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ATTITUDE DETERMINATION AND CONTROL SYSTEM USING MAGNETIC TORQUE RODS

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This work approaches the aim to specify for the development of an attitude determination and control system considered elements to produce and simulate an further three axis dynamic attitude control for an microsatellite. The paper considers the elements necessary for creating and modeling a microsatellite control system in three axes. Mathematical modeling of the operation of magnetic torquer and a microsatellite control system in the MATLAB environment was carried out. Based on the simulation results, a three-axis stabilization method and sensors of the orientation, navigation and control system are described. Also environmental effects called disturbance torques which influences the movement of the microsatellite and drive it away from its original attitude will be explained.

Keywords: satellite attitude control, satellite, attitude determination, control system, magnetic torque.

Goals. Goal is to formulate a possible design for magnetic torquers. Magnetic and control moments produced by the magnetic torquer will be simulated in MATLAB as well as the magnetic field of the rods. The attitude model will be illustrated and equations for attitude models using only magnetic torquers and for attitude models using two magnetic torquers and one reaction wheel will be given. Also environmental effects called disturbance torques which influences the movement of the microsatellite and drive it away from its original attitude will be explained.

Formulation of the problem. Substantial progress concerning the knowledge, understanding and implementation of attitude actuators is necessary to meet the increasing number of spacecraft's. Due to relative cheapness and short period of the development especially small satellites became an important factor for space exploration and instrument testing. Therefore the precise stabilization of angular, position and attitude has to be secured. Today's active controlled micro- and picosatellites use magnetic coils or reaction wheals as capability of three-axis stabilizing.

Attitude determination and control systems ensure the attitude and movement of satellites using magnetic coils which generate a controlled torque in the roll and yaw axes while sensors as magnetometer and sun sensors measure data to determine the current position.

1. Attitude Determination and Control System (ADCS)

The ADCS determines the attitude of the spacecraft by evaluating sensor information's and generates commands for the actuators to make the spacecraft point in the right direction based on mission requirements concerning accuracy and slew rate. The general elements are illustrated in Figure 1.

Thus the ADCS can be dividing in two different main parts: attitude determination and attitude control. While determination refers to the process of measuring and determining spacecraft position and orientation, attitude control refers to the process of orienting the spacecraft in the required direction [3].

1.1. Attitude stabilization

The variety of common axis stabilization methods includes three axis control, spin stabilization, and gravity gradient systems.

Three axis control means the complete stabilization of the spacecraft's orientation along all three axes. This type of axis stabilization is more expensive and necessary for satellites with dynamic pointing requirements. This method requires closed loop control and is usually very computationally intense. Benefits are the capability of using the spacecraft autonomously tracks any arbitrary pointing requirement and a huge flexibility of maneuvers the spacecraft is capable of performing [4].

Spin stabilized spacecraft's uses the conservation of angular momentum to maintain a constant inertial orientation of one of its axes [1]. Therefore the control of the spacecraft arises by spinning it and controlling the orientation of the axis by rotation. It is simpler and a less expensive design than three axis stabilization, but forces the pay-

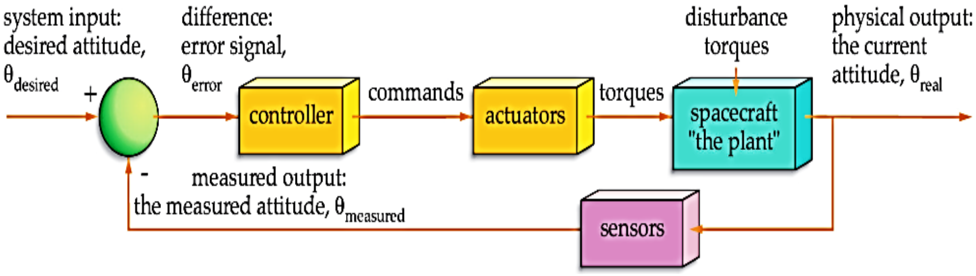

Fig. 1. General elements of an Attitude Determination and Control Subsystem (ADCS) [1]

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load to be in constant rotation [4]. Spin stabilization is useful, as long as we want our spacecraft to stay pointed in the same inertial direction [1].

Gravity gradient stabilization uses the gravity gradient disturbance torque in space which is a vector along the local geomagnetic field induction to keep the spacecraft oriented in local vertical or downward orientation. Usually this effect is maximized by deploying a small mass at the end of a very small bloom [4]. Gravity gradient stabilization offers a simple, reliable, inexhaustible capability of axis stabilization. The disadvantages are that this type of stabilization controls only two axes (pitch and roll); the accuracy is limited depending on the spacecraft's moments of inertia with a downward pointing accuracy of about ±10° and that this type is effective in low earth orbit because gravity varies with the square of the distance, it's not very effective beyond LEO [1].

1.2. Control method

Choosing the control method means to select one of the two common control methods as open loop and closed loop control.

Open loop control refers to a system that can't dynamically adjust the inputs based on what's actually happening. Thus the spacecraft's actuation mechanism is not autonomously driven by sensors and on-board control algorithm's but rather by commands from a ground station as turning to a desired direction. Open loop systems transfer the computational effort to the ground station and it is limited because commanding is only possible during the contact time with the ground station [4].

Closed loop control systems dynamical adjust inputs based on what is actually happening by sensor measurement. Thus the on board system alter the orientation and determine attitude and formulate corrective maneuvers. Compared to open loop control, it is more computationally intense [4].

1.3. Attitude determination

The performance of the attitude determination depends on the used sensors. Sensors are the essential element of closed-loop control systems which measure data based on what's happening in space to the system, describe the attitude in three dimensions and report this data to the control system. Common sensors for spacecraft attitude determination are sun sensors, star trackers, magnetometers, gyroscopes and GPS.

Sun sensors located the position of the sun and determine suns position with respect to the spacecraft body frame. Sun sensors give the azimuth and elevation of the sun vector and usually are combined with another sensor to describe the attitude in three dimensions. While simple sun sensor can provide 5° of pointing knowledge more complex instruments provide down to less than 1° [1].

Star trackers measure a spacecraft's attitude with respect to known star locations and compare these measurements to accurate maps of the brightest stars stored in the sensors data memory [1]. They are very precise, but also the most expensive attitude determination instrument [4].

Magnetometers measure the earth's magnetic field's direction and its strength local at the spacecraft and determine by a general model of earth's field the orientation of the spacecraft with respect to earth.

Gyroscopes determine spacecraft's attitude and changes of the attitude caused by principals of spinning mass. Every spinning mass has an conserved angular momentum while the gyroscope detect a spacecraft's angular motion [1].

GPS signals can also been tracked for determining the spacecraft's attitude. To implement a GPS system it needs usually two prime components as shown in Figure 2.

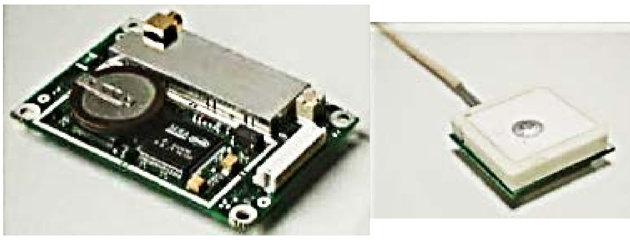

Fig. 2. The Phoenix GPS Receiver and the San Jose [5]

While the antenna receives the GPS signal and transmit it to the GPS receiver, the receiver convert and processes the signal into the instantaneous attitude information and forward the information to the ADCS unit [5]. This GPS system is in actual use for the Compass 1 picosatellite design by the University of Aachen, Germany.

1.4. Control systems

Actuators provide «torque on demand» to enable a rotation as needed to a specified position to meet mission requirements as taking pictures or data down- and uplink. Many different actuators are available and become often combined to apply accurately torque and achieve a desired attitude. Referring to the Federal Aviation Administration control systems conceptually can be divided into two general classes, passive and active attitude control systems [1].

Passive actuators operate in open loop systems and keep the spacecraft in the desired attitude. Active actuators require continuous feedback and adjustment thus they are able to act on command. Common passive actuators are for example gravity gradient stabilization and spin stabilization or dampers, which will not be further described because of the fact that for this ACDS active controlled actuators are requested and preferred. Common actuators are for example thrusters, reaction wheels and magnetic torquers.

Thrusters expel mass from the spacecraft into space pointing in one direction and therefore create a well defined torque. Usually more thrusters are arranged, pointing in opposite direction to enable a two or three axis attitude control. They are able to create a greater force or torque thus that they also can be used to change the spacecraft's orbit [4]. Unfortunately, the amount of fuel cause of weight and design reasons which a spacecraft can carry is limited. Therefore thrusters are used for short missions, but for longer missions thruster can be added as backup or temporary used as well depending on the mission [1].

Reaction wheels create torque in gaining an opposite torque by changing the spin rate of a flywheel. By increasing or decreasing the wheels speed the motors may apply a torque on the spacecraft in either direction about the axis of the wheel.

They are able of providing higher torques than magnetic torque rods, but they are more power intense, more expensive and prone to mechanical failure [4]. Typically, the ADCS uses at least three separate reaction wheels, oriented at right angles to each other for every axis. A forth wheel can be added for redundancy reason.

Magnetic torquers interact with earth's magnetic field and therefore are able to create controlled torque, which can adjust the attitude of a spacecraft by reversing the current in the rods or coils wires. Using magnetic torque is a simple and very common system and therefore used very often. Magnetic torquers are light, simple and low-power consuming. They use electrical power in order to generate controllable torques [3]. Magnetic torquers are combined with other attitude actuators as one or more reaction wheels, thus that all the actuator provide all the needed control to maintain the spacecraft's attitude, in low earth orbit up as in Geo Stationary orbit.

1.5. ADCS considerations

The high variety of ADCS elements enables many different combination possibilities while considering advantages and disadvantages of each component. After the selection process, the design layout of the ADCS can be described. First, a mathematical concept has to be established and after that the hardware can be produced in relation to the mathematical equations [5].

Due to the moment the component selection and description of the mathematical concept for this microsatellite is still in progress. Yet another topic of the microsatellites concept is the implementation and possible design of magnetic torquers. Therefore the design fundamentals will be considered in this paper.

2. Magnetic Torquer Design

In this paper the design of one type of magnetic torquer is considered. Magnetic rods consist out of many wires wound up together to a circle or square form. Magnetic rods as shown in Figure 3.

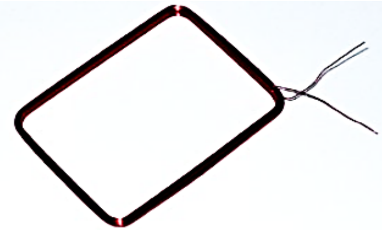

Fig. 3. Possible design magnetic rod [5]

The principle is to produce a controllable magnetic moment μ which interacts with the earth's magnetic field B_{earth} to produce torque. This torque is given by:

$$T = \mu x B_{earth};$$

The torque rods can only provide a torque relative to directions that are perpendicular to the local magnetic field vector [3]. The smallest magnetic field that earth's magnetic dipole generates above the equator for a specific altitude is given by:

$$B_{earth} = \frac{B_0}{R^3};$$

where $B_0 \approx 3 \times 10^{-5} T$ and with h referring to the altitude and R^3 referring to the earth radius which is 6350 km [3]. The necessary control torque to counteract disturbance torques at an altitude of 400 km would be as an example $10^{-6}\ Nm$. At that altitude B_{earth} would be $25 \mu T$. Therefore the magnetic moment gained by the magnetic torquers has to be $5 \times 10^{-2}\ Am^2$. Thus the aim is to generate control torques around this range.

2.1. General design requirements

General requirement for the torquer design are described in this chapter. The required satellite should be able to stabilize all three axes thus a minimum of three torquers should be considered overall. The magnetic moment will be calculated with MATLAB in this paper for one torque, furthermore all torquers should be simulated in one program to be able to illustrate the interaction with earth's magnetic field. The torquers are not used constantly due to the limited available power in space. Therefore as a first approach for the calculations the power budget for one torque should not exceed 500 mW and will be limited beginning with 100 mW. The maximum weight for the coils is not specified at the moment. Therefore the range of the calculation starts from 20 g and is limited to 50 g for each coil. The available voltage is considered to be 3 V and the length of a possible square coil side s is given by the microsatellites geometry of 400 mm^3.

The optimum is to obtain the best required magnetic moment with the smallest volume of wire by the given requirements.

2.2. Magnetic torque rods

A magnetic torque rod is a long metallic wire, wound up around a ferromagnetic or without ferromagnetic core. Magnetic rods with a ferromagnetic core generate a larger magnetic dipole moment with less room and lower power consumptions. In this paper magnetic torque rods with ferromagnetic core are not considered to meet the microsatellite weight requirements. Therefore the advantages of magnetic torque rods without ferromagnetic core are the low mass, its simplicity and that magnetic moments can accurate and easily adjusted. The magnetic moment is given by:

$$\mu = NIAn;$$

where n is the unit vector perpendicular to the loop, N is the number of turns, I is the current applied in the rod and A is the area of the coil plane [3].

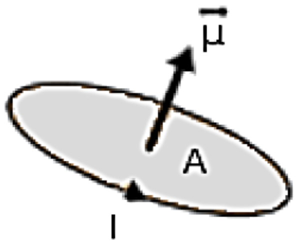

Fig. 4. Magnetic moment created by a single loop of wire [3]

Furthermore it is assumed that lw and Aw are the length and the cross-sectional area of the wire, with a square coil side length s [3]. Considering that geometry, the number of turns is given by:

$$N = \frac{lw}{4 * s};$$

Therefore the magnetic moment can be described as:

$$\mu = l_w A_w J \frac{S}{4};$$

while J is the given current density. The power consumption can be calculated by:

$$P = UI = UJA_w;$$

The power consumption can be optimized under the constraint that the accepted current density of copper on earth is about 5 A/mm^2. Therefore it is assumed an current density of $J = 1\ A/mm^2$ [3]. The resistance of the wire is given by:

$$R_w = \frac{\rho_e l_w}{A_w} = \frac{\rho_e V_w}{A_w^2};$$

where ρ_e is the electrical resistivity. Due to practical considerations a resistance must be added in series [3].

$$R_{series} = \frac{V^2}{P} - R_w;$$

Concerning to the general design requirements further assumptions are made for the following calculations. The electrical resistivity of copper is $\rho_e = 17.2\ \Omega m$ and the mass density of copper is $\rho_d = 8.2$ kg/dm^3. Considering the available voltage is fixed to 3 V, the power consumption decreases with the cross-sectional area A_w [3]. To meet the power budget of 100 mW A_w is considered to be 0.033 mm^2. Possible magnetic torques and control torques at an altitude of 400 km are calculate in MATLAB and shown in Figure 5.

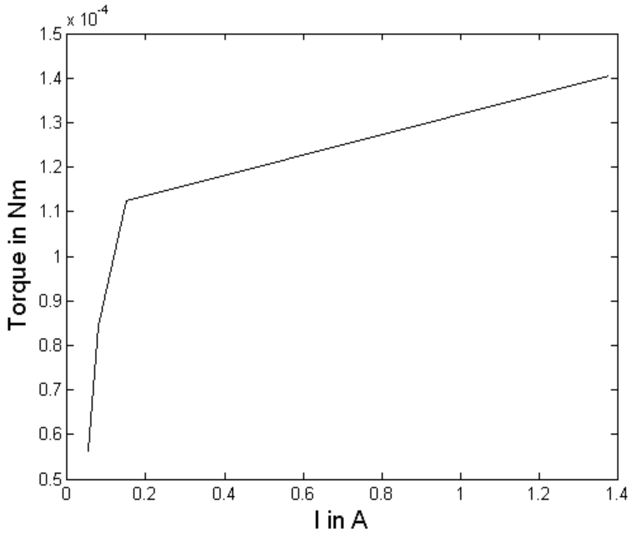

Fig. 5. Control torque by magnetic torque rods without ferromagnetic core

Referring to Figure 5 the possibly achievable control torque increases proportional to a rising electricity induction by the given design requirements in the calculated torque rod mass range from 20 g to 50 g per rod. Therefore the maximum obtainable magnetic moment is proportional to the mass of the rod and reaches due a mass of 50 g a control torque of $1.4 * 10^{-4}\ Nm$.

To illustrate the magnetic moment of the designed rod magnetic field to the magnitude, direction, length, and proximity by the calculated electric current a MATLAB simulation of the Biot-Savart law is considered. The Biot-Savart law describes the magnetic field generated by an electric current and is given by:

$$B = \frac{\mu_0}{4\pi} I \int_C \frac{dl \times \hat{r}}{r^2};$$

In this simulation the square loop is in the X and Y plane and magnetic field is evaluated at every point in the Y and Z plane while is $X = 0$. Magnetic field formation for the calculated magnetic rod as shown in Figure 6.

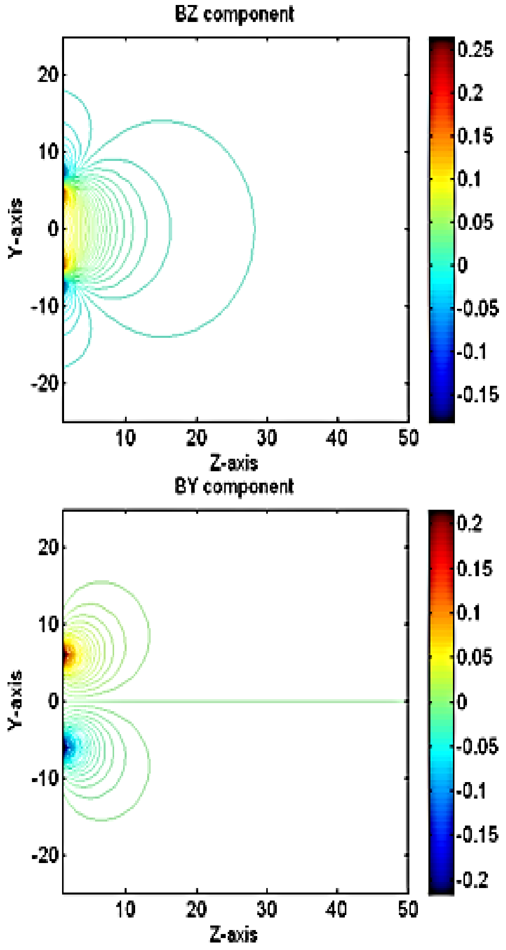

Fig. 6. Magnetic field formation for the calculated magnetic rod

3. Attitude Model

The attitude of a spacecraft describes the orientation of a body fixed coordinate frame referring to an external frame as shown in Figure 7. The body fixed coordinate frame (called body frame) is usually given by roll, pitch, and yaw angles, where roll is a rotation about the X axis, pitch is a rotation about the Y axis, and yaw is a rotation about the Z axis.

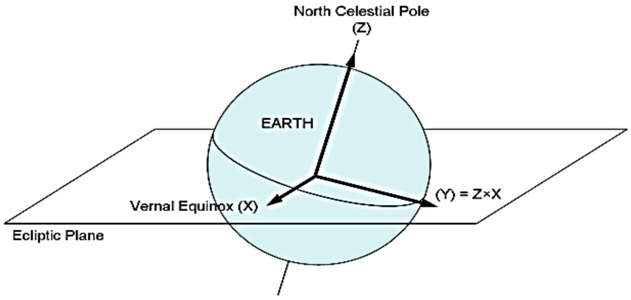

Fig. 7. Illustration of the Earth Centered Inertial coordinate frame [4]

The Orientation of 3-D Space objects can be described in several ways. The common method to describe the orientation of a spacecraft uses Euler angles and quaternions. Euler angles describe the orientation of a set of body fixed axes relying to an inertial reference frame. The Euler angles of any orientation can be described as three separate angles φ, θ, and ψ, that act to rotate the body along the body fixed axes from the inertial axes as shown in Figure 8.

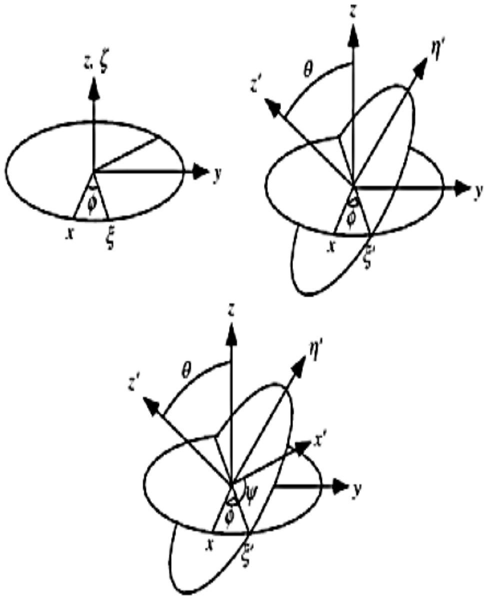

Fig. 8. Euler angles [2]

Most of the algorithms shown in the considered literature assume the application of reaction wheels with or without the use of thrusters for three axis stabilization.

Thus it is mentioned that attitude models with only magnetic torquers challenges the control of the system to be not entire controllable. Therefore one attitude model with only magnetic torquers and one including reaction weels will be described in the following chapters.

3.1. Attitude model with only magnetic torquers

The control torque is constrained to the plane perpendicular of the local geomagnetic field. This always leaves one uncontrollable direction parallel to the local field. Thus attitude stabilization is possible for two axes and in higher orbits even for a three axis stabilization while the magnetic field direction in an inertial space as seen from the spacecraft is varying during the orbit. Therefore orbits with low inclinations can lead to less efficient control since the magnetic field direction is not varying much [3]. The attitude dynamics can be expressed by using the Euler's equations with gives the following equation:

$$I\frac{d\omega}{dt} = -\omega \times I\omega + T_{coils} + T_{dist};$$

This basic control law could be used to define the ideal control torque for a 3 axis stabilization control depending on the strengths of earth's magnetic field [3]. Referring to the literature attitude maneuvers can be simulated and described by using parameters as shown in Figure 9.

3.2. Attitude model of magnetic torquers with one reaction wheel

To be able to include one reaction wheel in the considered attitude model for a desired torque the generated magnetic moment vector needs to be implemented which can be described as:

$$\mu = [\mu_x, \mu_{xy}, \mu_z]^T;$$

In order to get full controllability one magnetic torquer within the magnetic torque matrix should be replaced by a reaction wheel (referring to description in reference [3]). If a reaction wheel with its axis aligned along the Z body axis, the control equations becomes the following:

$$\begin{pmatrix} T_x \\ T_y \\ T_z \end{pmatrix} = \begin{pmatrix} 0 & B_z & 0 \\ -B_z & 0 & 0 \\ B_y & -B_x & 1 \end{pmatrix} \begin{pmatrix} \mu_x \\ \mu_y \\ h_w \end{pmatrix};$$

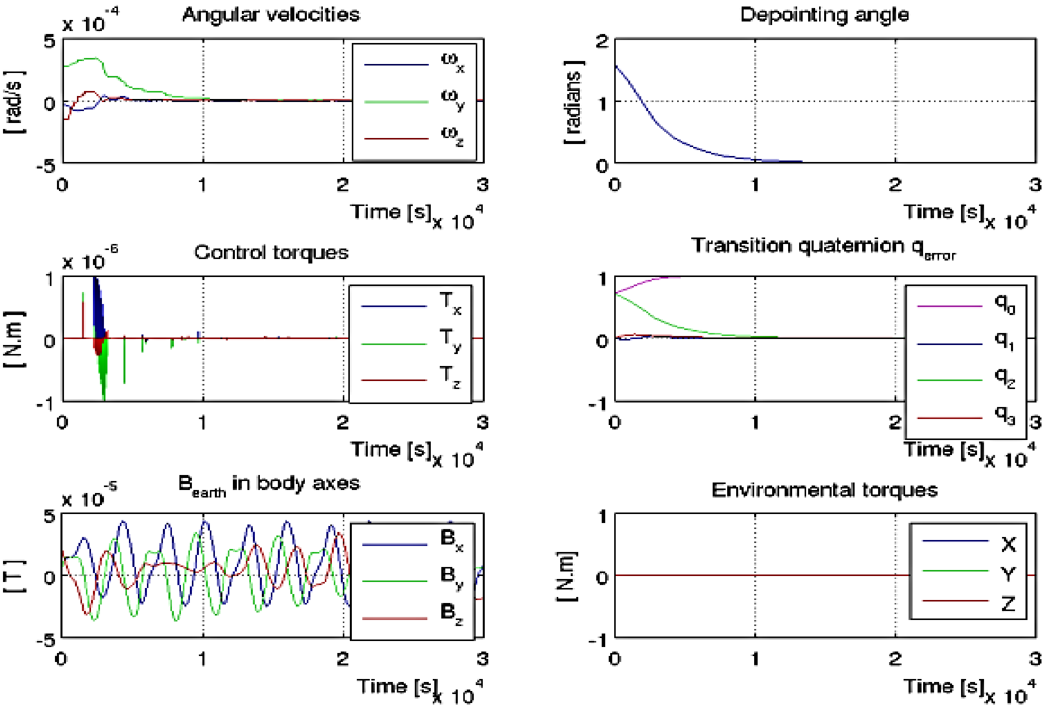

Fig. 9. Attitude maneuver using only magnetic torquers, no disturbance torques considered [3]

Table 1

Spacecraft affecting disturbance torquers [4]

Disturbance Torque	Equation for Magnitude	Estimated Torque, n · m	Comment
Magnetic Field	$\tau_m = DB$; D = spacecraft magnetic dipole = $5\ A \times m^2$; B = Earth's magnetic field local to spacecraft = $2.58 \times 10^{-5}\ T$.	1.3×10^{-4}	Dipole generated by active control of magnetictorque rods. Magnetic field strength at 350 *km*.
Nutation Damper	$\tau_d = k_d \omega_{\omega/b}$; k_d = damping constant = 0.001; $\omega_{\omega/b}$ = rate of fluid bulk moving relative to container walls.	1.2×10^{-4}	Assumes 8° half-cone of nutation at 10 RPM. k_d tuned to match required 15 *hr* time constant.
Atmospheric Drag	$\tau_a = \frac{1}{2}(\rho C_d A_V V^2)(c_{pa} - cg)$; ρ = local atmospheric density = $1.50 \times 10^{-11}\ kg/m^3$; C_d = coefficient of drag = 2; A_V = cross sectional surface area in ram direction = $0.164\ m^2$; V = spacecraft velocity = $7697\ m/s$; $c_{pa} - cg$ = distance from aerodynamic center of pressure to center of gravity = $0.01\ m$.	1.3×10^{-6}	Assumes predicted atmospheric activity at 350 *km* during solar max.
Gravity Gradient	$\tau_g = \frac{3\mu}{2R^3} \lvert I_x - I_y \rvert \sin 2\theta$; μ = Earth's gravity constant = $3.986 \times 10^{14}\ m^3/s^2$; R = orbit radius = $6.728.140\ m$; $I_x - I_y$ = largest and smallest spacecraft moments of inertia; θ = maximum deviation of I_x from nadir or zenith = $\frac{\pi}{2}$ radians.	1.3×10^{-7}	Assumes a maximum $I_x - I_y$ of 16 $kg \cdot m^2$
Solar Pressure	$\tau_{sp} = \frac{F_s}{c} A_s (1 + q) \cos i\, (c_{pa} - cg)$; F_s = solar constant = $1.367\ W/m^2$; c = speed of light; A_s = cross sectional area in direction of the sun = $0.164\ m^2$; q = reflectance factor = 1; i = angle of incidence to the sun = 0 radians; $c_{pa} - cg$ = distance from solar center of pressure to center of gravity = 1 *cm*.	1.5×10^{-4}	Assumes worst-case reflectance $q = 1$ and $c_{ps} - cg$ same as in atmospheric drag calculation

This is a possible solution of an attitude model with one reaction wheel which improves the momentarily controllability, and would have the advantage of being lighter and cheaper than an ADCS with three reaction wheels [3]. These considerations can be useful for future types of attitude control with more explicit requirements.

4. Disrurbance Torques

Spacecraft's are affected by different disturbance torques in an earth orbit, which the attitude control system must either tolerate or manage and have to be included to attitude model simulations. These torques are gravity gradient, solar radiation pressure, magnetic field effects, and aerodynamic forces. The most significant ones are listed and quantized in the following Table 1.

Resume and further prospects. This paper showed and described considered elements during the feasibility studies for a microsatellite developed by «Igor Sikorsky Kyiv Polytechnic Institute – Faculty of Aircraft and Space Systems» concerning a possible ADCS. This paper comprised three objectives. The first objective was to verify and to characterize possible elements of the ADCS. The second objective was to establish a possible preparation of the magnetic torquers including the assumed design requirements and application aspects. And the third objective was to show further design aspects of the magnetic coils related to environmental effects called disturbance torques which influences the movement.

Perspectival the mathematical model of the ADCS should be completed including the equations for the magnetic rods and test hardware should be evaluated. Concerning the magnetic rods the temperature drift of the wire resistance and the material should be included to the simulation as well as to research for the feasibility of using another wire material then copper as for example silver.

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СИСТЕМИ ОРІЄНТАЦІЇ ТА НАВІГАЦІЇ МІКРОСУПУТНИКА З ВИКОРИСТАННЯМ МАГНІТНИХ КОТУШОК

Анотація
Дана стаття присвячена розробці системи орієнтації та навігації мікросупутника з використанням магнітних котушок. В роботі розглянуті елементи необхідні для створення і моделювання системи управління мікросупутника по трьох осях. Проведено математичне моделювання роботи магнітних котушок і системи управління мікросупутника в середовищі MATLAB. На основі результатів моделювання описаний метод стабілізації по трьох осях і робота датчиків системи орієнтації, навігації та керування. Також описуються збурюючі моменти від впливу навколишнього середовища, які впливають на рух мікросупутника і зміщують його від початкового положення.
Ключові слова: управління орієнтацією супутника, супутник, визначення просторової орієнтації, система управління, магнітний момент.

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СИСТЕМЫ ОРИЕНТАЦИИ И НАВИГАЦИИ МИКРОСПУТНИКОВ С ИСПОЛЬЗОВАНИЕМ МАГНИТНЫХ КАТУШЕК

Аннотация
Данная статья посвящена разработке системы ориентации и навигации микроспутника с использованием магнитных катушек. В работе рассмотрены элементы необходимые для создания и моделирования системы управления микроспутника по трем осям. Проведено математическое моделирование работы магнитных катушек и системы управления микроспутника в среде MATLAB. На основе результатов моделирования описан метод стабилизации по трем осям и работа датчиков системы ориентации, навигации и управления. Также описываются возмущающие моменты от воздействия окружающей среды, которые влияют на движение микроспутника и смещают его от первоначального положения.
Ключевые слова: управления ориентацией спутника, спутник, определения пространственной ориентации, система управления, магнитный момент.