle of momentum conservation leads to a simplified description of the dynamics. This simplified expression holds for a satellite with perfect attitude control and double gimballed solar arrays. The dynamics of the rotating solar array can be neglected. The dynamic reduces to the dynamics of the momentum wheels as shown in [4]:

$$\begin{aligned} I_x \cdot \dot{\omega}_x + n \cdot I_z \cdot \omega_z &= T_x \\ I_y \cdot \dot{\omega}_y &= T_y \\ I_z \cdot \dot{\omega}_z - n \cdot I_x \cdot \omega_x &= T_z \end{aligned}$$

The resulting equation is linear so that using the torques of the previous chapter an analytic solution can be found to describe the momentum wheel dynamics for a satellite with double gimballed momentum wheels. These analytic equations can be used for verification of the simulation environment as well as to find an optimal controller.

5. Momentum and Power Control

The integrated momentum and power controller is based on the following strategy.

- Eclipse phase
  The solar arrays are not in use for power generation, so they can be tilted such to generate an appropriate gravity gradient *control torque* to unload the wheels.
- Sun phase
  The solar arrays are tilted to maximize the sun incident angle in order to optimize the electrical power generation. This results in a parasitical gravity gradient *disturbance torque*, which is absorbed as well as the other external disturbance torques by the momentum wheels. The constraint for power maximization is determined by the limited gravity gradient's control torque capability in the eclipse phase. The controller is therefore also activated for the momentum unloading in the sun phase, whenever the eclipse passage is too short to unload the momentum.

A nonlinear time optimal approach is chosen for the momentum unloading controller. For the linear momentum system, it can be shown that Pontryagin's maximum principle leads to a time optimal 'bang-bang' controller [6], which exerts maximum torques

$$u = -M \cdot \mathrm{sgn}\left(\frac{1}{I_x^w} \cdot \psi_1(t)\right)$$

where M is the maximum torque that can be produced and ψ is the solution of the differential momentum equation:

$$\psi_1 = A \cdot \sin\,(nt + \alpha)$$

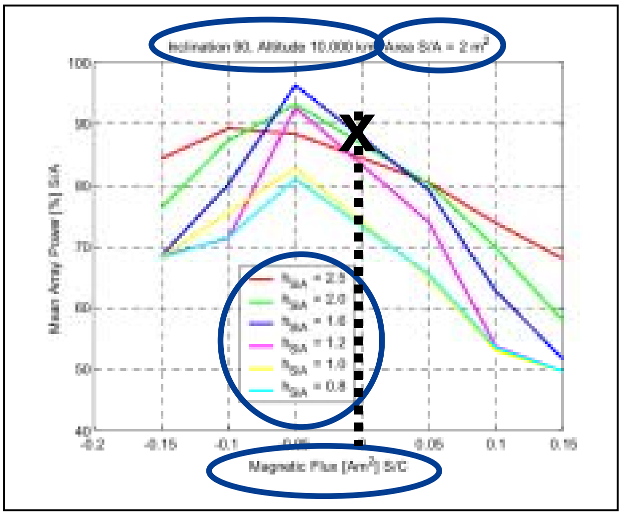
The maximum torque of a double gimbaled array can be computed from the formula of the gravity gradient torque, which shows the torque with respect to the solar array angles. A function can be established to calculate the array angles to exert the appropriate control torque.
To avoid controller chattering, an activation level is defined for the wheel speed, which turns the controller off, when a small level of momentum is reached. This approach is feasible, because it is not necessary to perfectly unload the momentum but to limit it.

6. Simulation Results

Simulations have been performed for different orbit altitudes and parameters. It can be shown that the gravity torque depends especially on the solar array height *h,* as the momentum loading in the coupled roll/yaw system is the dimensioning momentum component. Therefore, simulations are performed for different solar array heights, while the solar array width *b* is always chosen such that the resulting array area is held constant (see Fig.(3)).

All external torques are simulated except the aerodynamic torque. For the magnetic torque, the magnetic moments of the satellite body have also been varied to show the influence of the magnetic field of onboard electronic devices and to allow for an additional system design parameter.

For all configurations, the *mean power* generated over *one year* in orbit is shown.



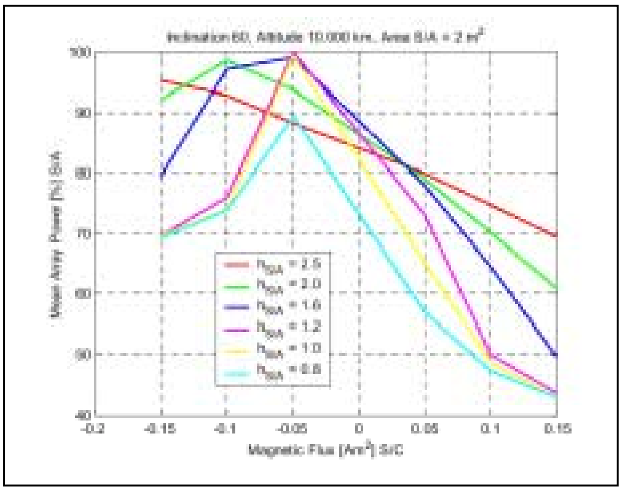


Fig. (3) Mean solar array power over one year for different mission configurations (inclination) and satellite configurations (SA-geometry, magnetic moment) combined power & momentum control.

In the plots is shown on the x-axis the variation of the magnetic moment of the satellite, produced by onboard electronic devices. The magnetic moment is kept constant and equal to the value shown for all 3 axes. On the y-axis the mean power generated by the array is shown over one year in orbit. The values are plotted in percent of the maximum nominal power. The orbit altitude is 10.000 km, while two plots are shown for an inclination of -20° and 40°. Different curves are shown for different solar array geometries. The solar array area is always kept constant to be about 2 square meters

The simulation studies have shown a considerable optimization potential in terms of solar array geometry. This means that for a specific satellite configurations considerable improvements (up to 20 %) in mean power can be achieved by an appropriate SA shape.

7. Summary

The feasibility of a double gimbaled solar array used for optimum power generation and momentum unloading has been shown.

Dimensioning parameters for the proposed scheme have been identified to be the solar array geometry, orbit parameters, and the remaining residual disturbance torques of the satellite bus (e.g. magnetic moment). Optimizing all parameters leads to a significant improvement of the generated power.

Current activities are concentrating to derive analytical relationships for the different design parameters. These relationships are planned to be implemented into a system design tool. A second research line is dealing with improved strategies for the momentum unloading, taking into account the eclipse unloading constraints for he sun phase power control.

Literature

[1] X. Chen, W.H. Steyn, S. Hodgart, Y. Hashida, Optimal Combined Reaction-Wheel Momentum Management for Earth-Pointing Satellites, Journal of Guidance, Control, and Dynamics, pp. 543-550, Vol. 22, No.4, 1999.

[2] T.F. Burns, H. Flashner, Adaptive Control Applied to Momentum Unloading Using the Low Earth Orbital Environement, Journal of Guidance, Control, and Dynamics, pp. 325-333, Vol. 15, No. 2, 1992.

[3] H. Flashner, T.F. Burns, Spacecraft Momentum Unloading: The Cell Mapping Approach, Journal of Guidance, Control, and Dynamics, pp. 89-98, Vol. 13, No. 1, 1990.

[4] K. auf der Heide, K. Janschek, Drallregelung von Nanosatelliten mittels schwenkbarer Solargeneratoren (Momentum Management by double gimballed solar arrays for nano-satellites), German Aeropsace Congress (DGLR), Hamburg 2001.

[5] J.A. Wertz, W.J. Larson, Space Mission Analysis and Design, Kluwer, 1991.

[6] A.E. Bryson Jr.,Y.C. Ho, Applied Optimal Control, Taylor & Francis, 1975.