A Multi-mode Attitude Determination and Control System for SUNSAT

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ABSTRACT - New advanced control techniques for the attitude determination and control of the SUNSAT microsatellite are presented. The attitude sensors and actuators on a small satellite are limited in accuracy and performance due to physical limitations, e.g. volume, mass and power. To enhance the application of sophisticated payloads such as a high resolution imager within these confines, a multi-mode control approach is proposed, whereby various optimized controller functions are utilized during the orbital life of the satellite. To keep the satellite's imager and antennas earth pointing with the minimum amount of control effort, a passive gravity gradient boom, active magnetic torquers and a magnetometer are used. A "cross-product" detumbling controller and a robust Kalman filter angular rate estimator are used for the preboom deployment phase. A fuzzy controller and magnetometer full state extended Kalman filter are utilized for libration damping and Z-spin rate control during inactive imager periods. During imaging, when high performance is required, additional fine resolution earth horizon, sun and star sensors plus 3-axis reaction wheels are employed. Full state attitude, rate and disturbance estimation is obtained from a horizon/sun extended Kalman filter. A quaternion feedback reaction wheel controller is used to point or track a reference attitude during imaging and a near-minimum time, eigenaxis rotational reaction wheel controller during large angular maneuvers. Momentum dumping is done, using magnetic torquers and a "cross-product" algorithm. A new recursive magnetometer calibration method is also implemented to enhance the magnetic in-flight measurements.

1. INTRODUCTION

SUNSAT [1] is a 60 kg, almost cubical (0.45m) microsatellite scheduled for launch in March 1997 as a piggy-back payload on a Delta mission. The orbit configuration will be polar and slightly elliptical with a perigee at 450 km and an apogee at 800 km. The orbit is not sun-synchronous but will have a nodal drift of 0.77°/day. The main payload is a push broom imager capable of stereo imaging. The 3-colour, 3456 pixel imager will have a sub-satellite ground pixel spacing of 15 meter (51.8 km swath width) at an altitude of 600 km. For accurate pointing and low image distortion, the attitude must be determined to within 0.1°, the attitude controlled to within 0.2° and angular rates damped to below 0.005°/s.

SUNSAT will be an earth pointing satellite (body Z-axis towards nadir) to keep the imager in a nominal direction for usage and to permit acceptable antenna gain. A gravity gradient boom and tip mass will be deployed from the top of the satellite to earth-stabilize it using the minimum amount of control energy. The satellite will be kept in a slow Z-spin during normal operation (not during imaging), for improved solar thermal distribution. The four solar panels on the X/Y facets will thereby receive an even solar illumination, resulting in an improved life span of the solar cells. A simple momentum transfer to a Z-axis reaction wheel will despin the satellite before imaging sessions.

The following attitude control hardware is utilized: 3-Axis reaction wheels to provide an accurate, continuous and fast control capability. The maximum wheel angular momentum is 0.25 Nms per wheel. 3-Axis magnetic torquer coils to ensure high reliability and long life actuation. Magnetic torquing is used as the primary active stabilization method to do libration damping, Z-spin rate control and momentum dumping of the reaction wheels. A gravity gradient boom of 2.5 meter with a tip mass of 5 kg will be deployed for passive attitude control.

The following attitude sensing hardware is utilized: 3-Axis fluxgate magnetometer to measure the strength and direction of the geomagnetic field vector. This information is used to calculate the magnetic torques and to obtain full attitude information by comparing the measurements to geomagnetic field model data. 2-Axis horizon sensor utilizing a linear CCD array to obtain mainly pitch and roll attitude angles within a ±15° range to an accuracy of 0.02°. A fine slit sun sensor of similar CCD technology to obtain mainly yaw attitude information within a ±60° range to an accuracy of below 0.1°. A star sensor where a 10°×10° image is projected onto a 752×582 interline transfer CCD. The dynamic range of the star sensor is limited to the range Mv 2.5 and Mv 6.5 to measure the angular position of a star to an accuracy (1σ) of at
least 12 arcsec. A minimum of 3 separated stars must be detectable within the sensor's field of view to employ a full attitude determination algorithm using a star catalogue.

Due to the high processing load required by the multi-mode attitude determination and control software, it was decided to dedicate a T805 (transputer) microprocessor to this task. To keep the system redundant, most ADCS functions can also be implemented on one of the onboard computers on SUNSAT, whenever the dedicated processor might fail. Under normal conditions low level interfacing to the sensors and actuators will be done by a 80C31 microcontroller.

2. MODELS AND CONVENTIONS

The conventions used throughout this paper corresponds fully with the NASA standard as also defined in Wertz [2]. Two coordinate systems are used, the first being the spacecraft's body coordinates and the second the orbital coordinates. The body Z-axis is parallel to the boom's deployment direction but taken as positive opposite to this direction. The X and Y axes are perpendicular to two of the side solar panels. The orbital Z_0-axis is in the nadir direction, the Y_0-axis in the orbit anti-normal direction and the X_0-axis complete the orthogonal set. The X_0-axis will be in the orbit velocity direction for a circular orbit.

The attitude of the satellite can be defined by Euler angles (ordered series of right hand positive rotations from the X_0Y_0Z_0 set of orbital axes to the XYZ set of body axes). If we use a 1-2-3 sequence of rotations we can define the first rotation as a roll angle \( \phi \) (around X_0), the second as a pitch angle \( \theta \) (around an intermediate Y’-axis) and the last rotation as a yaw angle \( \psi \) (around Z). An attitude transformation matrix \( A \) (also called a direction cosine matrix) with elements trigonometric functions of \( \phi, \theta \) and \( \psi \) can then be used to transform a vector from the orbit to body coordinates [2]. Although the Euler angle representation has a clear physical interpretation it unfortunately suffers from singularities in the pitch angle. A better attitude representation without singularities and trigonometric functions in the transformation matrix is the Euler symmetric parameters [2]. All control and estimation algorithms in this paper will use the quaternion representation for attitude. Due to a lack of physical interpretation of the quaternion, the Euler angles will however be used to present the attitude where needed.

The spacecraft's attitude motion in orbit can be represented by the Euler dynamic and kinematic sets of equations when the quaternion representation is used. The dynamics of the spacecraft in inertial space will be governed by the following vector equation:

\[
I \dot{\omega}_B = N_{GG} + N_M + N_D - \omega_B \times (I \omega_B' + h) - h
\]  
(1)
where,
\[
I \quad = \text{moment of inertia tensor}
\]
\[
\omega_B' = [\omega_x \ \omega_y \ \omega_z]^T \quad = \text{inertially referenced body angular rate vector}
\]
\[
N_{GG} \quad = \text{gravity gradient torque vector [2]}
\]
\[
h = [h_x \ h_y \ h_z]^T \quad = \text{reaction wheel angular momentum vector}
\]
\[
N_M = M \times B \quad = \text{magnetic torque vector}
\]
\[
M = \text{vector dipole moment from all magnetic torquer coils}
\]
\[
B = \text{geomagnetic field vector in body coordinates}
\]
\[
N_D = \text{external disturbance torque vector}
\]
The kinematics (attitude) of the satellite of the satellite can be updated by the following vector equation:

\[
\dot{q} = \frac{1}{2} \Omega q
\]  
(2)
with,
\[
\Omega = \begin{bmatrix}
0 & \omega_{oz} & -\omega_{oy} & \omega_{ox} \\
-\omega_{oz} & 0 & \omega_{ox} & \omega_{oy} \\
-\omega_{oy} & -\omega_{ox} & 0 & \omega_{oz} \\
-\omega_{ox} & \omega_{oy} & -\omega_{oz} & 0
\end{bmatrix}
\]  
(3)
where,
\[
\omega_B' = [\omega_{ox} \ \omega_{oy} \ \omega_{oz}]^T \quad = \text{body angular rate referenced to orbit coordinates}
\]

3. PREBOOM DEPLOYMENT PHASE
After separation from the launcher and before the gravity gradient boom is deployed, a robust Kalman filter is used to estimate the body angular rate components of the satellite. A direct method to measure these rates can be done by using rate gyroscopes, but due to power and cost constraints on SUNSAT it was rather done indirectly by using measurements from the magnetometer. The rate of change of successive magnetometer vector measurements can be used to extract the angular body rates. The Kalman filter to estimate the orbit referenced angular body rates is described fully in [3]. The results and a few equations follow next.

The system model for the discrete Kalman filter can be obtained from (1) assuming a symmetrical satellite with almost equal moments of inertia (MOI) around all axes ($I_{xx} = I_{yy} = I_{zz} = I_{xyz}$) before boom deployment:

$$\omega^o_B = \omega^f_B = I^{-1}_{xyz} (N_M + N_D)$$  \hspace{1cm} (4)

The discrete system model (sampling period $T_s$) then used in the Kalman filter is:

$$x(k+1) = \omega^o_B(k+1) = \Phi x(k) + \Gamma u(k) + s(k)$$  \hspace{1cm} (5)

with,

$$\Phi = 1, \quad \Gamma = I^{-1}_{xyz} T_s 1$$

$$s(k) = N[\mathbf{0}, Q(k)]$$ \hspace{1cm} (Discrete zero mean system noise with covariance matrix $Q$)

The discrete measurement model can be obtained by assuming small angular rotations during each sampling period. The direction cosine matrix transforming successive vector measurements is then approximated as:

$$A \left\{ \omega^o_B(k) \right\} = \begin{bmatrix} 1 & \omega_{oz}(k)T_s & -\omega_{oz}(k)T_s \\ -\omega_{oz}(k)T_s & 1 & \omega_{oz}(k)T_s \\ \omega_{oy}(k)T_s & -\omega_{oy}(k)T_s & 1 \end{bmatrix}$$  \hspace{1cm} (6)

then,

$$v(k) = A \left\{ \omega^o_B(k) \right\} v(k-1)$$  \hspace{1cm} (7)

A time-variant measurement model follows directly from (6) and (7) as,

$$y(k) = v(k) - v(k-1) = H(k)x(k) + m(k)$$  \hspace{1cm} (8)

with,

$$H(k) = \begin{bmatrix} 0 & -v_z(k-1)T_s & v_z(k-1)T_s \\ v_z(k-1)T_s & 0 & -v_z(k-1)T_s \\ -v_z(k-1)T_s & v_z(k-1)T_s & 0 \end{bmatrix}$$

$$m(k) = N[\mathbf{0}, R]$$ \hspace{1cm} (Discrete zero mean measurement noise with covariance matrix $R$)

Simulations showed the Kalman filter to be robust and to converge within half an orbit to the true orbit referenced angular rates from any initial tumbling rate vector, as long as the sampling period is at least a factor of ten smaller than the tumbling period. When the magnetometer is used to supply the vector measurements the maximum estimation error after conversion will be equal to the orbital angular rate $\omega^w_o$. (The geomagnetic field vector rotates once w.r.t. the orbital coordinate set for a polar orbit)

As soon as the body angular rate values become known, a detumbling controller can be switched on to prepare the satellite for boom deployment. The idea is to align the body angular momentum vector to the satellite's Y-axis and the orbital normal direction. Furthermore, the Y-spin rate must conserve the angular momentum before and after boom deployment. All this will take place if the orbit referenced X and Z body rates are zeroed and the Y body rate satisfies a reference value, computed from the Y-axis MOI ratio. The boom will be deployed as soon as the angular rate error become small enough, at an instant when the +Z facet is pointing approximately towards zenith (assuming rapid boom extension). This will ensure a right side up capturing of gravity gradient for the satellite and small residual libration after boom deployment. The magnetic torquers and a cross-product control law [4] is used to prepare for boom deployment:

$$M = e \times B$$ \hspace{1cm} (9)
\[
e = K_w \left[ \omega_B^0 - \omega_{\text{ref}} \right] = K_w \begin{bmatrix} \omega_\alpha \\ \omega_\beta \\ \omega_\gamma - \omega_{\gamma-\text{ref}} \\ \omega_\nu \\
\end{bmatrix}
\]

(10)

where,
\[
K_w = \text{diagonal gain matrix} \quad \text{and} \quad \omega_{\gamma-\text{ref}} = \omega_\gamma \left( 1 - \frac{I_{\gamma}}{I_{\nu}} \right)
\]

Figure 1 shows a typical boom deployment sequence for SUNSAT during simulation. Initially the satellite was tumbling at an average, inertially referenced body rate of \( \bar{\omega}_B^I = \begin{bmatrix} 10 & 0 & 20 \end{bmatrix}^T \) revolutions per orbit (rpo). The boom extends to its full length within a few seconds. The Y-axis MOI values are, \( I_{\gamma} = 2 \text{kgm}^2 \) and \( I_{\nu} = 40 \text{kgm}^2 \). The satellite is tumbling at the initial rates until the halfway mark of the first orbit (50 minutes). The small oscillation on the orbit referenced rate components is due to a modulation with the orbital rate. Then the cross-product controller is enabled. The X- and Z-axis rates are zeroed and the Y-axis controlled to the reference rate of -19 rpo. All this is achieved within half an orbit. The boom is then deployed at the start of the second orbit when the +Z facet is pointing more or less towards zenith. It can be seen that all the orbit referenced angular rate values became close to zero after boom extension, with the result that the satellite is rotating inertially within the orbital frame. Gravity gradient capturing has been achieved and the boom axis is parallel to the nadir vector.

![Figure 1](image)

**Fig. 1** Angular rates during the boom deployment sequence

### 4. LIBRATION DAMPING

After boom deployment the sole stabilization purpose of the magnetic torquers will be to damp the external disturbance induced pitch and roll librations and to maintain a reference yaw rate or Z-axis spin. Small librations will ensure an approximately nadir pointing attitude and a constant Z-spin will improve thermal distribution on the satellite’s structure. Due to a continuous variation of the geomagnetic field vector the control possible by magnetic torquing is constrained - the magnetic torque vector will always be orthogonal to the field vector. The cross-product law (9) as already used to detumble the satellite, can also be used as a simple method of damping librations and regulating a Z-spin, with:

\[
e = K_w \left[ \omega_B^0 - \omega_{\text{ref}} \right] = K_w \begin{bmatrix} \omega_\alpha \\ \omega_\beta \\ \omega_\gamma - \omega_{\gamma-\text{ref}} \\ \omega_\nu \\
\end{bmatrix}
\]

(11)

In an attempt to improve the performance of the simple cross-product controller an alternative based on fuzzy logic has been designed. This controller is fully described and analysed in [3] [5] and no further detail except the block diagram of figure 2 will be given here. The controller consists of three MISO fuzzy control laws, one for each orthogonal magnetic
torquer. Each control law embodies a fuzzy rule base to decide on the control desirability and output level when using the corresponding torquer. A choice is then made to use the most favourable torquer during the next control interval.

![Block diagram of the fuzzy libration damping and Z-spin controller](image)

**Fig. 2** Block diagram of the fuzzy libration damping and Z-spin controller

Figure 3 shows a typical SUNSAT simulation of the fuzzy libration damping controller from an initial pitch and roll nadir pointing error of 10° and -5° respectively. The initial Z-spin was 5 rpo and commanded at the end of the third orbit to 10 rpo. The magnetic control commenced at the start of the second orbit. The final libration of ±2° in the pitch angle is mainly due to gravity gradient disturbance in the slightly elliptical ($e = 0.03$) orbit and aerodynamic disturbance. The roll offset (7.5° for $\omega_{z1} = 10$ rpo) is caused by gyroscopic coupling.

![Fuzzy libration damping/Z-spin control ($e = 0.03$)](image)

**Fig. 3** Fuzzy libration damping/Z-spin control ($e = 0.03$)

5. **SATELLITE STATE DETERMINATION**

Two extended Kalman filters (EKF) to determine the full satellite state from pairs of vector information will be utilized. The first vector will be obtained from any angular sensor measurement, e.g. utilizing a magnetometer, earth horizon, sun and star sensors. The second vector will be obtained from a reference model of the sensor data in a fixed coordinate frame, e.g. using geomagnetic field, satellite orbit, earth oblateness, sun orbit and star catalogue models. Two implementations of the basic EKF algorithms will be used for SUNSAT. The first EKF will make use of magnetometer measurements and an IGRF geomagnetic field model to supply the vector pairs needed for determination of the satellite’s attitude to an angular accuracy of below 1°. The second EKF will use measurements from the higher accuracy sensors
(CCD type horizon, sun and star sensors) and orbital models plus star catalogues to determine SUNSAT’s attitude to below 0.1° angular accuracy.

The derivations of the SUNSAT EKF algorithms is presented fully in [3] [6] and no detail except some results will be given here. The discrete full (8 element) state vector to be estimated, is defined as:

\[ x(k) = \begin{bmatrix} \omega_B^T(k) & q^T(k) & n_{doy}(k) \end{bmatrix}^T \]  

(12)

where,

\[ n_{doy}(k) = \text{slowly varying disturbance torque caused by aerodynamic pressure variations around } Y_0 \]

The innovation used in the EKF is the vector difference between the measured vector and the orbit modelled vector, transformed to the body coordinates by the estimated attitude transformation matrix:

\[ e(k) = v_{meas}(k) - A[q(k)]v_{orb}(k) \]  

(13)

e.g. where,

\[ v_{meas}(k) = \frac{B_{meas}(k)}{||B_{meas}(k)||}, \quad v_{orb}(k) = \frac{B_o(k)}{||B_o(k)||}, \quad \text{for the magnetometer based EKF.} \]

with,  

\[ B_o = \text{geomagnetic field model (IGRF) vector} \]

Simulation studies have shown that the SUNSAT magnetometer based EKF has an improved convergence and stability performance compared to the Psiaki et.al. method [7]. Full state tracking and convergence within an orbit is possible from any initial attitude angular error of ±90°. Figure 4 presents typical results from the Horizon/Sun sensor EKF when measurement data is only available for short periods of time (when a valid sunlit horizon or sun within the field of view is detected). The status of the innovation windows indicating sensor measurement availability (high = available, low = unavailable) is displayed in the lower part of figure 4. No attitude knowledge and rate values to an accuracy of 90% were assumed initially. Convergence is achieved within one orbit and accurate attitude tracking (< 0.1°) is established during periods when data from all the sensors becomes available simultaneously.

Fig. 4 Typical pointing angle estimation of the Horizon/Sun EKF

6. REACTION WHEEL CONTROL

To enable 3-axis stabilization three reaction wheels with mutually perpendicular axes are used. The reaction wheels on SUNSAT are only used during main payload imaging. Pointing and tracking controllers are used to obtain accurate and
low distortion images. Large angular slew maneuvers are optimized using a new near minimum time eigenaxis rotation algorithm. The globally stable quaternion feedback control law of Wie [8] was modified to be implemented on SUNSAT not as an inertially referenced but as an orbit referenced pointing control law:

\[ N_{\text{wheel}} = K\mathbf{q}_{\text{vec}} + D\mathbf{\omega}_B^0 - \mathbf{\omega}_B^I \times (I\mathbf{\omega}_B^I + \mathbf{h}) + \mathbf{N}_{GG} \]  \hspace{1cm} (14)

where,

\[ K = k\mathbf{I}, \quad D = d\mathbf{I} \quad \{\text{Scaled diagonal MOI matrices}\} \]

\[ \mathbf{q}_{\text{vec}} = [q_{3e} \quad q_{2e} \quad q_{3e}]^T \quad \{\text{vector part of the error quaternion}\} \]

The error quaternion \( \mathbf{q}_e \) will be the quaternion difference between the commanded \( \mathbf{q}_c \) and current satellite quaternion:

\[
\begin{bmatrix}
q_{1e} \\
q_{2e} \\
q_{3e} \\
q_{4e}
\end{bmatrix}
= 
\begin{bmatrix}
q_{4c} & q_{3c} & -q_{2c} & -q_{1c} \\
-q_{3c} & q_{4c} & q_{1c} & -q_{2c} \\
q_{2c} & -q_{1c} & q_{4c} & -q_{3c} \\
q_{1c} & q_{2c} & q_{3c} & q_{4c}
\end{bmatrix}
\begin{bmatrix}
q_1 \\
q_2 \\
q_3 \\
q_4
\end{bmatrix}
\hspace{1cm} (15)
\]

A tracking control law when SUNSAT is not 3-axis stabilized, but rather tracking a reference angular rate was also implemented: (The commanded quaternion must be propagated by the target’s kinematic equation)

\[ N_{\text{wheel}} = K\mathbf{q}_{\text{vec}} + D(\mathbf{\omega}_B^0 - \mathbf{\omega}_{\text{ref}}) - I\mathbf{\omega}_{\text{ref}} - \mathbf{\omega}_B^I \times \mathbf{h} + \mathbf{N}_{GG} \]  \hspace{1cm} (16)

where,

\[ \mathbf{\omega}_{\text{ref}} = \text{reference angular rate vector of the target} \]

Remote sensing satellites often have to embark on minimum time large angle slew maneuvers to point their imaging payload at different targets within a short span of time. The algorithm designed for SUNSAT attempts to balance the opposing goals of minimum time and minimum control effort when using reaction wheels. A practical solution to these opposing goals might be to minimize the maneuver time around the eigenaxis. An eigenaxis rotation results in the shortest angular path and therefore also in near minimum control effort. The algorithm is described in [3] [9]. It is essentially an open loop Bang-Off-Bang compensation method combined with a closed loop rate tracking regulator.

Figure 5 compares the new eigenaxis optimized method to a normal eigenaxis quaternion feedback controller, similar to equation (14). However, in the case of the normal eigenaxis controller, deviations from the eigenaxis will occur due to reaction wheel speed and torque saturation during large angular rotations. In figure 5 the first part (up to 1000 seconds) shows the behaviour of the new optimized controller and the second part (from 1500 seconds) shows the behaviour of a normal eigenaxis controller. It is clear that the roll, pitch and yaw angles have a symmetrical profile (indicating true eigenaxis rotations) when using the new controller. Although the comparative slew times were almost identical, the new controller needed almost 50% less control effort.

Momentum buildup on the reaction wheels naturally occurs due to the influence of external disturbance torques, for example, from gravity gradient torques, aerodynamic and solar forces. A cheap and effective means of unloading this momentum, is by making use of the magnetic torquers. Although optimal methods to do desaturation control has been designed for SUNSAT [3] [10], it was decided to implement a simple cross-product controller practically. The gain in desaturation efficiency could not be justified by the added complexity of the optimal controllers. The standard “cross-product” desaturation controller is [11]:

\[ \mathbf{M} = K_m (\mathbf{h} \times \mathbf{B}) / \| \mathbf{B} \| \]  \hspace{1cm} (17)

with,

\[ K_m = \text{controller gain constant} \]
7. MAGNETOMETER CALIBRATION

A three-axis magnetometer can measure the geomagnetic field strength and direction to high accuracy if it is well calibrated and free from magnetic disturbances. If the magnetometer is used in magnetorquing control applications only, a crude sensor will suffice. In contrast, when doing attitude determination to accuracies better than 0.5° per axis an accurate sensor will be required. A new RLS (Recursive Least Squares) algorithm [3] for in-flight magnetometer calibration (scale factor, misalignment and bias) will be used on SUNSAT.

A first order calibration model (with $B_{calib}$ the calibrated measurement and $B_{meas}$ the raw measurement) is used:

$$B_{calib}(k) = G(k)B_{meas}(k) + b(k)$$  \hspace{1cm} (18)

with,

- $G$ = scale factor/misalignment matrix
- $b$ = bias vector

If the calibrated vector is recursively compared to the modelled geomagnetic field vector (from IGRF model) to obtain the error to be minimized by a least squares method:

$$e(k) = y_{model}(k) - y_{calib}(k) = A(k)B_{meas}(k) - [G(k)B_{meas}(k) + b(k)]$$  \hspace{1cm} (19)

The attitude transformation matrix $A$ is therefore needed to transform the orbit modelled geomagnetic vector to body coordinates. Measurements from other attitude sensors are thus essential for the accuracy of this self-calibrating algorithm. The error vector can then be divided into three standard scalar least square parameter estimation problems:

$$e_i(k) = y_i(k) - \Phi^T(k)\Theta_i(k) \hspace{1cm} i = 1, 2, 3$$  \hspace{1cm} (20)

Simulations have shown convergence from any arbitrary (reasonable) scale factor, bias and misalignment errors to the true parameters to be completed within one orbit. Figure 6 presents the convergence of the bias vector from no initial knowledge to the true value of $b = [-5.0 \ 5.0 \ -8.0]^T \ \mu T$.

8. CONCLUSIONS

The multi-mode attitude determination and control system for SUNSAT was presented in this paper. The satellite control research at Stellenbosch University was directed mainly to the development of advanced attitude determination and control algorithms for LEO micro satellites with stringent performance, cost and power requirements, such as SUNSAT.
Most of the results are general enough and can be applied to a wide range of space missions. The new results were extensively tested by computer simulation and compared to current state of the art methods where applicable. All the sensors and actuators mentioned, were also developed and manufactured locally and will be space tested soon.

Fig. 6 Bias value convergence for a RLS calibrated magnetometer

REFERENCES


