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Conference Paper *in* IEEE Aerospace Conference Proceedings · April 2011

DOI: 10.1109/AERO.2011.5747481 · Source: IEEE Xplore

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The Development of a Software and Hardware-in-the-Loop Test System for ITU-PSAT II Nano Satellite ADCS

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Abstract— In this work, we present the operational concept of ITU-PSAT II, the reconfigurable fault-tolerant ADCS architecture and the associated Software and Hardware-in-the-Loop Test System for three-axis active control. ADCS of ITU PSAT II consists of three distinct hardware layers integrating sensors, actuators, and ADCS computer over the CAN bus. A multi mode control algorithm which acts over different operation modes and actuator / sensor failure scenarios, along with simulation results, is provided. For testing operational properties of ADCS, a Software and Hardware-in-the-Loop Test system were developed. The system includes direct satellite-bus emulation, integration of satellite hardware, an air bearing table and the Helmholtz Coil magnetic field emulator.[1 2]

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1. INTRODUCTION

ITU-PSAT II is the second student satellite project of ITU Controls and Avionics Laboratory, which aims to demonstrate on-orbit an advanced ADCS for nano-satellites (1-10 kg) with high precision three axis control needs. In this work, we present the operational concept of ITU-PSAT 2, the reconfigurable fault-tolerant ADCS architecture and the associated Software and Hardware-in-the-Loop Test System for three-axis active control.

ITU-PSAT I (Figure 1) which was developed and manufactured by same team, was launched successfully from India at 23rd of September 2009 (Figure 2). Satellite is still known to be functional up to this date. Aside from ITU ground station, its beacon signal has been received and identified by many countries (including Japan, Norway, Italy, Germany and USA) as well as amateur radio operators around Turkey. More on the development process of ITU PSAT I can be found in [1,5].

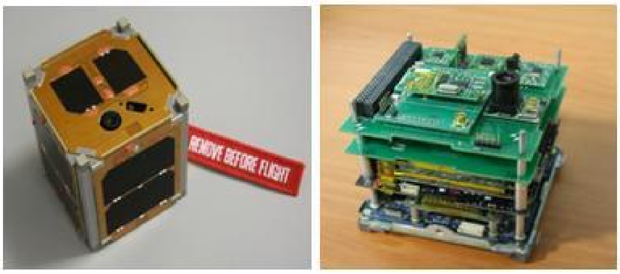

Figure 1 - ITU PSAT I launch model and its interior hardware

Nano-satellites are used and envisioned for a wide range of applications including lower-cost testing and proof-of-concept demonstration of in-development subsystems [21, 22, 23, 27], examining of GPS signals [24], providing low-cost communication links [20, 25], taking photographs [26], and for scientific purposes such as low earth radiation measurement [28], plasma density and magnetic field measurements [25]. Beside single satellite operations it is expected in near future that groups of pico-, nano- and micro-satellites (satellite constellations) will be working as a group on orbit to perform space science [29] and service missions. Satellite constellations superiority in risk distributing risk, higher target area visit frequency, distributed measurement capability and backup ability [32] make it an ideal choice for missions such as measurement of earth's magnetic field alterations [31] and providing lower-cost global communication services [30]. We refer the reader to [15-19] and [33-38] for surveys of nano-satellite applications and new nano-satellite technologies.

In order for nano-satellites to perform more capable missions such as interferometry [4], precise attitude control [4] and more capable bus designs [2,3] are needed. ITU-PSAT II is considered as a follow-on to the ITU-PSAT I project, and it aims develop and manufacture a multi-purpose bus and a 3-axis controlled platform which can be used in future missions with different payload configurations including higher-resolution imagers. In this paper we will present the Software in the Loop (SIL) and the Hardware in the Loop (HIL) testing systems developed in the Controls and Avionics Laboratories to test the

[1] 978-1-4244-7351-9/11/$26.00 ©2011 IEEE.
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functionality and the validity of our fault tolerant and reconfigurable attitude determination and control system (ADCS) of ITU PSAT-II.

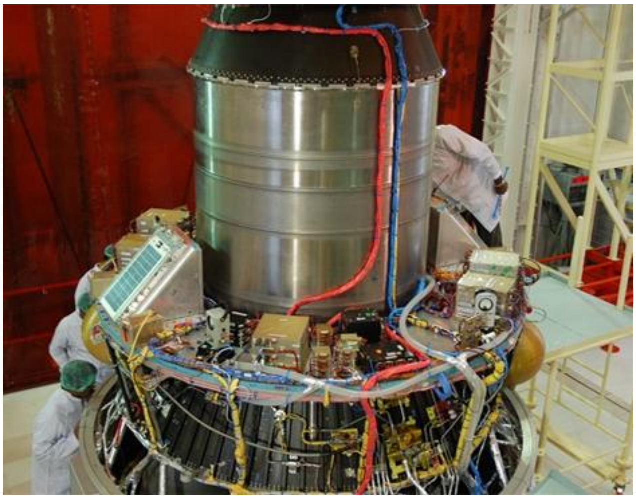

Figure 2 - ITU PSAT I is on the launch platform – compartment of payload

Preliminary Design of ITU PSAT II and Fault Tolerant and Reconfigurable ADCS

ITU PSAT II is planned to be a nano satellite of 10x10x30 cm dimension in 3U form with a maximum of 4 kg weight. It will operate on a 640-840 km altitude sun synchronous orbit. The spacecraft will have three axis active attitude control, extra power transmission via expandable solar planes and composite side panels. As a payload, it is planned to include a camera in addition to experimental attitude determination and control hardware. It is main mission is to take high resolution images (100-200 meters per pixel) in nano satellite standards and transmit them to the ITU ground stations.

Weight, power and volume budgets of the prototype model (Figure 3) are presented in Table 1.

ADCS of ITU PSAT 2 consists of three distinct hardware layers integrating sensors, actuators, and ADCS computer over the CAN bus. The sensor layer embeds a set of low-cost inertial and magnetic sensors, sun/earth sensors, a GPS receiver and an in-house developed star-tracker.

The actuator layer includes a redundant assembly of reaction wheels, magnetic torquer coils and an experimental set of uPPTs (micro pulse-plasma thrusters) with one-axis micro CMG (control moment gyro). Embedded within the ADCS Computer is a filter which allows the asynchronous fusion of filtered sensor data with outputs from the orbit and attitude propagation algorithms. Propagation algorithms simulate the spacecrafts dynamical response (including effects of disturbances, and uncertainties in sensor and actuator models) and compare it with actual sensor data to improve accuracy of determination of spacecrafts attitude and orbit position. These filtered and fused state data are fed to a fault-tolerant and reconfigurable control layer. The control layer is divided into different operation modes and associated control strategies depending not only on the actual spacecraft operation mode (such as de-tumbling or high-precision attitude control for image capturing) but also depending on the health-status of the individual sensors and the actuators.

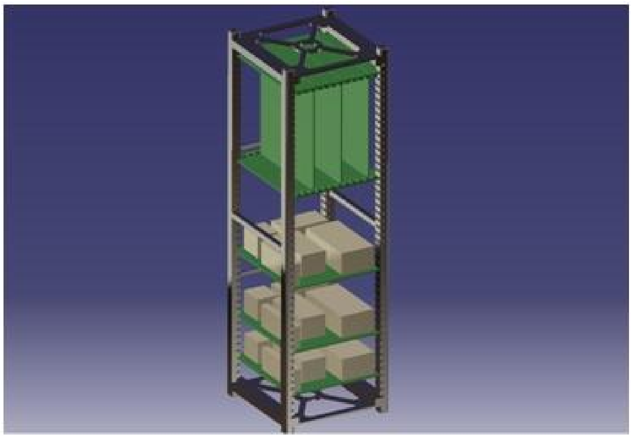

Figure 3 - Conceptual design of the prototype structural model, without camera payload

Table 1. Total Weight, Power and Volume Budget

Subsystem	Dimensions (mm)	Mass (gr)	Power (W)
OBC	80x80x40	250	0.5
EPS	90x90x70	500	2
Beacon	80x80x20	100	0.3
PLIU	25x60x80	200	0.3
ADCS	100x100x80	907	2
Communication	100x100x40	250	2
Structure	100x100x300	600	0
Panels		350	0
GPS	90x60x30	100	1
Nano Moment Control Gyro	30x30x50	150	0.3
Camera + Hardware		400	1
Star Tracker Hardware	90x90x20	150	0.4
Total		*3957*	*9.8*

Outline of HIL and SIL Testbeds

Given the failure-prone nature of space-tolerance increased COTS and in-house developed hardware elements, it becomes essential to

a) develop on-orbit software which can detect, overcome or provide graceful degradation of performance across various failures

b) test the ADCS software hardware on ground for a complex set of hardware and software failure scenarios.

For that reason, we have developed a Software and Hardware-in-the-Loop Test system to test the actual behaviour of the control system elements during execution of real mission of the spacecraft including failure scenarios.

The in-house developed S/HIL system provides high-precision orbit, attitude and environment "truth-model" propagation with 3D orbit and attitude visualization using COTS STK software. In addition, the simulator allows direct satellite-bus emulation and actual integration of spacecraft hardware and software including failures such as erroneous sensor readings, actuator failures and limited actuator operations. In the extended configuration, the ADCS system operation is tested in 3D using air bearing table and the Helmholtz Coil frame emulating the on-orbit magnetic environment. Specifically, the extended configuration of the HIL simulation system comes as close as possible to the execution of actual attitude maneuvers in the space (such as attitude transitions, de-tumbling and switching between modes), testing the whole control and filtering algorithms in hardware environment, while simulating the mechanical behaviour with actual sensor and actuator mechanisms on the air bearing table coupled with on-orbit magnetic field generating Helmholtz Coil system. This paper provides insights to iterative process of ADCS design using this specific Software and Hardware-in-the-Loop Test System.

Iterative process of ADCS design through HIL and SIL simulations are displayed in **Error! Reference source not found.**. Design cycle starts with initial design of control and estimation algorithms. Then these algorithms are tested in a nominal scenario (no actuator or sensor failure occurs) for different control modes. If SIL simulation results are not satisfactory, we return back to design step and re-evaluate the design based on simulation results. When nominal SIL scenario results are satisfactory, we move to failure SIL scenario where we induce actuator and sensor failures during simulation. If results are not satisfactory, we only revise health management algorithms since nominal control and estimation algorithms are proven to be working good at earlier step. To make sure that health management doesn't interfere with nominal conditions nominal scenario is re-done. If all SIL simulation results are satisfactory, we move to HIL simulation. If HIL results are not satisfactory, limitations of hardware or ADCS software are noted. Hardware is revised to make sure that it meets the control requirements for the mission and ADCS software is revised to incorporate existence of hardware in simulation loop.

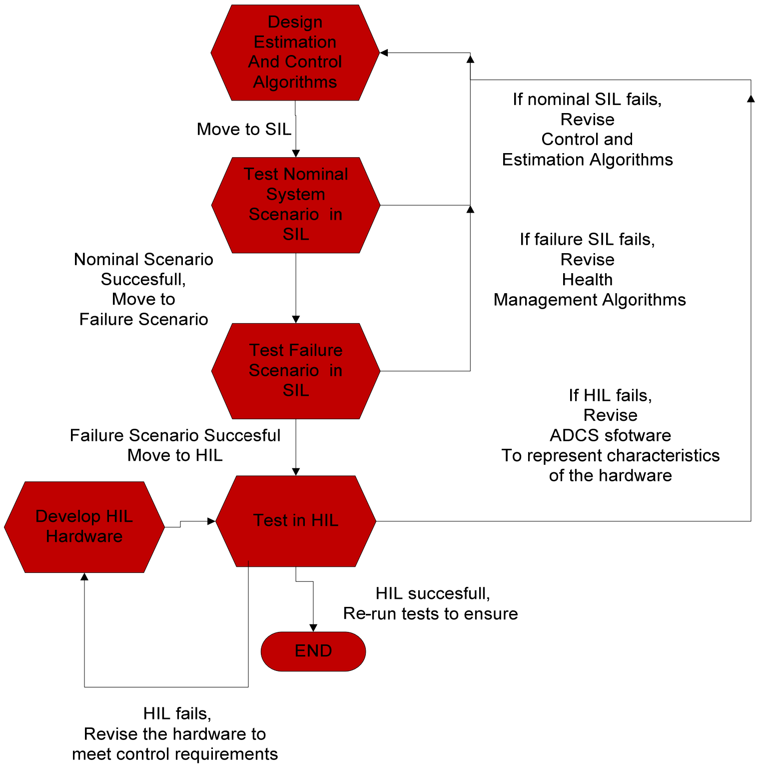

Figure 4 - Iterative ADCS Design with SIL and HIL Simulations

Structure of the Paper

Paper is structured as follows; second section details the ADCS system by providing its architecture as well as actuator and sensor hardware. Third section explains control methodology developed for the mission to achieve a fault tolerant and reconfigurable control system. Next section introduces SIL simulation setup and simulation results for nominal and failure scenarios. Fifth section explains the components of HIL simulation and experimental testbed. Final section delivers conclusions and future work.

2. ADCS ARCHITECTURE AND HARDWARE COMPONENTS

General outline of ADCS and its components are displayed in Figure 5. When taken as an iterative process, one work cycle of the system can be explained as follows: spacecrafts attitude and angular velocity changes based on its dynamics and exerted torque on it from previous cycle from actuators and environmental disturbances. Changes on attitude dynamics are inferred from sensors (which also takes measurement from space environment such as magnitude of magnetic field vector). Sensor readings are sent estimation and filtering layer.

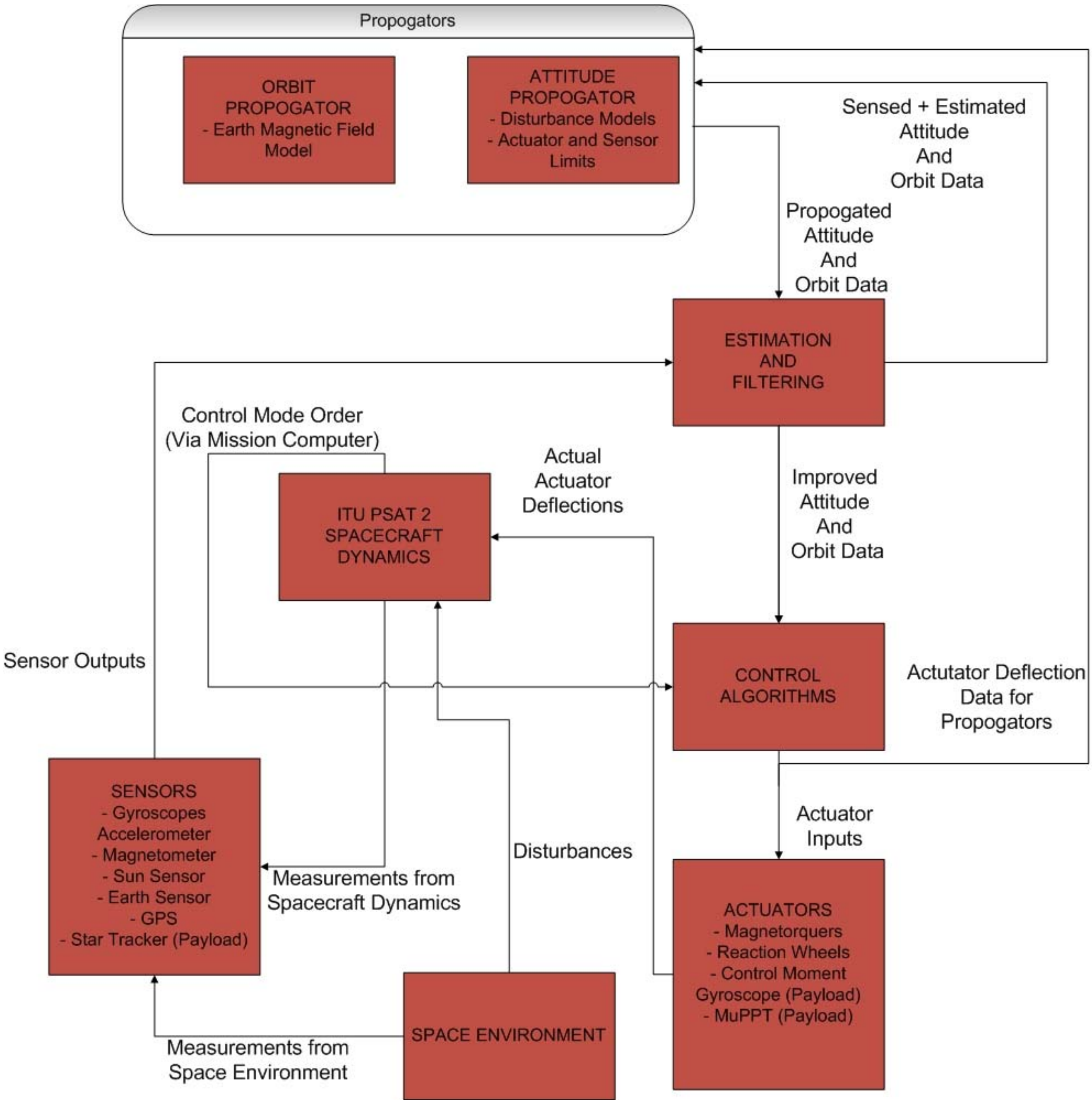

Figure 5 - ADCS Architecture

Apart from actual dynamics of the spacecraft, an orbit and attitude propagator is also included in ADCS software, which simulates full dynamics of the aircraft by integrating standard spacecraft attitude equations [6] and orbit equations with perturbations [7]. These data along with sensor readings are sent to estimation and filtering layer, where all sensor readings and propagation data is fused to generate an accurate state vector to be fed to control layer. These data is also sent back to propagation algorithms to update the current estimated state, so that they can function better at next cycle. Finally estimated states are sent to control layer, where it takes the order of current control mode from mission computer and generates commanded torque based on fault tolerant architecture which will be explained in detail in next section. Finally commanded torque is converted into actuator deflections based on actuator associated with current control mode and toque is exerted on the spacecraft; then the next cycle begins.

Sensors

Sensors on the spacecraft include low cost accelerometers and gyroscopes, earth/sun sensors and a set of magnetometers. A star tracker is also going to be included for experimental purpose. Both hardware and software of the star tracker is completely being developed in the lab.

Attitude Propagator

Attitude propagator is an in-built observer, which generates attitude data based on mathematical model of spacecraft and gets updated via estimation layer based on fused sensor data. Equations used by propagator can be found on any standard spacecraft dynamics textbook such as [6].

We have used quaternion attitude parameterization for the model and assumed rigid body dynamics with seven variables in total (four quaternions and three dimensional angular velocity vector). In addition, propagator also takes into account of disturbance torques on the spacecraft, which were modeled as Gaussian white noise, with amplitudes chosen based on estimates from various sources as displayed on Table 2. Order of magnitude of total disturbance also gives us a rough estimate on necessary torque that has to be provided by actuators.

Table 2. Estimates for Magnitude of Disturbance Torques

Disturbance	**Magnitude**	**Unit**
Aerodynamic Drag	3,6245E-08	N.m
Gravity-Gradient Drag	1,2315E-07	N.m
Solar Pressure Torque	4,9212E-09	N.m
Residual Magnetic Field Torque	1,28E-07	N.m
Factor of Safety	3	
Total Disturbances with FoS	8,7696E-07	N.m

In order to make attitude propagator more reliable, low order transfer function models of actuators were also incorporated into software.

Orbit Propagator

Orbit propagator, simulates the orbital trajectory of the vehicle based on initial conditions provided by TLE file. Propagator integrates the orbital differential equations which includes perturbation effects such as J2, in order to simulate satellites position on the orbit. Figure 6 displays the trajectory of the spacecraft based on orbital elements in inertial and Earth frames. Figure 8 shows output of the propagator; time history of orbital elements

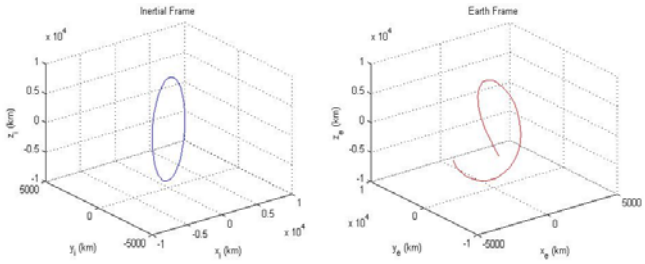

Figure 6 - Trajectory of spacecraft in inertial and Earth frames

This propagator also includes an Earth Magnetic Field model, which allows it to compute the magnetic field vector based on position of the spacecraft on the orbit. An estimate of magnetic field vector components, based on orbit propagation is shown on Figure 7.

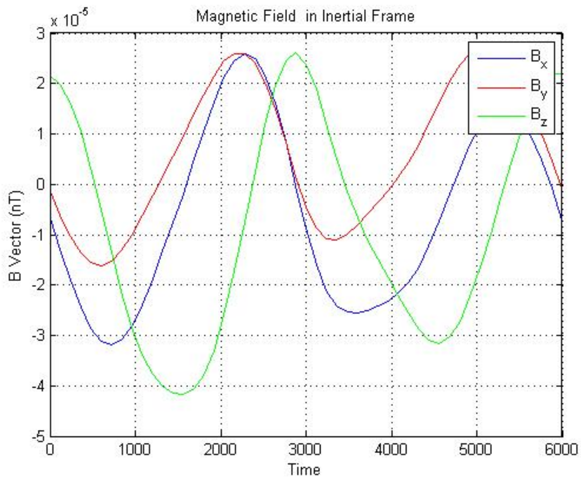

Figure 7 - Time history of magnetic field vector components

Estimation and Filtering

Principal task of this layer is to provide noise free and reliable state data to control algorithms which is achieved via fusion of readings of various sensors and improvement of estimations by using a Kalman Filter. Since propagation algorithms simulates nonlinear dynamics, an Extended Kalman Filter algorithm is used in this layer, to improve efficiency of estimator on operating regions which involve nonlinear dynamics.

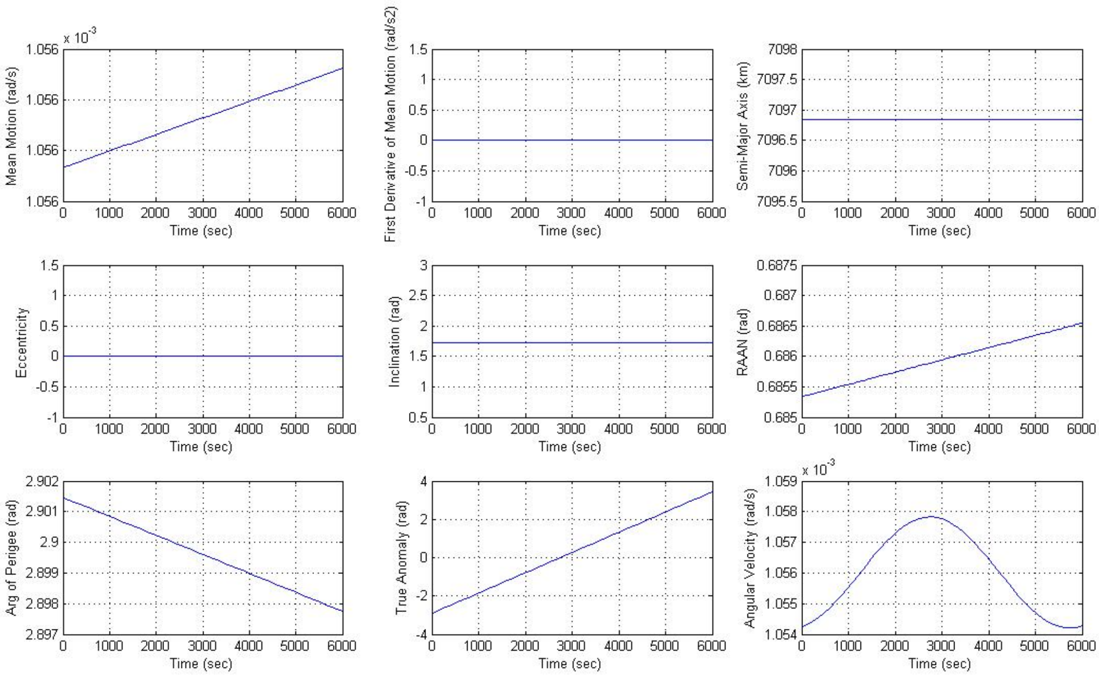

Figure 8 - Time History of Orbital Elements

Actuators

Current actuator assembly for ADCS system consists of magnetic torque generators and reaction wheels as primary torque generators. A control moment gyroscope (CMG) and uPPTs are also included in the system, but only for experimental purposes (thus can be considered as a part of the payload). Most of the actuators which are a part of HIL or going to be a part of flight model are in-lab-built.

Magnetic torquers are placed inside external panels as three torquers per axis of the spacecraft (nine in total). Design of torquers is based on disturbances from environment and required torque. Final design properties of magnetic torquers are provided in Table 3.

Table 3. Properties of Magnetic Torque Generators

Coil Area	6,40E-03	m^2
Number of Turns	2,00E+02	-
Min. Flux	2,50E-01	Amper
Min. Magnetic Field	2,00E-05	Tesla
Min. Torque	6,40E-06	Nm

Design of reaction wheels is based on requirement of spinning the spacecraft in 3˚/s, for mass and inertia values provided in table 1. Properties of wheels are given in Table 4.

Table 4. Properties of Magnetic Torque Generators

	Magnitude	Unit
Momentum Storage Capacity	1,05	mNms
Rotor Moment of Inertia	1,75E-3	kgm^2
Torque Requirement	0,233	mNm
Maximum Torque	1	mNm

Development of a Control Moment Gyroscope for nano sized satellite is still an ongoing research topic. It is desired to replace reaction wheels with moment control gyroscopes for nano sized satellites in the future, due to increased control authority and accuracy. Current CMGs designed for micro and mini sized satellites can provide slew rates between 0.1-10 ˚/s. Based on this data, desired properties for CMG to be built for ITU PSAT II is given on Table 5.

Table 5. Desired Properties of nano CMG

Property	**Magnitude**
Weight	150 g
Power Consumption	0,5 W
Torque	1 mNm
Slew Rate	0,5-1 ˚/s
Accuracy	± 10 ˚

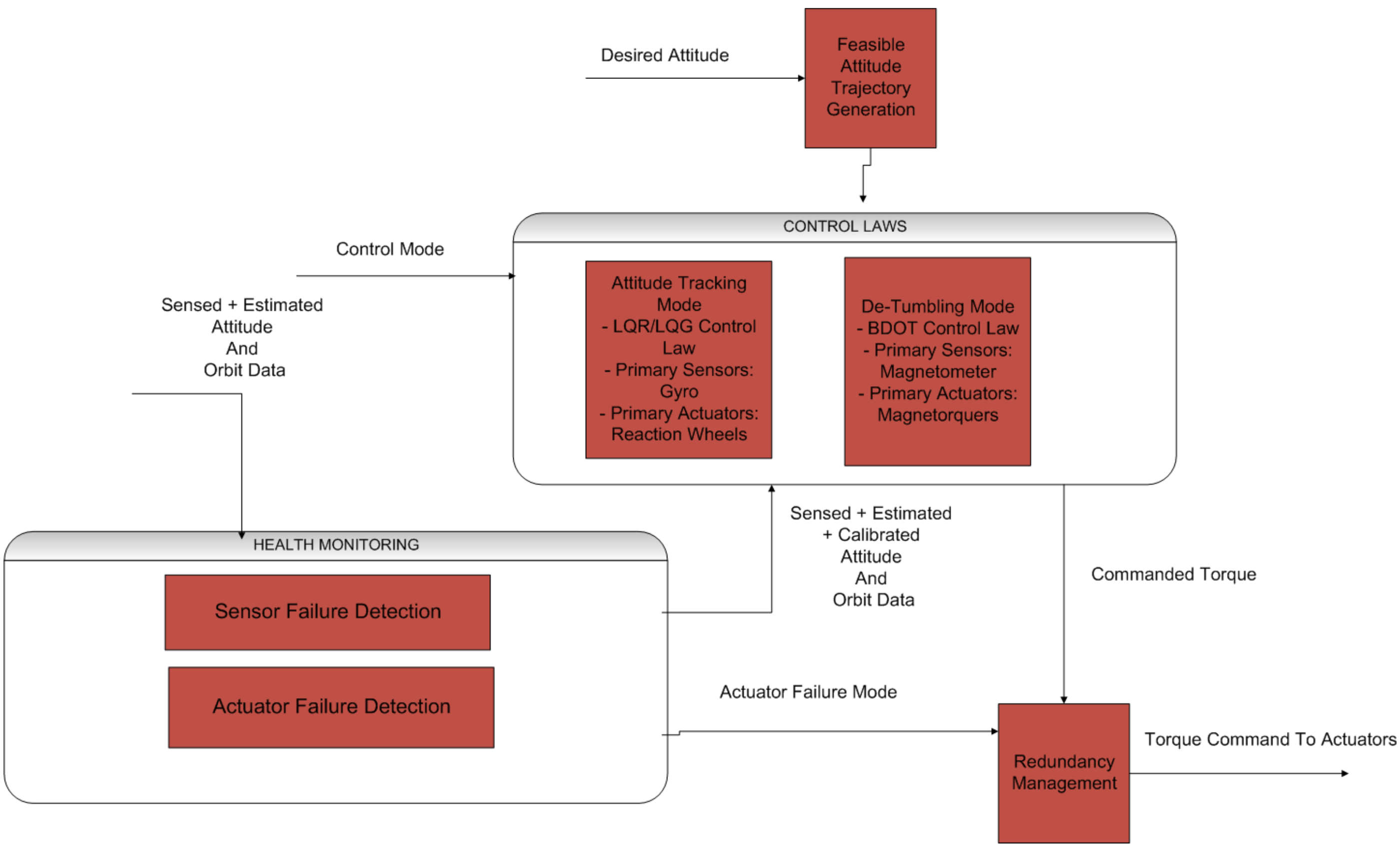

Figure 9 - Fault Tolerant and Reconfigurable Control Architecture

3. Fault Tolerant and Reconfigurable Control System

One of the main aims of the project is to develop and test precise control algorithms which have robustness against unpredicted actuator failures and/or erroneous sensor readings, beside standard properties like asymptotic stability. These topics have dealt a great interest in spacecraft control in last fifteen years. Reconfiguration of control architecture in case of actuator failure, performance degradation, sensor bias have been extensively studied in terms of adaptive and intelligent spacecraft control in works such as [8-9]. Design of control laws which take into account actuator saturation and parametric uncertainty [10] and a more extreme case of controlling spacecraft in underactuated situations [11] has also been of interest.

Based on the current literature we have blended number of these existing architectures into our system for test and implementation purposes. Control architecture of ITU PSAT II is displayed on Figure 9. Two main components of this architecture is control laws and health monitoring software.

From operational point of view, spacecraft has two control modes, De-Tumbling mode: which is activated to stabilize the spacecrafts attitude from an initial high velocity. Atttitude Tracking mode on the othe other hand, maneuvers the spacecrafts attitude from its initial position to a desired orientation.

In theoretical context it is possible to successfully design and implement these controllers, but as it is pointed out in the introduction section, due to avoid possible failures and saturation we have to introduce extra components to architecture, such as “Health Monitoring” layer. Basically, this layer inspects sensed and estimated state vector to analyze if there is a bias in the sensors (noise in sensor have already been handled by EKF in estimation and filtering layer) and if there is an failed actuator. After detection, this layer takes action toward these failures, such as calibrating the sensor bias and feed the calibrated sensor data to control layer.

Primary fault scenarios that we deal are actuator and sensor failures due to degradation of performance or complete shutdown at worst situation. Failure detection algorithms for sensor and actuator works independent of each other therefore it is possible to deal with a multiple failure scenario where sensor and actuator failure happens at the same time.

Due to avoid severe actuator failures all primary actuators in spacecraft are assembled in a redundant manner. Thus total required torque computed by control layers have to be distributed among redundant actuators, which is handled by redundancy management. This is also when reconfigurable control action comes in; whenever an actuator failure is detected, redundancy management is informed by health management layer to change its configuration, so that no torque is sent to failed actuator anymore.

Finally, to avoid actuator saturation, which is likely to be in attitude tracking mode for large errors between desired and current dynamics; an feasible attitude trajectory generator is added to architecture to convert a simple attitude command to a sequence of attitude waypoints for smoother tracking.

These layers are explained in more detail below:

De-Tumbling Mode

Especially after initial launch phase, spacecraft begins to spin in all directions with relatively high angular velocity, therefore have to be stabilized before entering into mission phase. Although reaction wheels can be used for this purpose, high accuracy is not needed and wheels are likely to saturate for under high angular velocities.

For this reason we prefer to use a B-DOT controller, which extracts the magnetic field vector as observed from spacecraft frame from magnetometer readings, takes it derivative and feed it to Magnetorquers to generate a torque, which will derive the derivative of magnetic field vector to zero. Since magnetic field vectors magnitude can be assumed to be constant during stabilization phase, driving its derivative to zero results in driving spacecrafts angular velocity to zero. Application of B-DOT algorithm to a cube sat mission can be found on [12].

Attitude Command Tracking Mode

When spacecraft is stabilized around any attitude (via De-Tumbling Mode) then it cane pointed into any direction by using this mode. Basically this mode makes use of linearized spacecraft equations and applies an optimal control method by minimizing a linear quadratic performance index (LQR). Since estimation layer includes a Kalman Filter, overall control scheme can be considered as a LQG controller. These types of control algorithms have guaranteed robustness margins and moreover, actuator saturation constraints can be embedded into performance index to avoid them [13].

This control algorithm uses full state feedback, and makes use of redundant assembly of reaction wheels to generate desired torque. When used in vicinity of trim point controller performs well, but actuator saturation becomes harder to handle when large maneuvers are requested.

Sensor Failure Detection

Sensor readings degraded by noise is handled by sensing and estimation layer in Figure 5, as well as biased readings. These kinds of biases are likely to happen in accelerometers and gyros, which must be handled online in order to avoid sending false state data to control algorithms.

Filtering and estimation layer achieves bias estimation, by comparing sensed state data and state data estimated by estimation layer. It generates an initial bias estimation then updates this estimate based on difference between state vectors generated by sensor data and estimation layer. More on convergence of this technique and its details can be found on references listed on the beginning of the section.

Another possible scenario is failure of sensors (either turning off or giving highly erroneous data) during mission, rendering some of the sensor useless. Sensor failure detection layer in Figure 9 identifies the failed sensor, and isolates it from the estimation layer such that output of the failed sensor doesn't affect the performance of the estimation layer.

Actuator Failure Detection

Actuator failure is possible to detect in two ways. One ways is to continuously monitor health of the actuators by hardware, and confirm it is working. If it is not possible a Multiple Model Switching Scheme (which is explained in detail in [14]) can be used to detect failures.

Main scheme of failure detection is shown on Figure 10. Principle of this detection algorithm is to generate state data based on all possible actuator failure scenarios (very similar to attitude propagation), and compare it with the current state data, to generate a performance index (which is integrated over time). When performance index associated with that mode is small compared to others, actual model is more likely to be in that failure mode. When minimum is taken over all possible failure data, the argument which minimizes the performance index indicated the failure mode of the actual system. If nominal model appears to minimize it, system continues to perform in full configuration. Whenever a failure mode minimizes the index, failure is detected and redundancy management is informed.

Since an underactuated spacecraft is very difficult to control precisely, redundant numbers of actuators are usually found in spacecrafts to take caution against actuator failures. For instance, in our system we use 4 reaction wheels, although 3 of them is enough to stabilize the spacecraft around any desired attitude.

Since all control algorithms on Figure 9, generate a 3 dimensional commanded torque Redundancy management (sometimes referred as control allocation layer) have to distribute them among redundant actuators. This is most simply achieved by using pseudo inverse matrices.

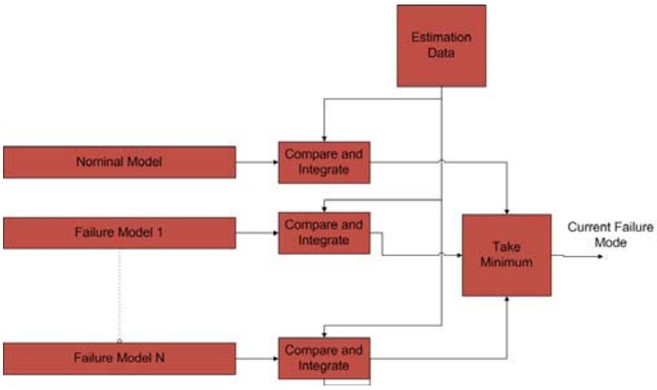

Figure 10 - Multiple Model Failure Identification

Actuator Failure and Redundancy Management

Whenever there is a an actuator failure, failure detection layers inform redundancy management and redundancy management reallocates control by discarding the failed actuator by using a different set of pseudo inverse matrices. Thus overall control layer reconfigures itself under actuation failure

Actuator Saturation and Feasible Trajectory Generation

Since LQR/LQG control of attitude tracking mode depends on linear model, its performance degrades under high angular velocities and large attitude errors. This is most likely to happen when a large attitude maneuver is commanded and spacecraft is forced to converge rapidly to reference point, which usually saturates the reaction wheels.

One way to avoid the saturation is to decompose large attitude command into a sequence of smaller magnitude attitude commands, then drive the spacecraft to final reference by taking each attitude waypoint at a time. An example is displayed in Figure 11. Spacecraft is commanded to translate to desired attitude (given in bottom right position) from its initial position (given in top right position). Instead of trying to go directly commanded attitude, spacecraft is forced to follow intermediate points between initial and desired attitude. These waypoints get updated whenever spacecraft gets near to them, thus overall tracking motion is not time based but event based.

4. SOFTWARE IN THE LOOP (SIL) SIMULATION

Before starting to integrate sensor and actuator hardware into simulation environment, control algorithms must be tested on software environment. Particular difficulty with this situation is how to make a develop a "truth model" which will act as the actual state of the spacecraft in the space, and will be tried to estimate as good as possible by filtering layer and sensor emulations. It is pointless to use models in attitude and orbit propagators as truth models,

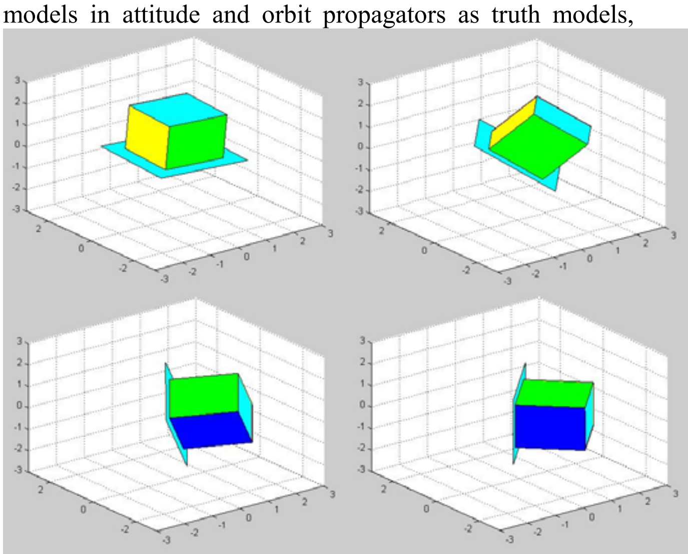

Figure 11 - Rotation sequence for large angle attitude maneuver

since this situation will not present any interesting results due to indifference between estimated and actual states other than disturbance. For this purpose we incorporate a commercial space mission simulator "Satellite Toolkit" (STK) developed by Analytical Graphics Inc. into our simulation setup. Satellite Tool Kit is widely used software for developing and analyzing virtual space missions. STK can be used for calculating contact time between ground and satellite, determining critical vectors such as magnetic field vector and determining other attitude information rapidly by communicating with a MATLAB/Simulink model [19].STK serves as a truth model since it takes into account disturbances and perturbations from large number of sources and uses mathematical models far more complex than the ones we used in propagation algorithms. In addition since it is possible to extract the actual state data from STK we can see how well estimation layer works.

Components

Components of SIL simulation is shown on Figure 12. A cycle of simulation can be explained as follows: ADCS software (including propagation, filtering and control algorithms) is provided either by an interface in STK Rack (INT) or by some outside source (displayed as laptop in the Figure 12), such that anyone with and ACDS software can connect to simulation setup and test his/her algorithms. Software receives state update from STK over UDP channel, sensor emulation (such as noisy readings, magnetometer calibration errors, gyro biases) are applied to state vector to simulate actual sensor reading during mission.

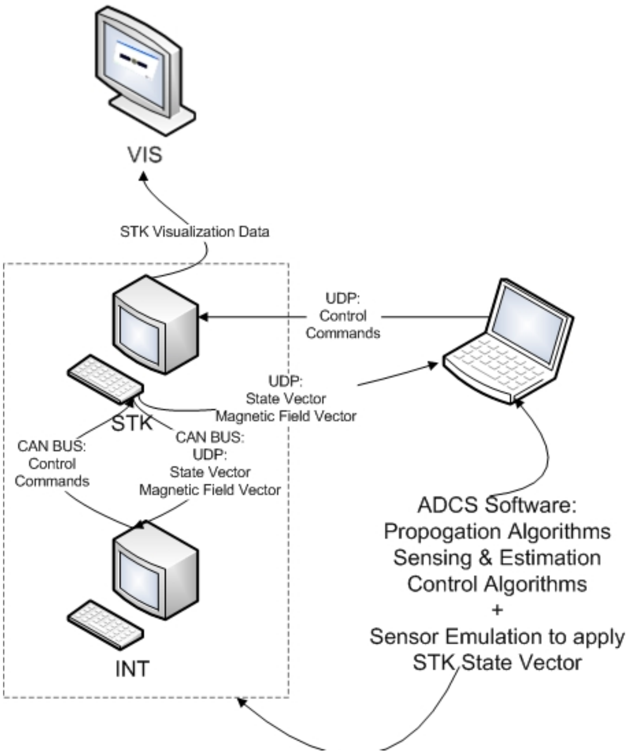

Figure 12 - Overview of SIL Simulation Setup

Based on these data ADCS software applies the filtering and control architecture detailed in second section to generate commanded torques to spacecraft and sent (via CAN Bus in INT, via UDP in outside source) to STK software to simulate its effect on spacecraft model. Communication between ADCS software (which is modeled in MATLAB Simulink) and STK is achieved through another Simulink model in STK computer, which also adjusts the synchronicity between two software. Finally STK open a special 3D visualization window to observe attitude and orbit of the test spacecraft and this visualization is displayed via a projector.

STK screenshots from an SIL simulation is displayed in Figure 13 and Figure 14.

Nominal Scenario SIL Simulation Results

We consider two different scenarios for simulations in SIL. First one assumes that no actuator or sensor bias occurs and system functions healthy during mission. This simulation helps us to see if proposed control and filtering algorithms performs well.

For De-tumbling mode, we assume high initial angular velocity (0.5 rad/sec in y-axis 0.25 rad/sec in others), and control mode is De-tumbling mode. Simulation results are displayed in Figure 15.

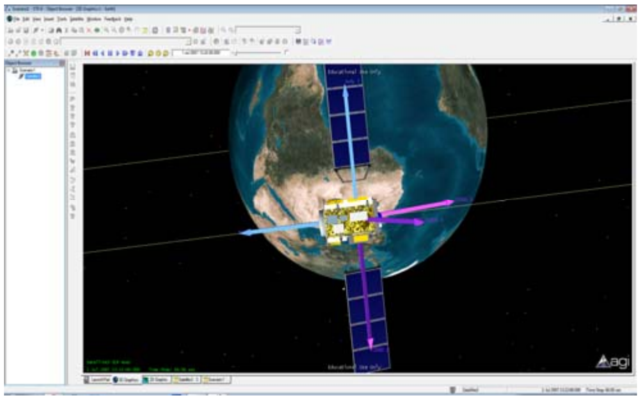

Figure 13 - Satellite model in STK

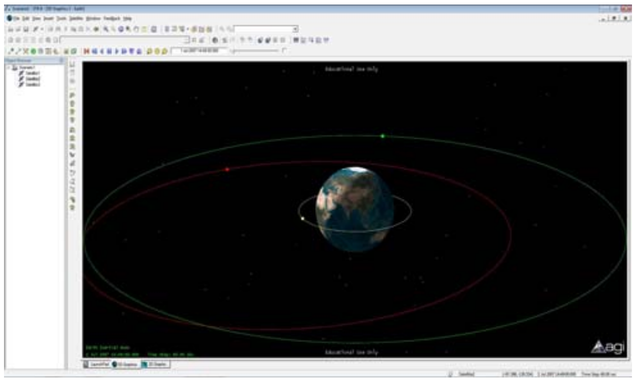

Figure 14 - Spacecraft Orbit in STK

Based on Figure 15, nominal scenario for De-tumbling mode seems satisfactory. Angular velocity history shows that spacecraft has been stabilized on all three axes, around 500 seconds, and derivative of magnetic field vector has been driven near to zero (note that perfect stabilization is not possible due disturbances acting on the spacecraft and sensor noise). To show the feasibility of the control law, Figure 15 also contains commanded torque to spacecraft and corresponding current requested from Magnetorquers. Since inputs are in feasible limits and time response is satisfactory, we conclude nominal SIL simulation for De-tumbling mode.

For nominal scenario on attitude tracking mode, we consider maneuvering from one attitude to another which involves a large difference in terms attitude angles, which was already displayed in Figure 11. Simulation results are displayed in Figure 16.

Based on Figure 16 overall performance of Attitude Tracking controller for nominal scenario is satisfying. Top left figure shows attitude tracking error in angles, which gets updated whenever spacecraft is sufficiently close to target waypoint, when it reaches the final waypoint (which was ordered by mission computer) it stops and stays there. Performance of Extended Kalman filter is shown on top right figure in terms of error quaternion between spacecrafts actual quaternion and estimated quaternion, error quaternion equal to unit vector means that estimation algorithm worked

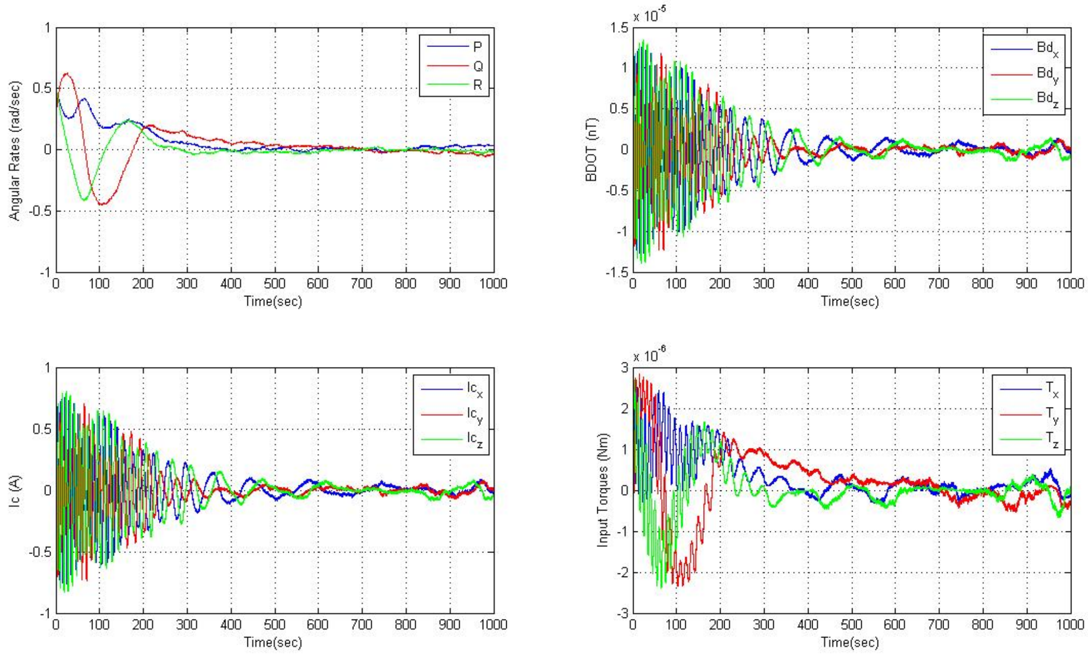

Figure 15 - Nominal Scenario SIL Performance of De-Tumbling Mode

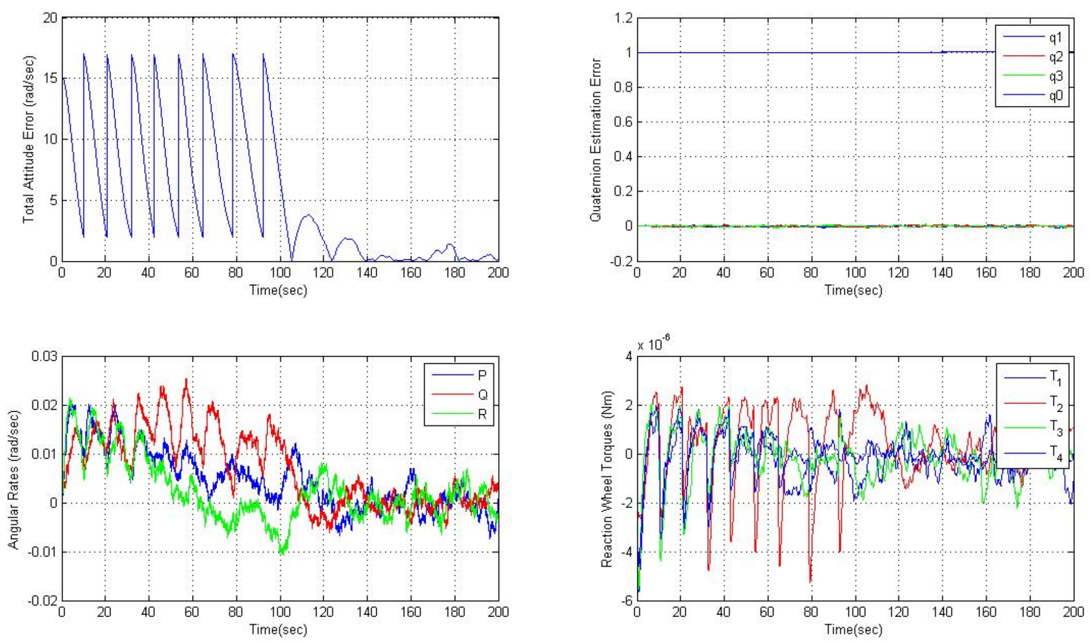

Figure 16 - Nominal Scenario SIL Performance of Attitude Tracking Mode

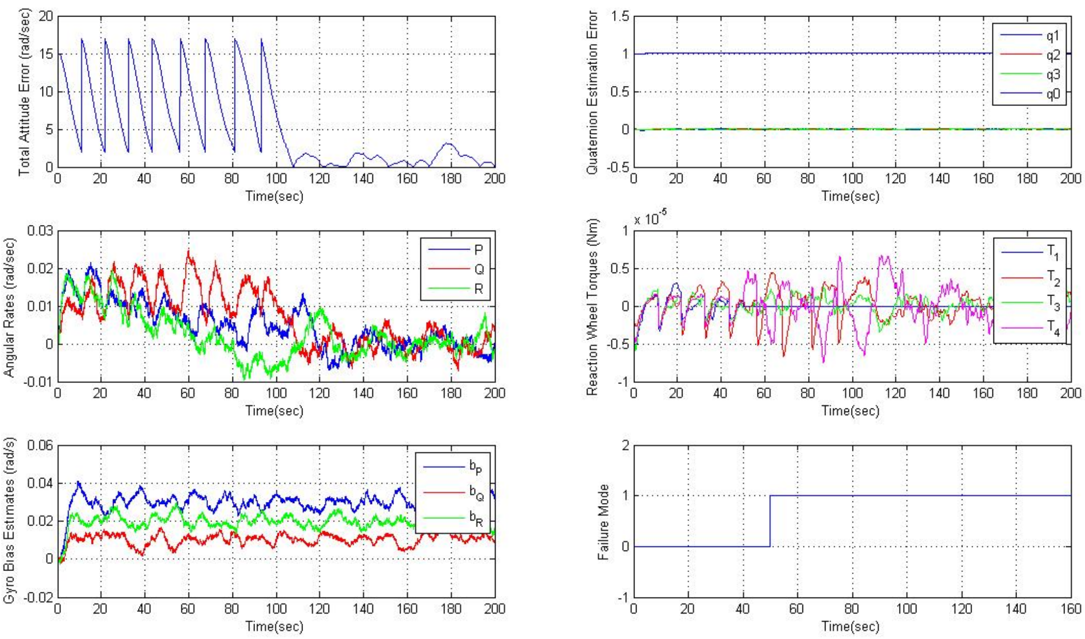

Figure 17 - Failure Scenario SIL Performance of Attitude Tracking Mode

well and converged to actual state of the spacecraft. Bottom left figure shows time history of angular velocity and bottom right figure shows torque outputs of four reaction wheels. None of the wheels have saturated, mainly due to redundant control and feasible attitude trajectory generation, which decomposed the maneuver into smaller segments.

Failure Scenario SIL Simulation Results

To test the performance of health management system we consider a failure scenario for Attitude Tracking mode. Control objective is same as in nominal case, but gyroscopes readings are biased and one of the reaction wheels fails during the operation. Gyro biases are 0.03, 0.01 and 0.02 rad/s for x,y and z axes respectively, at 50^{th} second of the simulation first reaction wheel fails. Simulation results are given in Figure 17.

Health management parameters are given at the bottom of the figure. Bottom left figure shows sensor bias estimates, which are very close to actual estimates but a bit blurred due to noisy sensor readings. Actuator failure at 50^{th} second is also identified successfully by fault detection layer, as we see failure mode jumps from 0 to 1 (where 0 means nominal, 1 means reaction wheel 1 failed) on bottom right figure. Due to rapid reaction of reconfigurable control system, tracking error and estimation errors are almost identical to nominal scenario. We conclude that health management layers are competent for SIL simulation and move to HIL simulations.

5. HARDWARE IN THE LOOP SIMULATION

As displayed in **Error! Reference source not found.**, our experience on SIL simulations with iterative design resulted on control estimation and health management algorithms which are suitable for a nano sized satellite with three axis precise attitude control. In last step actual hardware (sensor package, actuators, mission computer) have to be integrated into simulation environment along with experimental platforms which will simulate the effects of space environment.

General diagram of HIL simulation is given on Figure 18. This Rack system consists of two rack PCs, one optional monitor which may display simulation of satellite and satellite hardware simulation system providing real responses to data which are: S/C, Environment, orbit and attitude propagation and emulations of bus and sensor, retrieved by interface rack running MATLAB Simulink model. Interface Rack generates data updated by satellite hardware simulation system and STK Rack generates position and magnetic field values as in SIL (using another MATLAB Simulink model) as response and also displays satellites movements.

For simulating space environment, two primary components were integrated into simulation. First one is a in-lab-built frictionless air bearing table, to simulate spacecrafts attitude dynamics and seconds one is Helmholtz Coil system for simulating Earth's magnetic field on the orbit of the spacecraft. These coils are able to provide a homogenous magnetic field around spacecraft. When combined with air bearing table, spacecraft should be able stabilize itself using magnetometers to sense the field created by the Helmholtz Coils, and stabilize itself on the air bearing table via exerting torque with magnetorquers.

Table and coils integrated together is shown on Figure 19. Whole HIL system along with STK visualization via projection is displayed in Figure 20.

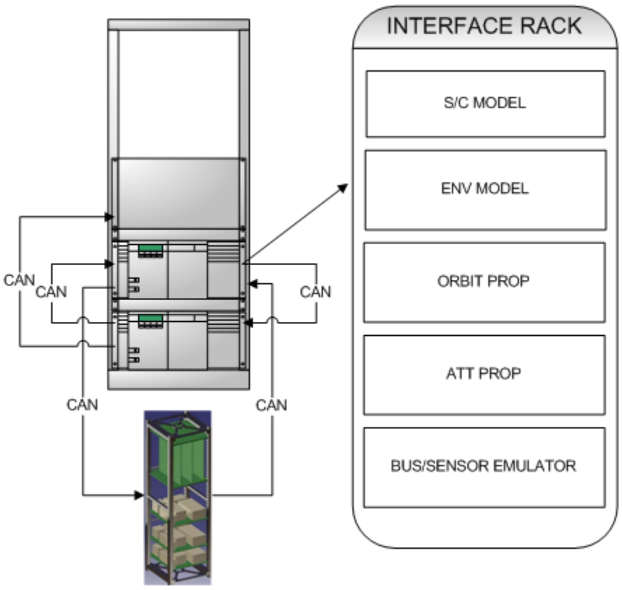

Figure 18 - Overview of HIL Simulation Setup

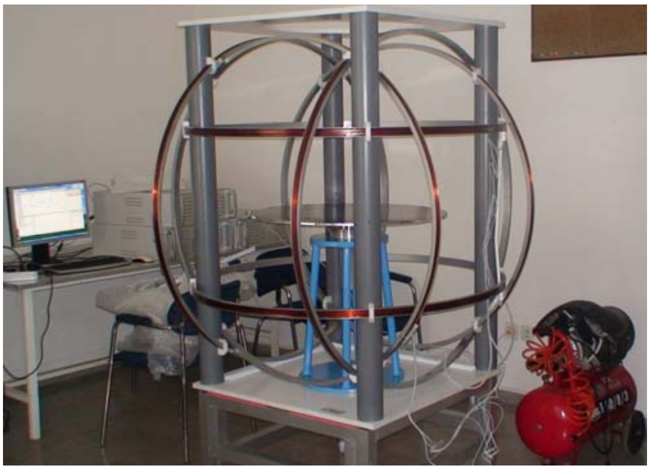

Figure 19 - Integrated Air Bearing Table and Helmholtz Coil Platform in HIL Simulation

Figure 20 - HIL Simulation and STK Visualization

6. CONCLUSION

In this work, we have introduced ITU PSAT II, a nano satellite project with full 3 axis control. We have provided details for the ADCS software and hardware, emphasizing fault tolerant and reconfigurable properties of the architecture. An ADCS design process which involves SIL and HIL simulation steps were displayed, and existing architecture is verified by SIL simulations under both nominal and failure scenarios. Future work involves testing the control, estimation and health management algorithms in HIL environment.

ACKNOWLEDGEMENTS

This work is supported by TUBITAK (The Scientific and Technological Research Council of Turkey) 108M523 Project.

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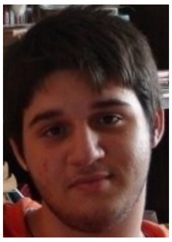

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