AN ATTITUDE CONTROL SYSTEM FOR SUMBANDILASAT
AN EARTH OBSERVATION SATELLITE

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ABSTRACT
This paper describes the attitude determination and control system to support the multi-spectral earth observation main payload of the SumbandilaSAT microsatellite. The satellite has only a single main Y-body mounted solar panel and the attitude control system must ensure a nominal sun-pointed attitude under all non-imaging conditions during the sunlit part of the orbit. The control actuators employed are 3-axis magnetic torquer rods and reaction wheels. During initial detumbling and safe mode operations a simple new magnetic control law is used to bring the satellite to a sun-pointed Y-spinning attitude for maximum solar power collection.

From this sun-pointed, spinning attitude an intermediate control mode is entered when the Y-reaction wheel is utilised as a momentum wheel, to absorb the body spin rate and to inertially stabilise the angular momentum vector towards the sun direction. During the intermediate mode the magnetic rods are used to maintain the momentum vector size and direction and to do nutation damping. The pitch angle is also controlled using the Y-wheel, to keep the main imager payload as close as possible to an earth-pointed attitude and to thermally stabilise the imager telescope.

The final and nominal attitude control mode is entered when a zero biased 3-axis reaction wheel controller is enabled, for: 1) sun tracking for optimal solar power collection, 2) target tracking during viewfinder use or during imaging downloading communication with a ground station and 3) pushbroom imager scanning with a forward motion compensation capability. During the nominal mode the magnetic rods are used to dump the angular momentum from the reaction wheels during sun tracking periods.

A short introduction to the Sumbandila satellite will be given. All the control modes, the attitude sensors and estimators utilised, will be introduced in the paper. Specifically, a unique agile viewfinder control mode to manually select targets for subsequent high resolution image scanning, when a control ground station is available within the communication footprint of the satellite, will be explained more thoroughly.

1. INTRODUCTION
SumbandilaSAT is the second low earth orbiting (LEO) microsatellite fully developed and built in South Africa. The satellite was manufactured at Sun Space & Information Systems, a delegate company of the University of Stellenbosch. The expected launch date will probably be towards the end of 2008 from a Russian submarine in the North Sea. The orbit selected is a 510 km circular sun-synchronous 10h00 am/pm orbit. The main payload is a 6-spectral band pushbroom imager with a ground sampling resolution of 6.25 meter per pixel. The spectral bands were chosen to best suit agricultural applications e.g. crop planning, food security, water resource monitoring and also to support remote sensing applications such as disaster management, urban and environmental planning. One of the functions of the onboard ADCS is to ensure accurate pointing and stabilisation of the imager boresight towards selected targets on the ground. The high resolution imager has a swath of 45 km at nadir and will scan the ground surface in the multispectral bands using a forward motion compensation method, i.e. performing a pitch rotation while imaging, to slow down the ground scanning velocity of the imager boresight as to increase the integration time on the CCD imager arrays.

The SumbandilaSAT ADCS hardware and their main characteristics are listed in Table 1 and an internal view of the 82 kg satellite is shown in Fig.1. The absolute attitude sensors, i.e. the 3-axis magnetometer, sun, earth horizon and star tracker sensors are used to measure various vectors in body coordinates and by comparing these with corresponding modelled vectors in the reference coordinates (inertial and orbit frame), the attitude can be estimated. The fibre optic rate gyroscopes (FOGs) are used to measure the satellite’s angular body rates (inertially referenced) during fast and large attitude manoeuvres when the absolute attitude sensors become ineffective to use. During these periods the attitude will be propagated using an Extended Kalman filter (EKF) and the FOGs measured body rate vector. The attitude actuators will be 3-axis magnetic torquer rods and reaction wheels. The torque rods are used during initial detumbling (after separation from the launch vehicle) to bring the satellite from a random tumbling motion to a slow controlled spin, while pointing the Y-facet with main solar panel towards the sun. The 3-axis reaction wheels will then be used to stabilise the satellite in a nominal sun tracking attitude and during earth imaging operations in a nadir pointing attitude. Due to external aerodynamic, magnetic and solar secular disturbance torques, angular momentum build-up can occur on the reaction wheels. The magnetic torquer rods will then be used to dump this wheel momentum and maintain near zero speeds on the wheels during nominal flight.
Earth observation satellites are normally preprogrammed to image specific target areas on earth. Due to cloud coverage and off-pointing errors the selected target locations are often not imaged adequately. A solution to these problems, when the target locations and satellite ground station are both within the satellite’s footprint, will be to use an onboard TV camera as feedback to the ground station and steer the main imager boresight towards the required target areas before capturing the high resolution imaging data [1]. Another application is for the military or during intelligence gathering to image selected remote targets of interest using a mobile ground station within a radius of several thousand kilometers via a LEO satellite. A satellite view finder is also an excellent tool to demonstrate the agility of LEO satellites in real time.

The satellite view finding system can be considered as a complete stand-alone strap-on system for any satellite. The space component consists of:

1. a Wide Angle (WA) TV camera (approximately 100 km swath from a 500 km orbital altitude)
2. a Narrow Angle (NA) TV camera (swath of about 2.5 km from 500 km) with a ground sampling resolution of about 4 meters per pixel
3. a 3-axis reaction wheel and fiber optic gyro system to point and slew the boresight of the main imager and the co-aligned WA and NA TV cameras
4. FM modulator to combine and up-convert the TV signals on the satellite’s X-band downlink transmitter

The ground segment consists of:

1. FM demodulator from the X-band receiver to output the two TV signals
2. A TV monitor to display the WA and NA images in real time
3. Personal computer to input the ground operator’s viewfinder steering commands (e.g. using a joystick) and control mode selections and to interface with the ground station’s telecommand system for real time uplink transmission

Due to the limited time during a normal satellite pass over a ground station (5 to 10 minutes), the satellite will be attitude controlled to track a selected target area, preferably in advance of ground station visibility (1: target tracking mode). As soon as the satellite TV signals are received, the ground operator can re-point the boresight by rate steering of incremental commands (2: target steering mode). These X/Y steering commands slew the boresight along and/or across the satellite’s flight direction. The rotation angle (yaw) around the boresight is kept at zero to prevent any disorientation of the operator and to keep the main imager sensor orthogonal to the satellite’s velocity direction for pushbroom operation.

When the operator is satisfied that he has a desired target located in the NA field of view (FOV), he can command the attitude control system to lock onto the target (3: target lock mode). By onboard processing the target direction vector is first determined and then subsequently propagated to generate a reference attitude and angular rate vector for use by the attitude control system. When the operator wants to enable pushbroom imaging of the selected target area, the satellite’s pitch rotation rate is adjusted within a few seconds to the required forward motion compensation (FMC) level to enable pushbroom imaging (4: imaging mode).

2. INITIAL & SAFE MODE DETUMBLING

After release from the launcher stage the satellite will be in a random tumbling attitude. The maximum expected angular rate magnitude should be less than 10°/s from the launch vehicle specification. To maximise the solar power collection the main solar panel mounted on the satellite’s +Y-facet must be sun pointing. Two small backup solar panels on the ±X-facets will assist in
solar power collection during the random tumbling stage and during a controlled Y-Thomson spin [2] when the satellite aligns the body +Y-axis with the orbit anti-normal direction. Using only a simple B-dot magnetic controller [3] the X and Z body angular rates can be zeroed and a magnetic Y-spin controller can be employed to enable a Y-Thomson spin reference of -2°/s. These controllers are very simple to implement in the microcontroller of the Magnetic (sensor & actuator) Interface Unit (MIU) and do not initially require the onboard ADCS processor to be commissioned. The magnetic control laws for the torquer rods during the detumbling stage are:

\[
M_y = K_d \frac{d\beta}{dt} \quad \text{for} \quad \beta = \arccos \left( \frac{B_y}{|B|} \right) \\
M_x = K_s \left( \omega_{xy} - \omega_{xy}^{\text{ref}} \right) \text{sgn}(\beta) \quad \text{for} \quad |B_x| > |B_y| \\
M_z = -K_s \left( \omega_{zy} - \omega_{zy}^{\text{ref}} \right) \text{sgn}(\beta) \quad \text{for} \quad |B_z| > |B_y|
\]

(1)

with,

\[ M = [M_x, M_y, M_z]^T \] = Magnetic moment vector of torquer rods
\[ B = [B_x, B_y, B_z]^T \] = Magnetometer measured B-field
\[ \beta = \text{Angle between body +Y-axis and local B-field} \]
\[ K_s \text{ and } K_d = \text{Damping and spin controller gains} \]
\[ \omega_{xy}^{\text{ref}} = \text{Reference Y-body spin rate} \]

(typical -2°/sec for SumbandilaSAT)

The Y-spin controller requires knowledge of the orbit referenced Y-body angular rate \( \omega_{xy} \) of the satellite. This can be estimated using a simple, robust Kalman filter using only the magnetic field vector measurements. The algorithm for this rate estimator can be found in [4]. The advantages of this estimator are 1) no orbit nor geomagnetic reference field knowledge i.e. no SGP4 nor IGRF reference models will be required, 2) it has low computational demand and can also be implemented in the MIU microcontroller and 3) it is robust against modelling errors (e.g. inaccurate satellite moment of inertia parameters) and will not diverge. The main disadvantage are that the rate estimates can contain errors similar in size to the orbit angular rate \( \omega_{xy} \) (orbit mean motion and also the mean relative rate of the body B-field measurements to the rotating orbit coordinates). The latter will not present any control problems as long as the Y-spin rate is much larger than \( \omega_{xy} \).

A stable Y-Thomson attitude will be achieved within 1 or 2 orbits depending on the magnitude of the initial body rates. SumbandilaSAT will be launched in a 10am/10pm sun-synchronous orbit, meaning that the sun angle \( \beta_s \) normal to the orbit plane (+Y-facet of the satellite in Y-Thomson) will be roughly 60° to the sun (= 50% power collection from the main solar panel). To increase the available power even further the Y-spin axis can be precessed to the sun vector. This is done by a novel magnetic precession controller, using only an approximate sun vector measurement obtained from a Coarse Sun Sensor (CSS). The CSS consists of 6 solar cells mounted on the 6 facets of an imaginary cube. To prevent shadowing of these cells they are actually mounted on 2 pedestals with 3 orthogonal cells each, in the ±Z-facets of the satellite. The aim is to precess the +Y-axis towards the sun direction when not orbiting in eclipse and to apply the Y-Thomson controller of Eq.(1), when in eclipse. The sun-seeking precession controller is based on a magnetic cross-product control law to choose the optimal torquer rod magnetic moment for precessing the +Y-body axis (negative of Y-spin angular momentum vector) towards the inertially fixed sun direction. The sun-seeking control law is implemented as:

\[
\Omega_{\text{precess}} = k \left( \textbf{V}_{\text{body}} \times \textbf{S}_{\text{sun}} \right) = K_p \beta_s \left[ -S_z S_x \right]^T \\
\beta_s = \arctan \left( \frac{S_x^2 + S_y^2}{S_y} \right) \\
N_{\text{req}} = \Omega_{\text{precess}} \times \textbf{H}_{y-\text{spin}} = K_p \beta_s \left[ \textbf{H}_{y-\text{spin}} \right]^T \\
M = \frac{B \times N_{\text{req}}}{|B|} = \frac{K_p \beta_s \left[ \textbf{H}_{y-\text{spin}} \right]}{|B|} \left[ -S_z, -S_z, \frac{-S_x B_y}{S_z B_x - S_y B_z}, \frac{-S_x B_y}{S_z B_x - S_y B_z} \right]^T
\]

(2)

with,

\[ S_{\text{CSS}} = [S_x, S_y, S_z]^T \] = CSS unit vector measurement
\[ \beta_s = \text{Angle between body +Y-axis and CSS unit vector} \]
\[ \textbf{H}_{y-\text{spin}} = \text{Y-spin angular momentum vector along negative Y-body direction} \]
\[ K_p = \text{Sun-seeking controller gain} \]

The combination of the control laws in Eqs.(1) and (2) will result in a simple magnetic means of maintaining a stable Y-spin rate, while roughly pointing the +Y-facet towards the sun direction. These controllers are implemented autonomously in the MIU to bring the satellite without any ground station interaction to a stable sun pointing, Y-spin attitude to maximise solar power collection. This initial safe attitude is possible with a minimum ADCS system. Only the MIU, a flux gate magnetometer, the CSS and the 3-axis torquer rods are required. For redundancy reasons, two MIUs are used, the second unit on hot standby and both units have access to the required sensors and torquer rods. The controllers above will also be used as a safe mode fallback if the onboard ADCS processor fails to actively control the satellite. The MIUs will act as watchdogs for the ADCS processor and when not serviced regularly, they will take command autonomously and bring the satellite to a safe sun-pointing Y-spin attitude until the
problem can be corrected at the next ground station interaction.

3. WHEEL CONTROL MODES

3.1 Y-Momentum Wheel Mode

From the Y-spin attitude the Y-axis reaction wheel will be used as a momentum wheel to absorb the body momentum and bring the satellite to a stable and approximately earth pointing attitude, while maintaining the alignment of the Y-momentum vector to the sun. As this sun aligned Y-wheel controller requires not only angular rate knowledge, but also pitch angle knowledge, a full attitude and angular rate estimator will be required. An Extended Kalman Filter (EKF) is implemented to estimate the full angular state of the satellite from the flux gate magnetometer and CSS vector measurements (in body coordinates) and the corresponding modelled vectors (in orbit reference coordinates), see [4] for a detailed derivation. The 7-element discrete state vector to be estimated, is defined as:

$$\dot{x}(k) = \begin{bmatrix} \dot{\omega}_x(k) \\ \dot{\theta}_y(k) \\ \dot{\theta}_z(k) \end{bmatrix}$$

(3)

with,

$$\dot{\omega}_x(k) = \begin{bmatrix} \dot{\omega}_x(k) \\ \dot{\theta}_y(k) \\ \dot{\theta}_z(k) \end{bmatrix}$$

= Body inertial referenced angular rate vector estimate

$$\dot{q}(k) = \begin{bmatrix} \dot{q}_1(k) \\ \dot{q}_2(k) \\ \dot{q}_3(k) \\ \dot{q}_4(k) \end{bmatrix}$$

= Orbit referenced quaternion vector estimate

The innovation used in the EKF is the vector cross-product of a measured body reference unit vector and a modelled orbit reference unit vector, transformed to the body coordinates by the estimated attitude transformation matrix \(A[q(k)]\):

$$e(k) = v_{\text{body}}(k) \times A[q(k)] v_{\text{orbit}}(k)$$

(4)

with,

$$v_{\text{body}}(k) = B_{\text{mag}}(k) \bigg/ B_{\text{mag}}(k) \quad \text{or} \quad S_{\text{CSS}}(k)$$

$$v_{\text{orbit}}(k) = B_{\text{ref}}(k) \bigg/ B_{\text{ref}}(k) \quad \text{or} \quad S_{\text{sun}}(k)$$

The Y-wheel control law to absorb the Y-body momentum and control the pitch angle to an approximately earth pointing attitude, while maintaining the Y-axis sun aligned, is:

$$N_{ny}(k) = K_{py} \arcsin \left( q_2(k) \text{sgn}(\dot{q}_4(k)) \right) + K_{dy} \dot{\theta}_y(k)$$

(5)

with,

$$K_{py} \text{ and } K_{dy} = \text{Proportional and derivative gains}$$

To maintain the Y-wheel momentum at a certain reference level (corresponding to the Y-body momentum in Y-spin mode) and to damp any body nutation rate in the X and Z axes, a magnetic cross-product control law is utilised [3]:

$$M(k) = \frac{e(k) \times B(k)}{B(k)}$$

(6a)

with,

$$e(k) = \begin{bmatrix} K_y \dot{\theta}_y(k) \\ K_y \dot{\theta}_y(k) - h_{wy-ref} \end{bmatrix}$$

(6b)

and

$$K_y = \text{Nutation damping gain}$$

$$h_{wy-ref} = \text{Y-wheel reference angular momentum}$$

(typical -0.3 Nms for SumbandilaSat)

The cross-product controller of Eq.(6) is only applied during eclipse, during the sunlit part of the orbit the sun-seeking magnetic controller of Eq.(2) is implemented. During initial commissioning the Y-momentum control mode is used to calibrate and determine the alignment of all the accurate attitude sensors, i.e. the CMOS matrix sun and earth horizon sensors and the CCD star tracker. After the in-orbit calibration and alignment parameters have been determined, the measurements from these sensors can then be included in the EKF of Eq.(3) to improve the attitude and rate estimation accuracy. Next the satellite will be ready for the full 3-axis reaction wheel control mode to be employed.

3.2 Zero-bias 3-Axis Reaction Wheel Mode

From the sun-aligned Y-momentum wheel mode, the X and Z reaction wheels can be activated and a 3-axis sun-pointing reaction wheel controller implemented. The globally stable quaternion feedback controller of Wie [5], [8] was modified to become an orbit referenced pointing control law. The quaternion and rate reference vectors can be generated from a sun orbit model for the sun-pointing attitude, or it will be zero vectors for the nadir-pointing attitude or any specified constant attitude reference for a specific roll, pitch or yaw requirement. The 3-axis reaction wheel control law (wheel torque vector) to be used for all cases, is:

$$N_i(k) = K_{p1} [l_i \dot{q}_{ref}(k) + K_{p2} \dot{\omega}_i(k)] - K_{d2} \dot{\omega}_i(k) \times [l_i \dot{\omega}_i(k) + h_i(k)]$$

(7)

with,

$$K_{p1} = 2 \omega_i^2 \quad K_{p2} = 2 \zeta_2 \omega_i$$

(Pointing gains for dominant bandwidth and damping)

$$I = \text{Satellite moment of inertia tensor}$$

$$h_i(k) = \text{Measured angular momentum of wheels}$$

$$\dot{\omega}_i(k) = \begin{bmatrix} \dot{\theta}_x(k) \\ \dot{\theta}_y(k) \\ \dot{\theta}_z(k) \end{bmatrix}$$

= Body orbit reference angular rate estimate
\[ \mathbf{q}_{\text{err}}(k) = \begin{bmatrix} q_{1e}(k) & q_{2e}(k) & q_{3e}(k) \end{bmatrix}^T \]

where

\[ \mathbf{q}_{\text{err}}(k) = \mathbf{q}_{\text{com}}(k) \Delta \hat{\mathbf{q}}(k) \]

\[ \mathbf{q}_{\text{com}}(k) = \text{Commanded quaternion, e.g. sun reference} \]

\[ \Delta = \text{Quaternion delta operation} \]

The nominal reaction wheel control mode will do sun-pointing in the sunlit part of the orbit and nadir-pointing i.e. \( \mathbf{q}_{\text{com}}(k) = \begin{bmatrix} 0 & 0 & 1 \end{bmatrix}^T \), in eclipse. The nadir-pointing attitude will ensure optimal antenna coverage for ground communication during eclipse and thermal stability for the imager telescope. Continuous momentum management of the reaction wheels is done using a simple cross-product magnetic controller [3]:

\[ \mathbf{M}(k) = K_m \frac{\mathbf{h}_e(k) \times \mathbf{B}(k)}{\| \mathbf{B}(k) \|} \quad (9) \]

with,

\[ K_m = \text{Momentum dumping gain} \]

3.3 Viewfinder Control Modes

During the viewfinder control modes 1 to 4 described in the Introduction section, the high resolution attitude sensors (i.e. star trackers) can not be used due to the high slew rates. Other attitude sensors e.g. sun and earth horizon sensors may supply only sporadic data due to their limited FOV; i.e. the roll or pitch angle can become large and the earth sensor may lose the horizon angle measurement from the sensor, the sun may also move out of the FOV of the sun sensor due to satellite body rotations. The satellite’s attitude is then determined from the propagated fast (10 Hz) fibre optic gyro (FOG) rate measurements, with absolute attitude and rate bias corrections from the slow (typically 1 Hz) attitude vector sensors e.g. magnetometers, horizon and sun sensors (when valid measurements are available). The optimal blending of these fast and slow measurements can be done in a FOG Extended Kalman Filter (FOGEKF) of which the full derivation and implementation are discussed in [6].

The estimated full 7-state vector of the FOGEKF is:

\[ \dot{x}(k) = \begin{bmatrix} \dot{\mathbf{q}}(k) \\ \dot{\mathbf{b}}(k) \end{bmatrix} \quad (10) \]

Two possible scenarios exist for earth targets to be tracked from the satellite: The first case is when a target is known a-priory and the target’s coordinates uploaded to the satellite, the second case is when the target is selected manually in real time by the ground operator through the TV feedback signal. The target tracking generators to calculate the commanded quaternion \( \mathbf{q}_{\text{com}}(k) \) and angular rate vectors for the reaction wheel controllers are derived fully in [7]. The geometry during tracking to calculate the satellite to target vector in orbit reference coordinates, is shown in Fig.2.

All view finder control modes are implemented using the 3-axis reaction wheels of the satellite. The wheel controller sampling time \( \Delta T \) is chosen equal to the FOG sampling time (i.e. the EKF propagation sampling time) of 0.1 seconds. The wheel torque command vector \( \mathbf{N}_w \) is calculated using a quaternion feedback control law similar to Eq.(7), when target tracking is required. If we define the integral of the quaternion error vector as \( \dot{\mathbf{q}}_{\text{err}}(k) \), where \( \dot{\mathbf{q}}_{\text{err}}(k) = \dot{\mathbf{q}}_{\text{err}}(k-1) + \dot{\mathbf{q}}_{\text{err}}(k) \Delta T \), then for the target tracking and target lock control (view finder modes 1 and 3):

\[ \mathbf{N}_w(k) = K_{p2} \mathbf{I} \dot{\mathbf{q}}_s(k) + K_{p2} \mathbf{I} \dot{\mathbf{q}}_{\text{com}}(k) + K_{p2} \mathbf{I} \dot{\mathbf{q}}_{\text{com}}(k) \Delta T \]

with,

\[ K_{p2} = 2 \left( \dot{\mathbf{\omega}}_{\text{e}}^2 + 2 \zeta \dot{\mathbf{\omega}}_{\text{e}} \right) / \Delta T \]

\[ K_{12} = 2 \zeta \dot{\mathbf{\omega}}_{\text{e}} + 1 / \Delta T \]

\[ K_{22} = 2 \dot{\mathbf{\omega}}_{\text{e}}^2 / \Delta T \]

\[ \dot{\mathbf{\omega}}_{\text{e}}, \zeta = \text{Dominant second order closed loop specifications} \]

\[ \mathbf{\omega}^0_{\text{com}}(k) = \text{Orbit referenced target tracking angular rate vector} \]

The main difference between the reaction wheel controller presented in Eq.(11) and the one presented in [8], is that the attitude and angular rate values are all referenced to the orbit coordinates and not inertial coordinates, to ensure a nominal earth pointing attitude. The gyroscopic feedforward term is included to compensate for the gyroscopic torques generated during view finder operations, due to the fairly high levels of body angular rate and reaction wheel momentum at these times.
Figure 2: Target Tracking Geometry

During target steering control (view finder mode 2) the rate steering vector $\omega_{steer}$ with $X$ & $Y$-axis commanded values (as received on the uplink from the ground operator via the joystick input) and a zero value for the $Z$-axis, are included to implement a wheel control torque. This controller will steer the imager boresight around the target (along and across track), while maintaining a zero yaw angle around the body $Z$-axis,

$$N_a(k) = K_{D2} \left[ \omega_p^O(k) - \omega_{com}^O(k) - \omega_{steer}(k) \right] - \omega'_p(k) x \left[ \omega'_p(k) + h_n(k) \right] + K_{P2} I_{q_{err}}(k) + K_{I2} I_{q_{err}}(k) \quad \text{only for Z-axis} \quad (12)$$

Finally, the imaging controller (view finder mode 4) will 3-axis stabilise the satellite within the orbit reference coordinates to enable the pushbroom imager to scan (in the pitch axis) the ground target at the required FMC ratio (FMC = 1 means at ground velocity speed). The $X$ and $Y$ reaction wheel speeds will be open-loop adjusted according to the satellite/reaction wheel MOI ratio, to obtain the required satellite pitch ($Y$-axis) FMC angular rate. As soon as the required $X/Y$ wheel speeds are reached (within a few seconds), the imaging control law is activated,

$$N_a(k) = K_{P3} I_{q_{err}}(k) + K_{D3} \left[ \omega_p^O(k) - \omega_{com}^O(k) \right] - \omega'_p(k) \times \left[ \omega'_p(k) + h_n(k) \right] \quad (13)$$

with,

$$K_{P3} = 2\omega_n^2, \quad K_{D3} = 2\zeta\omega_n.$$  Imaging controller gains for dominant closed loop bandwidth and damping

4. SIMULATION RESULTS

All control modes were tested in a hardware-in-the-loop (HIL) simulation environment, where the satellite orbit, the satellite attitude dynamics and kinematics plus all the ADCS sensors and actuators were modelled and simulated with their expected measurement noise parameters, FOV restrictions and torque disturbances, on a Personal Computer external to the satellite. Table 2 gives some information on the satellite, orbit, ADCS sensors and actuators as relevant to the simulation tests.
The flight code was executed on the satellite ADCS computer and the control commands and TV signals (for view finding) communicated through RF links between the satellite and the ground station (i.e. the hardware component).

Fig.3 shows the initial detumbling performance from an initial angular rate vector $\omega_0(0) = [2 \ 0 \ 1]^T \text{s}^{-1}$. During the first orbit no control was done and from 6000 seconds the detumbling and Y-spin controller of Eq.(1) was enabled. Within less than 1 orbit the satellite was controlled into a -2°/sec Y-Thomson spin using magnetorquers only. The body angular rates were estimated by a robust angular rate Kalman filter utilising only the raw magnetometer vector measurements.

Figs. 4 and 5 show the performance of the sun-seeking control law of Eq.(2) activated at 3000 seconds to precess the body Y-spin vector towards the sun. At 12 000 seconds the vector orthogonal to the main +Y-axis solar panel is within 15° to the sun vector ($\beta_s$ angle). The Y-momentum wheel controller of Eq.(5) is then activated and the magnetic control law of Eq.(6) is used to damp X/Z nutations and to maintain the Y-wheel momentum, during eclipse while during the sunlit part of the orbit, the sun-seeking magnetic control law of Eq.(2) is utilised. It can be seen from Fig.5 that the $\beta_s$ angle reduce to about 10°, resulting in more than 98% of the maximum available main solar panel power (according to the cosine-law of incidence). During the Y-momentum wheel control mode, the magnetometer and CSS EKF estimator was used to supply the required attitude and rate feedback for the wheel and magnetic controllers.

Fig.6 presents the RPY attitude control results when the nominal 3-axis reaction wheel controller mode is implemented, i.e. during the sunlit part of the orbit the +Y-axis (solar panel) is sun pointed and during eclipse the +Z-axis (antennae and imager) is nadir pointed.

Fig.7 presents some agile manoeuvres during two imaging sessions at FMC-4. Initially the satellite was nadir pointing and utilising the FOGEKF with all attitude sensors for accurate estimates. At 25 seconds the satellite rotates to a sun-pointing attitude and at 90 seconds the first imaging session starts when the satellite rotates back to a nadir-pointing attitude. At 180 seconds the satellite does a roll and pitch off-pointing manoeuvre to setup the correct attitude for target #1. At 240 seconds the Y-wheel ramps for 10 seconds to obtain the FMC-4 body rate and from 250 to 284 seconds imaging can take place to scan a strip of approximately 60 km on the earth’s surface. At 330 seconds the satellite rotates to a sun-pointing attitude again, then nadir pointing at 395 seconds and at 490 seconds an off-pointing manoeuvre for target #2. At 550 seconds the 10 second ramp will start and a FMC-4 scan for the second target from 560 to 594 seconds. All the commands above will be scheduled autonomously onboard the satellite based on time predictions of target #1 and #2 centre of image instances.

Figs. 8 and 9 present a typical view finding control sequence. At 25 seconds the satellite rotates to a sun-pointing attitude and at 130 seconds the target tracking mode is commanded. The satellite rotates to the preselected target and starts to follow it with the +Z-axis (view finder camera’s bore sight). At 380 seconds the target steering mode is activated. Initially several negative X-rate (roll) increments are send, then finally several positive X-rate increments, until the X-rate is zeroed again. At 440 seconds a target lock mode command is uplinked from the ground station. The satellite then started to track the ground target, until the pitch angle reached about +8° at 490 seconds. At this instant a target imaging mode command is issued. The tracking pitch rate of about -0.6 deg/sec was then adjusted in about 4 seconds to the correct FMC-4 rate, using a maximum wheel torque to dump the satellite angular momentum onto the Y-axis reaction wheel (see Fig.9). From about 496 to 536 seconds the pushbroom imager scans the selected ground target area below at a FMC-4 rate. Finally at 536 seconds, the satellite reverts back to sun-pointing control. The initial target tracking command can be scheduled for autonomous onboard execution to ensure an already target-pointed satellite when the first view finder TV signals are received. The rest of the commands above are transmitted in real-time by the ground station operator to steer, lock and image a newly selected target.

5. CONCLUSION

This paper presents the relevant control modes, some implementation details, an overview of the ADCS, and also a novel view finder system for the SumbandilaSAT satellite. Attitude and rate estimates are obtained from onboard Kalman filters (Rate, EKF & FOGEKF), using the attitude vector sensors and FOG rate sensors. The initial and safe-mode detumbling controllers are presented as well as a novel sun-seeking magnetic control law. A Y-momentum sun pointing controller that ensures initial 3-axis stability for commissioning of the full ADCS and a 98% maximum power budget, is presented. The complete suite of 3-axis reaction wheel controllers for nominal, imaging and view finder usage, are presented. The seamless switching between these different reaction wheel controllers are also clear from the various simulation tests. The HIL simulation results indicate good performance during all control modes. Currently the view finder steering mode based on angular rate incremental commands from the ground operator promises acceptable behaviour, in spite of the substantial feedback delay of 1 to 2 seconds. The latter result is achieved mainly through the onboard reaction wheel system with rate feedback control assistance, e.g.
during target steering mode manoeuvres.

6. REFERENCES


| Table 2: Parameters (orbit/satellite/actuators/sensors) used in the HIL simulation |
|----------------------------------|----------------------------------|
| **Orbital parameters** | 510 km circular 10h00 am/pm SS orbit |
| **Satellite mass properties** | 82 kg, I = \[
\begin{bmatrix}
6.61 & -0.11 & 0.15 \\
0.11 & 8.64 & -0.09 \\
-0.15 & -0.09 & 4.46
\end{bmatrix} \text{ kgm}^2 |
| 3-Axis XYZ Reaction wheels | Wheel momentum: \( h_{wheel} \approx \pm 0.8 \text{ Nms} \) \\
| (wheel speed controlled) | Wheel torque: \( N_{wheel} = \pm 0.025 \text{ Nm} \) \\
| | Wheel torque noise < 0.001 Nm (RMS) \\
| | Speed control accuracy: \( \delta h_{wheel} < 10^{-4} \text{ Nms} \) \\
| | Speed controller sampling period = 0.1 sec |
| 3-Axis XYZ Magnetorquer coils | Magnetic moment: \( M = [5 \ 5 \ 5] \text{ Am}^2 \) \\
| (PWM controlled) | Minimum pulse resolution: \( \delta T = 0.01 \text{ sec} \) \\
| | Magnetic control sampling period = 1 sec |
| 3-Axis Fibre Optic Gyros (FOG) | Range: \( \omega_{max} = \pm 8 \text{ °/sec} \) \\
| (10 Hz angle increment output) | Measurement noise < 0.005 °/sec (RMS) \\
| | Random walk bias < 3 °/hr |
| 2-Axis Earth Horizon sensor (visible spectrum) | FOV = ± 40 ° (pitch and roll) \\
| (1 Hz nadir vector measurements) | Accuracy < 0.1 ° (RMS) |
| 2-Axis Sun sensor | FOV = ± 45 ° (azimuth and elevation angle) \\
| (1 Hz sun vector measurements) | Accuracy < 0.1 ° (RMS) |
| 3-Axis Magnetometer and 10th order IGRF model | Range = ± 60 µT \\
| (1 Hz B-field vector measurements) | Accuracy < 50 nT (RMS) for magnetometer \\
| | Accuracy < 300 nT (RMS) for IGRF model |
| Satellite GPS receiver and SGP4 orbit propagator | GPS Position accuracy < 10/50 m (RMS) \\
| (1 Hz measurement updates) | (cross & along track/radial) \\
| | SGP4 Position accuracy < 1000 m (RMS) \\
| | Total onboard timing error < 0.01 sec |
Figure 3: Initial detumbling → Y-Thomson spin of -2 °/sec

Figure 4: Y-Thomson spin → Sun pointing Y-momentum control mode
Figure 5: Sun angles during Y-Thomson spin → Sun pointing Y-momentum control mode

Figure 6: Nominal 3-axis control mode: Sun pointing & Nadir pointing in eclipse
Figure 7: Agile Imaging control modes: Sun pointing/Nadir pointing/Target off-pointing & FMC-4 Imaging

Figure 8: View Finder control modes: Sun pointing → Target 1) tracking, 2) steering, 3) lock & 4) imaging
Figure 9: Reaction wheel speeds during View Finder control modes