Attitude Determination and Control System for Palamede Microsatellite

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Abstract

In this paper the attitude determination and control system for Palamede microsatellite is discussed. Palamede is a microsatellite studied and developed at Politecnico di Milano, and it is currently in the assembly phase. The mission has two main goals: test a new type of triple junction solar cell and capture images of the Earth. A coarse sun sensor, one three-axis magnetometer and three magneto torquers constitute the hardware of the ADCS. The coarse sun sensor is constituted by six dedicated small solar cells, placed on the six side panels, while the magnetometer is a commercial one. With the available sensors the attitude of the satellite must be determined in different ways according to the different sunlight conditions. In the sunlit portion of the orbit there are two available options, namely a conventional three axis attitude determination algorithm and a Kalman filter based algorithm, while in the eclipse portion of the orbit only the Kalman filter based algorithm can provide a good attitude estimation. For what concerns the attitude control, two options are considered. The first option is a pure detumbling control based on the rate of change of the measured magnetic field, while the second option is a single-axis control of the nadir pointing, that allows to point the camera axis toward the Earth. The paper presents the design of the different control laws and the expected performances, validated in a simulated scenario.

Introduction to Palamede microsatellite

Palamede is a microsatellite designed by students of Aerospace Engineering Department of Politecnico di Milano [1]. It will be launched into a sun-synchronous low earth orbit. The main scientific purposes are observing the Earth with a CCD camera and testing a new type of triple junction solar cell. It is built using both terrestrial and space-qualified components, preferring the first one if possible. As a consequence, a secondary purpose is testing the possibility of using common terrestrial components in low-cost space applications. In addition, this project is an important and exciting training experience for involved students. The earlier studies carried out in this context analysed a wide range of solutions for the design of a microsatellite. These translate into different possible orbits, ranging from GTO to LEO, corresponding to different kind of missions. After this preliminary study, the baseline selected for this first mission was to put a satellite into a sun-synchronous low earth orbit with the objective to take pictures of the Earth. Anyway, since at this moment a launch opportunity has not yet been definitely selected, the design process has been carried out in a way to obtain a spacecraft extremely tolerant to...
orbital variation. In this way we will be able to gather different launch opportunities, of course as secondary payloads, carrying the spacecraft to an orbit inclined more than 30° and 450 - 1000 km high. At this stage of the project, the orbit decided is a sun-synchronous one, at an altitude of 650 Km and an inclination of 98°. The launcher foreseen at the moment to carry Palamede to space is the Dnepr. The main spacecraft features are the following (see fig. 1):

− aluminium alloy structure, 400x400x400mm, for a total mass below 30kg;
− electrical power system composed of 5 solar arrays made by triple junction cells, a Li-Ion battery, a battery charge regulator and a DC-DC converter;
− ADCS: nadir pointing stabilised with 1 magnetometer, 6 sun sensors, 3 magneto-torquers;
− PC-104 standard electronics, not redundant, shielded by aluminium box;
− passive thermal control.

Figure 1: Layout of Palamede microsatellite.

Environment and requirements for ADCS system

The design of the satellite has considered a wide set of orbital parameters, but due to the compact shape and small size the disturbance torques are relatively small in all conditions analysed. The greatest disturbance should come directly from the residual magnetic moment of the magnetic torquers. Numerical simulations of the nominal mission have allowed to estimate a gravity gradient and solar radiation torque in the order of $10^{-7}$ Nm, an aerodynamic torque in the order of $10^{-8}$ Nm, and a magnetic disturbance torque in the order of $10^{-5}$ Nm. The requirements for the nominal attitude pointing are not very stringent, as common for this class of small
satellites [2]. In order to allow a satisfactory image acquisition the residual angular velocities must be lower than 2 deg/sec.

**ADCS layout**

The attitude determination and control system relies on two sensors for attitude determination, a magnetometer and a coarse Sun sensor, and on three magnetic torquers for attitude control. The magnetic torquers are manufactured by Zarm Technik, and are capable of providing a linear dipole moment up to 15 Am$^2$, at a supply voltage of 14 V and input current of 0.08 A, with a power of 1.12 W. They are space qualified and perfectly fit into the structure of the satellite, having an overall length of 0.33 m. For the purpose of torque control, they will be considered perfectly linear devices with saturation, as depicted in Figure 2. The magnetometer is a standard three axis fluxgate magnetometer, with a nominal sensitivity of 12 nT/mV per axis and a measurement range of ±60000nT. It is the second space qualified component on board, and its calibration function is the following:

\[
\begin{bmatrix}
    B_x \\
    B_y \\
    B_z
\end{bmatrix}
= 
\begin{bmatrix}
    12.048 & 0.135 & 0.067 \\
    -0.112 & 12.063 & -0.126 \\
    -0.064 & 0.178 & 12.040
\end{bmatrix}
\begin{bmatrix}
    V_x \\
    V_y \\
    V_z
\end{bmatrix}
- 
\begin{bmatrix}
    53.305 \\
    42.233 \\
    60.760
\end{bmatrix}
\]

\[
B_{xyz} = CV_{xyz} - \text{Offset}
\]

where $B$ represents the measured magnetic field and $V$ the sensor output. Deviations from the linear calibration function are lower than 0.05%.

The greatest disturbance on the sensor output is given by the magnetic field generated by the actuators. In case the sensor and the actuator are operating at the same instant, to infer the correct value of the Earths’ magnetic field, the magnetic field generated by the actuator at the location of the sensor has to estimated and properly taken into account. A model of the magnetic field generated by the actuator, at distances greater than the actuators’ length, is given by

\[
\begin{bmatrix}
    B_r \\
    B_t
\end{bmatrix}
= \left(\frac{\mu_0}{4\pi}\right) \cdot M \cdot \frac{R/L - \cos(\vartheta)/2}{\left[R^2 - R L \cos(\vartheta) + L^2/4\right]^{3/2} \sin(\vartheta)/2} - \frac{R/L - \cos(\vartheta)/2}{\left[R^2 + R L \cos(\vartheta) + L^2/4\right]^{3/2} \sin(\vartheta)/2}
\]

where $\mu_0$ is the magnetic permeability, $L$ represents the actuators’ length, $R$ the position of the sensor, and $M$ the magnetic dipole generated by the actuator (see figure 3).
The required measurement, i.e., the Earths’ magnetic field components along the principal inertia axis, is then provided by the following equation, in which $A$ represents a correction matrix, that takes into account the small relative rotation between principal inertia and geometrical axes of the satellite:

$$B_{pi} = A(C \cdot V_{xyz} - Offset - D \cdot M_{xyz})$$

(4)

The coarse Sun sensor is composed by 6 single junction solar cells, body mounted, one for each outer panel of the satellite. The size of each cell is 0.02x0.02 m. Each cell can be illuminated by a combination of Sun radiation or Earth radiation. Since the cells are located on opposite sides of a cube, the algorithm will consider always pairs of cells as one single entity. With reference to Figure 3, the outputs of Cell 1 and Cell 6, and Cell 2 and Cell 5, will be analyzed together. The position of the Sun is evaluated in three steps:

1) Detection of the Sun illuminated cells. Analyzing the output of opposite cells, the following situations are possible, with reference to Figure 4:
- $I_1 \approx I_6 \approx 0$, then either the satellite is in shadow or both sides of the satellite are parallel to the Sun radiation;
- the above considerations are repeated for Cells 2 and 5 and for Cells 3 and 4.

2) Eclipse detection. Whenever the previous step indicates that no Cell is directly illuminated by the Sun, the satellite is considered in eclipse and the Sun position is not evaluated.

3) Evaluation of the Sun position. Assuming the general case in which three Cells are illuminated by the Sun, for each the cosine law can be assumed valid and the Sun direction $R$, in the geometrical reference frame, can be determined as

$$R = \frac{I_1}{I_{max}} \cdot i + \frac{I_2}{I_{max}} \cdot j + \frac{I_3}{I_{max}} \cdot k = \cos(\alpha) \cdot \pm i + \cos(\beta) \cdot \pm j + \cos(\gamma) \cdot \pm k$$

(5)
where \( I_{\text{max}} \) is the current output in the case of Sun radiation perpendicular to the cell. The ± sign in equation 5 takes into account the case in which \( I_1 \) is replaced by \( I_6 \), or \( I_2 \) by \( I_5 \), or \( I_3 \) by \( I_4 \).

The real current output of a solar cell is not a pure sinusoidal function, nevertheless it has been estimated that the above procedure detects the Sun position with a maximum error below 5 degrees in the worst condition, and in the majority of cases below 0.5 degrees, that is considered acceptable for the current application.

**Attitude determination**

Two different algorithms have been designed for the attitude determination process on board Palamede microsatellite: an algebraic method, that can be used whenever both Sun position and magnetic field vector measurements are available, and a Kalman filter that can operate also when the Sun is not detectable. In any case, a correct knowledge of the position and the orbit of the satellite is necessary, so the first step in the attitude determination must be the propagation of the orbit, starting from two measurements available form the GPS receiver. In fact, due to power limitations, the GPS receiver can not operate continuously, therefore a numerical method integration scheme has been designed to compute the orbital parameters and the satellites’ position at any time, updated whenever a new GPS position fix is available. The overall scheme of the process is shown in Figure 5.

**Orbit determination and propagation**

Given position \( r \) and velocity \( v \), provided by the GPS receiver, it is possible to compute the orbit inclination and the direction of the line of nodes as [3]

\[
    h = r \Delta v \quad ; \quad i = \cos^{-1} \left( \frac{k \cdot h}{|h|} \right) \quad ; \quad n = \frac{k \wedge h}{|k \wedge h|} \quad (6)
\]
where $k$ indicates the direction of the Earth’s rotation axis. It is now possible to infer the right ascension $\Omega$ and the argument $u$, coincident with the true anomaly $\alpha$, since the orbit is considered circular and the argument of perigee $\omega$ is set to zero.

$$\cos \Omega = i \cdot n \Rightarrow \Omega = \begin{cases} \cos^{-1}(i \cdot n) & \text{if } n \cdot j > 0 \\ 360^\circ - \cos^{-1}(i \cdot n) & \text{if } n \cdot j < 0 \end{cases}$$

$$\cos u = \frac{n \cdot r}{|r|} \Rightarrow u = \begin{cases} \cos^{-1}\left(\frac{n \cdot r}{|r|}\right) & \text{if } r \cdot k > 0 \\ 360^\circ - \cos^{-1}\left(\frac{n \cdot r}{|r|}\right) & \text{if } r \cdot k < 0 \end{cases}$$

where $i$ is the direction of the vernal equinox and $j$ is perpendicular to $i$ and lies in the equatorial plane.

Once the orbit has been determined at the time instant in which the GPS receiver is switched off, it is necessary to propagate the position of the satellite in order to allow a correct estimation of the relative position between Sun, Earth and satellite, and also the estimated value of the magnetic field for attitude determination. Orbit propagation is performed considering only perturbations due to the $J_2$ term in the gravitational potential function, and the consequent only orbital parameter variation is in the longitude of the ascending node. This is considered precise enough since it is estimated to fix the position with the GPS receiver once per orbit. Therefore, at any given time instant, the right ascension and the true anomaly are evaluated as

$$\frac{d\Omega}{dt} = -\frac{3}{2} J_2 \left(\frac{r_{eq}}{p}\right)^2 n \cos(i) = 2.1493 \times 10^{-7} \text{ (rad/s)}$$

$$\Omega_i = \Omega_{i-1} + \frac{d\Omega}{dt} (t_i - t_{i-1})$$

$$\alpha_i = \alpha_{i-1} + n \cdot (t_i - t_{i-1})$$

where $n=1.1068\times 10^{-3}$ rad/s represents the nominal orbital angular velocity. The position of the satellite, expressed in geocentric inertial coordinates, is then given by

$$\begin{align*}
R_x &= |R| \cdot \left[\cos(\omega + \alpha) \cdot \cos(\Omega) - \sin(\omega + \alpha) \cdot \sin(\Omega) \cdot \cos(i)\right] \\
R_y &= |R| \cdot \left[\cos(\omega + \alpha) \cdot \sin(\Omega) + \sin(\omega + \alpha) \cdot \cos(\Omega) \cdot \cos(i)\right] \\
R_z &= |R| \cdot \left[\sin(\omega + \alpha) \cdot \sin(i)\right]
\end{align*}$$

Given the Cartesian coordinates of the satellite, its latitude and longitude can be evaluated in order to predict the magnetic field vector, using a standard IGRF model. Calling $\alpha_G$ the position of the Greenwich meridian and $\alpha_E$ the angular velocity of the Earth, the east longitude $\phi$ and latitude $\delta$ become
\[
\alpha_G = \alpha_G^0 + \omega_E \cdot t ; \quad \phi = \tan^{-1}\left(\frac{R_y}{R_x}\right) - \alpha_G ; \quad \delta = \tan^{-1}\left(\frac{R_z}{\sqrt{R_x^2 + R_y^2}}\right)
\]  

(11)

This simple propagation of the satellites’ position leads inevitably to errors in the estimation of latitude and longitude, that in turn lead to errors in the estimation of the magnetic field, but nevertheless these errors are considered acceptable for the present application. Figure 6 shows the entity of the errors for a period of one day. The “true” position of the satellite is evaluated using the HPOP integrator available in STK, while the estimated position is evaluated adopting the above described procedure. It must be considered that in normal operation it is expected to reset the position error once per orbit, and in such case the errors are really negligible.

![Figure 6: errors induced by the orbit propagation algorithm.](image)

**Algebraic attitude determination algorithm**

This algorithm can be used only when the satellite is not in the shadow, but has the great advantage of simplicity and extremely low computational requirements. It is remarked that the mission profile requires to control the attitude only when the Sun is visible, so the limitations imposed by the algorithm are minor. The algebraic attitude determination algorithm is the classical TRIAD algorithm [4], implemented on the basis of the two available measurement vectors. Calling \( p \) the direction of measured magnetic field vector, and \( q \) the measured Sun direction, \( a \) and \( b \) the corresponding model unit vectors, expressed in the orbital reference frame, the
algorithm to determine the rotation matrix $A$, representing the attitude of the satellite in the orbital frame, is the following

$$ s_1 = p \quad s_2 = \frac{p\Lambda q}{|p\Lambda q|} \quad s_3 = p\Lambda s_2 \quad v_1 = a \quad v_2 = \frac{a\Lambda b}{|a\Lambda b|} \quad v_3 = a\Lambda v_2 \quad (12) $$

$$ [s_1 \quad s_2 \quad s_3] = A[v_1 \quad v_2 \quad v_3] \Rightarrow S = AV \Rightarrow A = SV^T \quad (13) $$

Figure 7 reports the performances of the algorithm, in a simulated scenario that considers the Sun always visible. It can be noticed that, as expected, along some limited portions of the orbit the estimation error is significant, due to the poor angular separation of the two measurements.

![Figure 7: performances of the algebraic attitude determination algorithm](image)

**Kalman filter attitude determination algorithm**

The Kalman filter designed for attitude determination is a standard linearized Kalman filter, also known as Extended Kalman Filter (EKF) [2,5,6]. It considers a nonlinear model of the system dynamics, with state vector including quaternion and angular velocity, and measurement vector that can either include only the magnetic field vector or both measurements, magnetic field and Sun direction. The quaternion that is part of the state vector represents the rotation between the principal inertia frame and the orbital frame. The algorithm performs the state estimation in eight steps:
1) Forward integration of the system dynamics, to evaluate the predicted state vector

\[ \dot{x}(t) = f(x(t), u(t), t) + w(t) \quad \text{with} \quad x(t) = \{ q(t) \} \Rightarrow \bar{x}_k \]

The system dynamics are represented in the standard form, where \( M_{MT} \) represents the magnetic control torque and \( M_D \) the external disturbance torque vector

\[ \begin{cases} I\dot{\omega} = M_{MT} + M_D - \omega \Lambda \omega \\ \dot{q} = \frac{1}{2} \Omega(\omega)q = \frac{1}{2} \Lambda(q)\omega \end{cases} \]

2) Linearization of the system dynamics

\[ \Delta x(t) = F(t) \Delta x(t) + w(t) \quad \text{with} \quad F = \begin{bmatrix} F_{11} & F_{12} \\ 0 & F_{22} \end{bmatrix} \]

\[ F_{11} = \frac{1}{2} \begin{bmatrix} 0 & \omega_z & -\omega_y & \omega_x \\ -\omega_z & 0 & \omega_x & \omega_y \\ -\omega_y & -\omega_x & 0 & \omega_z \\ -\omega_x & -\omega_y & -\omega_z & 0 \end{bmatrix}, \quad F_{12} = \frac{1}{2} \begin{bmatrix} q_4 & -q_3 & q_2 \\ q_3 & q_4 & -q_1 \\ -q_2 & q_1 & q_4 \\ -q_1 & -q_2 & -q_3 \end{bmatrix} \]

\[ F_{22} = \begin{bmatrix} 0 & (I_y-I_z)\omega_y & (I_y-I_z)\omega_z \\ (I_z-I_x)\omega_z & I_x & (I_z-I_x)\omega_x \\ (I_x-I_y)\omega_y & (I_x-I_y)\omega_x & I_y \end{bmatrix} \]

3) Evaluation of the state transition matrix \( \Phi_k \)

\[ \Phi_k = e^{F\Delta t} = \left[ I_{7\times7} + \begin{bmatrix} F_{11} & F_{12} \\ 0 & F_{22} \end{bmatrix} \Delta t \right] \]

4) Evaluation of the predicted state covariance matrix, where \( Q \) represents the process noise covariance matrix

\[ P_{k+1}^- = \Phi_k P_k \Phi_k^T + Q \]

5) Measurement update and evaluation of the residual \( b_k \).
Calling \( B \) and \( S \) the measured magnetic field and Sun direction vectors, the linearization of the measurement function \( h(x) \) depends on the availability of either measurement, in some cases the Sun will not be visible, but in any case no dependency on the angular velocity will be present.
\[ z_k = \begin{bmatrix} B_{pi} \\ S_{pi} \end{bmatrix} \quad \text{and} \quad H_k = \frac{\partial h(x)}{\partial x} \bigg|_{x=\hat{x}_k} \quad , \quad H_k = \begin{bmatrix} H_B \\ H_S \end{bmatrix} \bigg|_{x=\hat{x}_k} \] (20)

\[ H_B = \frac{\partial B_{pi}}{\partial x} = \begin{bmatrix} \frac{\partial A}{\partial q_1} B_{orb} \\ \frac{\partial A}{\partial q_2} B_{orb} \\ \frac{\partial A}{\partial q_3} B_{orb} \\ \frac{\partial A}{\partial q_4} B_{orb} \end{bmatrix} \begin{bmatrix} 0 \end{bmatrix} \] (21)

\[ H_S = \frac{\partial S_{pi}}{\partial x} = \begin{bmatrix} \frac{\partial A}{\partial q_1} S_{orb} \\ \frac{\partial A}{\partial q_2} S_{orb} \\ \frac{\partial A}{\partial q_3} S_{orb} \\ \frac{\partial A}{\partial q_4} S_{orb} \end{bmatrix} \begin{bmatrix} 0 \end{bmatrix} \] (22)

\[ \frac{\partial A}{\partial q_1} = 2 \begin{bmatrix} q_1 & q_2 & q_3 \\ q_2 & -q_1 & q_4 \\ q_3 & -q_4 & -q_1 \end{bmatrix} \quad \frac{\partial A}{\partial q_2} = 2 \begin{bmatrix} -q_2 & q_1 & -q_4 \\ q_1 & q_2 & q_3 \\ q_4 & q_3 & -q_2 \end{bmatrix} \] (23)

\[ \frac{\partial A}{\partial q_3} = 2 \begin{bmatrix} -q_3 & q_4 & q_1 \\ -q_4 & -q_3 & q_2 \\ q_1 & q_2 & q_3 \end{bmatrix} \quad \frac{\partial A}{\partial q_4} = 2 \begin{bmatrix} q_4 & q_3 & -q_2 \\ -q_3 & q_4 & q_1 \\ q_2 & -q_1 & q_4 \end{bmatrix} \] (24)

If only the magnetic field vector measurement is available, then \( H = H_B \).

6) Evaluation of the filter gain

\[ K_k = P_k^- H_k^T \left( H_k P_k^- H_k^T + R_k \right)^{-1} \] (25)

7) Error estimation and state prediction update

\[ \Delta \hat{x}_k = K_k b_k \quad \Rightarrow \quad \hat{x}_k = \hat{x}_k + \Delta \hat{x}_k \] (26)

8) Update of state vector covariance

\[ P_k = (I_{7\times7} - K_k H_k) P_k^- \] (27)

The performances of the Kalman filter are strictly dependent on the initial state assigned and on the appropriate definition of the process noise covariance matrix \( Q \) and the initial state covariance matrix \( P_0 \). To overcome the first problem, it has been decided to switch the filter on only when two measurements are available, therefore enabling a first guess estimation of the quaternion adopting the algebraic algorithm, and estimating a first guess of the angular velocity on the basis of the rate of change of the magnetic field vector. It has been noticed, from several numerical simulations, that there are no convergence problems only if the angular velocity is small, therefore a further constraint on the activation of the filter is a threshold on the angular velocity, above which the filter is not activated. The process noise and the initial state covariance matrices have been tuned also by numerical simulations, and the optimal values have been found as
\begin{equation}
P_0 = \text{diag}(1e3 \ 1e3 \ 1e3 \ 1e3 \ 100 \ 100 \ 100) \tag{28}
\end{equation}

\begin{equation}
Q = \text{diag}(100 \ 100 \ 100 \ 100 \ 0.1 \ 0.1 \ 0.1)
\end{equation}

Figures 8 and 9 report one of the results obtained in a simulated environment, considering the periodical presence of eclipse during which only one measurement is available. It appears that the behaviour of the filter is satisfactory, with a degradation in performances in the eclipse phase but only on the estimated quaternion.

Simulations that consider the continuous availability of both measurements show a better performance and a shorter convergence time, while simulations that consider the availability of only one measurement, the magnetic field vector, show comparable performances in the estimation of the angular velocity but a worse performance in the estimation of the quaternion.

Overall, it can be stated that the Kalman filter can truly provide an estimation of the state vector of the satellite even in the eclipse phase, thus allowing to perform a pointing control along the entire orbit. Comparing the two attitude determination methods it can be noticed that the Kalman filter provides overall a more precise estimation, it can cope with failures in the coarse Sun sensor, but requires greater computational resources. Considering that, due to power limitations, it will not be possible to perform a nadir pointing control in the shadow portion of the orbit, the adoption of the Kalman filter in the attitude determination phase is still under consideration.
Attitude control

Magnetic actuators are simple and affordable devices, however they can not make three axis control possible if no gravity gradient effect is present, as in the current application [7]. The attitude control of Palamede microsatellite has therefore only two operating modes: detumbling mode and nadir pointing mode.

Detumbling control mode

In its simplest form, the detumbling mode control requires only the measurements of the magnetometer, and is based on a negative feedback of the derivative of the measured magnetic field vector. The control torque and the actuator dipole moments are given by

\[ M_{MT} = -k_B \dot{B}_{pi} \quad ; \quad m = \frac{B_{pi} \wedge M_{MT}}{(B_{pi})^2} \]  

(29)

This simple feedback control has good performances when the angular velocity is large, since in this case it can be shown that the rate of change of the measured magnetic field vector depends primarily on the rate of change of the attitude of the satellite, therefore the rate of change of the rotational kinetic energy is guaranteed to be negative.
As the angular velocity becomes smaller, the intrinsic variations of the magnetic field along the orbit become more important, therefore the control can not guarantee energy dissipation. In the present case it has been verified that for angular velocities below 3 deg/s this control function becomes inadequate, therefore it has been decided to improve the detumbling control by switching to a negative feedback of the angular velocity, in the form $M_{MT} = -k_b \omega$. This requires knowledge of the angular velocity, which in the present application is available through the Kalman filter. The simulated performances of this dual control algorithm are very satisfactory and are reported in Figure 10. The first phase of the control depends only on the feedback of the rate of change of the magnetic field vector, and when the latter becomes small then the Kalman filter is activated. When estimated angular velocities are below the prescribed threshold, the control law is switched to the negative feedback of the angular velocity, thus allowing to reach at the end of the detumbling phase angular velocity components in the order of 0.2 deg/s.

\[
\dot{B}_{pi} = \dot{AB}_{orb} + \dot{AB}_{orb} = AB_{orb} \quad ; \quad \dot{E}_k = \omega^T M_{MT} = -k_b B_{pi}^T \dot{B}_{pi} \quad (30)
\]

An alternative detumbling control, that does not require the output of the Kalman filter, is based on a constant dipole moment control, with sign dependent on the rate of change of the measured magnetic field. When implemented in discrete form, the control law becomes

\[
M_{MT} = -m_x \cdot \text{sign}(\Delta B_{pi_x}) \Delta B - m_y \cdot \text{sign}(\Delta B_{pi_y}) \Delta B - m_z \cdot \text{sign}(\Delta B_{pi_z}) \Delta B \quad (31)
\]

Figure 10: performances of the detumbling control.
and the magnetic dipoles \( m_x, m_y \) and \( m_z \) are considered constant. Eventually, the values of the magnetic dipoles can be changed according to the order of magnitude of \( \Delta B_{pi} \), to fine tune the detumbling performances. Simulations not reported here for conciseness [8] show that the performances are comparable to the previous algorithm in terms of angular velocity damping, but require a longer time to stabilise the system, more or less two orbits instead of one.

**Nadir pointing control mode**

The availability of magnetic torquers as the only actuator make a true three axis control impossible. Therefore pointing control will be restricted to nadir pointing of the \( x \) axis of the satellite, coincident with the optical axis of the on board camera, thus allowing to properly capture images of the Earth. For this control mode, the Kalman filter must have reached convergence in order to provide the pitch and roll angles and angular velocities, that are used in a PD control to drive the \( y \) and \( z \) torquers, respectively aligned with the roll and pitch axes. This control mode is based on the assumption that the radial component of the magnetic field vector, \( B_x \), is not zero.

\[
\begin{align*}
B_{pi_x} & \Rightarrow \left\{ \begin{array}{l}
M_{MT_y} = m_z B_{pi_x} \\
M_{MT_z} = -m_y B_{pi_x}
\end{array} \right. \\
& \Rightarrow \left\{ \begin{array}{l}
m_x = 0 \\
m_y = -\frac{M_{MT_z}}{B_{pi_x}} \\
m_z = \frac{M_{MT_y}}{B_{pi_x}}
\end{array} \right.
\end{align*}
\]

As a net effect of the control, the satellite will align its \( x \) axis to the nadir direction, but the control will also produce a torque around this axis, in the form

\[
M_{MT_x} = m_y B_{pi_z} - m_z B_{pi_y}
\]

therefore the yaw rotation will be in principle not predictable. This can have effect on the performances, since in the image acquisition phase no limit on the yaw rotation is imposed, but the angular velocity must be less than 2 deg/s.

The simulated performances of the attitude control are shown in Figures 11 and 12. Assuming that a first detumbling control phase has been completed, the pointing control and the detumbling control are alternatively activated, depending on the orbit position and the estimated angular velocity. The attitude estimate is obtained with the Kalman filter. In the eclipse portion of the orbit, only detumbling control mode is used, while in the sunlit portion of the orbit the pointing control mode is activated unless angular velocity components are greater than 2 deg/s, otherwise detumbling mode is again activated. It can be seen that pointing control is capable of keeping pitch and roll angles close to zero, with angular velocities within the prescribed limits. In the eclipse portion of the orbit, the induced yaw angular velocity is slowly reduced, while the pitch and roll angular velocities tend, yet remaining within reasonable values. As the pointing control is activated anew, correct pointing conditions are recovered rapidly.
Angular velocity

\[ \omega_x (\text{deg/s}) \]

\[ \omega_y (\text{deg/s}) \]

\[ \omega_z (\text{deg/s}) \]

Figure 11: performances of the pointing control.

Euler angles

roll (deg)

eclipse eclipse eclipse

pitch (deg)

Figure 12: performances of the pointing control.
It is interesting to take a close look at the behaviour of the control as a function of the position along the orbit, where the radial component of the magnetic field has a great impact on the system performances. Twice per orbit, the $B_x$ component becomes almost zero, as reported in Figure 13. If this occurs with the satellite still far from its target attitude, then the actuators will easily saturate, while if attitude is already correct, or only detumbling mode is active, then there will be no visible effect on the control performances.

![Graphs of radial component of magnetic field, magnetic dipole moment of actuators, and orbit number vs. latitude](image)

Figure 13: performances of the pointing control.

**Conclusions**

The results obtained in a simulated environment have allowed to estimate the performances of different algorithms for attitude determination and control of Palamede microsatellite.

For attitude determination, the performances of the Kalman filter are superior to the algebraic attitude determination algorithm. However, the adoption of the Kalman filter is not yet defined as a baseline, due to the greater computational resources required. As soon as the on board software will be coded, tests will be performed to verify the compatibility of the Kalman filter with the CPU resources.

For attitude control, two modes have clearly been identified: a detumbling mode and a nadir pointing mode. Detumbling will be achieved with a constant dipole moment control, based on the sign of the rate of change of the measured magnetic field. Nadir pointing control will be instead guaranteed by a PD control along the pitch and roll axes, unless angular velocity components become greater than 2 deg/s
in which case the mode will be switched to pure detumbling. It must be considered that, based on limitations on available electric power, it has been decided to limit the pointing control only to the sunlit portion of the orbit. Further simulations will be performed to evaluate the effect of huge angular errors at the beginning of the pointing control, and eventually define a tilting strategy.

Acknowledgments

The authors wish to thank Ing. Fabio Bolandrina and Ing. Stefano Orsi for their useful contribution.

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