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Modular Attitude Control System for Microsatellites with Stringent Pointing Requirements

Dr. Simon C. O. Grocott
Manager, Attitude Control Systems, Dynacon Enterprises Limited
3565 Nashua Drive, Mississauga, Ontario, Canada L4V 1R1
(905)672-8828 x223, scog@dynacon.ca

Abstract. Advancing technology has allowed for the development of low cost attitude control hardware for microsatellites. However, the attitude control design and software development remain a significant cost driver. The Dynacon High Performance Attitude Control system is a modular control system that makes use of reusable algorithm modules enabling the attitude control system to be applied to several different spacecraft missions with very different performance requirements. The High Performance Attitude Control system is described in this paper, and simulation results are shown for a variety of spacecraft.

Introduction

Microsatellites have not typically been used for scientific missions that have moderate to high pointing accuracy requirements. Cost has been a major driver in using passive control systems such as gravity gradient stabilization and hysteric damping. However, with the advent of microsatellite sized (and priced) attitude control hardware, high accuracy three axis attitude stabilization is not beyond the means of microsatellite missions when modular, reusable control system software is available.

Three existing missions that require moderate to high accuracy three axis pointing are FedSat ([Australian] Federation Satellite), CHIPSat (Cosmic Hot Interstellar Plasma Spectrometer), and MOST (Microvariability and Oscillations of Stars). Each of these missions is using the High Performance Attitude Control system (HPAC) developed by Dynacon for precision three axis stabilization. A further proposed mission, Near Earth Space Surveillance (NESS)[1] has also baselined the HPAC system for its attitude control.

FedSat is an Australian research micro-satellite being built by the Co-operative Research Centre for Satellite Systems (CRCSS) whose constituents include Australian Universities, Government Research Organizations and Industry. Its primary scientific mission is Earth magnetometry. In order to meet its scientific mission requirements, it requires pointing accuracy of 4 degrees in each axis about a nominally earth pointing attitude. The attitude control hardware manifested consists of three magnetorquers, three reaction wheels, three solid state rate gyros, a three axis magnetometer, and three digital sun sensors.

CHIPSat is a NASA University Class Explorer (UNEX) mission centered at the University of California, Berkeley. Its scientific mission is to map the cloud of hot interstellar gas that surrounds our Solar System through a process of extreme ultraviolet spectroscopy. In order to meet its scientific goals it requires pointing accuracy of 8 degrees in roll, 4 degrees in yaw and 2 degrees in pitch. CHIPSat is an inertially pointing satellite. It maps the sky by pointing inertially in 316 different directions that allow the instrument field of view to span the sky. In any given orbit it must rapidly slew between 2-3 separate target directions to enable the satellite to cover the sky within the 18-month mission. Thus the attitude control is characterized by inertial pointing and rapid slewing. The attitude control hardware manifested consists of four reaction wheels, three magnetorquers, one three-axis magnetometer, four solid state rate gyros, a 22 degree half-angle field of view sun sensor, and a 15 degree half-angle field of view lunar sensor.

MOST is Canada's first space telescope. It is being developed for the Canadian Space Agency by Dynacon in conjunction with the Space Fight Laboratory at the University of Toronto Institute for Aerospace Studies, and the Department of Astronomy at the University of British Columbia. The scientific objective of the satellite is to perform very high accuracy photometric measurement of the variability in intensity of nearby stars.

In order to meet its scientific goals, it has very tight pointing requirements. It requires 25 arcseconds in pitch and yaw, and 1 degree in roll. MOST is also an inertially pointing satellite. It achieves its scientific requirements by pointing for very long duration (up to 40 days) at the same target. Thus the MOST mission has a very different level of pointing requirements from FedSat and CHIPSat. The attitude control hardware manifested consists of three reaction wheels, two magnetorquers, three solid state rate gyros, two three axis magnetometers, and two 57 degree half-angle field of view sun sensors. The most important piece of attitude control hardware, however, is the instrument itself. Sharing the focal plane with the science CCD is an ACS CCD that is used to create a co-boresighted star-tracker. The star-tracker has a very narrow field of view (1deg) as well as a low data rate (1 Hz), and therefore the attitude control contains a coarse pointing mode in which the star-tracker is not used, and a fine pointing mode in which it is.

These three missions have some important similarities, and important differences. The attitude control hardware is very similar for each mission. Each relies on reaction wheels for primary attitude control torque. Each uses magnetorquers to provide desaturation torque. They also rely on a core sensor suite of magnetometer, sun sensor and rate gyros. However, their objectives vary substantially; FedSat is an earth pointing satellite while MOST and CHIPSat have inertial pointing objectives. Furthermore MOST has very significantly more stringent pointing requirements. Because the attitude control requirements are quite different for these three missions, it is not unexpected that the attitude control algorithms and software would be very different. However, with a modular control approach it has been possible to use the Dynacon High Performance Attitude Control system (HPAC) for all three of these missions.

Algorithm Modules

HPAC is composed of a small number of algorithm modules. The modules themselves serve distinct functions in the control of microsatellites. For any particular mission, not all the modules may be necessary.

A block diagram of the logic flow through the control modules that make up the HPAC system is shown in Figure 1. There are seven fundamental building blocks to the HPAC system. These are: a mode executive, a detumble algorithm, ephemeris models, an Extended Kalman Filter based attitude

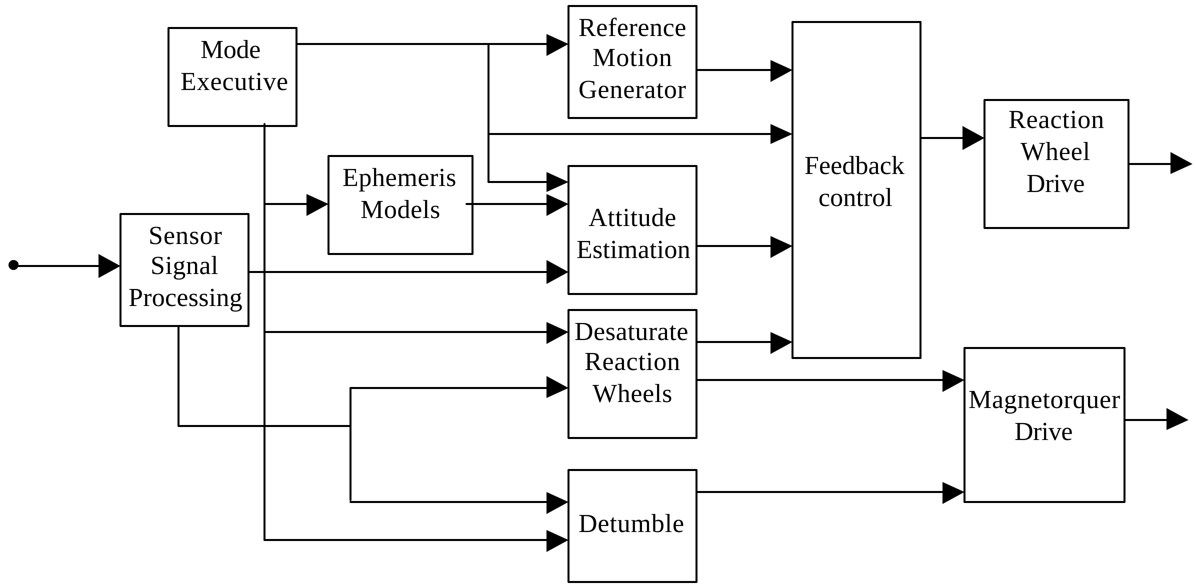

Figure 1 HPAC Modules

estimation module, a reference motion generator, a feedback control module, and a desaturation module.

The mode executive serves the role of coordinating the various modules and serving as a command interface to the system. This module ensures that the correct software modules are executed in the correct sequence. Of all the control modules this tends to vary the most from mission to mission due to the differing nature of the ACS commands that are required for a given mission.

The objective of the detumble module is to reduce the angular momentum in the satellite to a level that can be absorbed by the spacecraft reaction wheels. The detumble operation takes the spacecraft from an arbitrary initial velocity (subject to a 40 deg/sec limit) to near zero velocities by executing a B-dot detumble algorithm[2]**.** From magnetometer measurements, the rate of change of the magnetic field (B-dot) is calculated. Magnetorquer commands are generated that are proportional to B-dot. The resulting torque on the spacecraft slows the spacecraft relative to the Earth's magnetic field. This algorithm relies only on measurements of the magnetic field; no knowledge of the Earth's magnetic field is necessary to detumble the spacecraft.

Next, ephemeris models are needed to provide inertial reference to a variety of sensor signals. For example a solar ephemeris model is needed to determine the inertial direction of the sun. If the sole purpose of using a sun sensor is to point the spacecraft or a portion of the spacecraft towards the sun, this is not necessary. However, if the solar vector is being used as a reference to determine three axis attitude information, then it is necessary to place the solar vector in inertial space, and hence an ephemeris model is necessary.

Ephemeris models are necessary for a number of other sensors as well. Three axis attitude information can be extracted from magnetometer measurements. This requires knowledge of the expected magnetic field direction at the spacecraft location, which in turn requires an ephemeris model for the spacecraft orbit. Using these models and given an absolute time reference, the location of the satellite in its orbit can be determined. From this its location over the Earth can be determined. Knowing the location and altitude over the Earth, the magnetic field magnitude and direction is determined from the magnetic field model.

The CHIPSat mission makes frequent use of lunar observations and therefore has manifested a lunar sensor. Three axis inertial information is obtained from the lunar sensor through the use of a lunar ephemeris model.

All of these ephemeris models serve the role of providing an expected sensor output to which a sensor measurement can be compared. This information flows directly into the attitude estimation module.

The attitude estimation module fuses all the attitude sensor measurements together to provide an overall estimate of the spacecraft attitude. For this purpose an Extended Kalman Filter is used. The Extended Kalman Filter is the focus of the next section and is described in more detail there.

The reference motion generator is present to set a target orientation trajectory for the spacecraft. This may be to simply hold an attitude specified by an attitude quaternion, or it may be a trajectory for the spacecraft to slew from one orientation to another. The objective is to produce a target orientation that the control loop can follow.

Having created a trajectory for the spacecraft to follow, and having an estimate of the spacecraft attitude, all the necessary inputs are available to produce a servo control for the satellite. This is the role of the feedback control module and allows the spacecraft to follow the trajectory that has been generated.

Finally, assuming that reaction wheels are used for momentum storage, a module is available which provides for using magnetorquers to produce torques that will desaturate the wheels.

The emphasis of this paper is on the attitude estimation portion of the ACS modules and is the focus of the next section

Attitude Estimation

Attitude estimation is highly nonlinear due to the inherent nonlinearity in rotational kinematics, spacecraft attitude dynamics, and the nature of information that is available through common attitude control system sensors such as magnetometers. For this reason it is necessary to use nonlinear estimation processes in order to obtain estimates of the attitude of a spacecraft. The attitude estimation module that is used in the Dynacon HPAC system is based on an Extended Kalman Filter[3,4] (EKF).

The Extended Kalman filter is an extension of the linear Kalman filtering process that involves applying a linear Kalman filter to the linearization of a nonlinear process about the then current best estimate of the states. There are essentially four steps that are involved, state propagation, covariance propagation, state update, and covariance update.

During the state propagation process, the full nonlinear dynamics of the system are propagated using the best estimate of the state and a nonlinear model of the dynamics. Numerical integration is used to step the state estimate forward in time to the next time period for which measurement data is available.

Similarly, covariance propagation involves numerically integrating the state estimation error covariance through the same time period. In practice, the covariance propagation step can be implemented by calculating an approximation to the state transition matrix and propagating the discrete time Lyapunov equation.

Both the state and covariance propagation steps are independent of the sensor suite that is used to estimate the attitude. Therefore from mission to mission this code is essentially identical, having sufficient parametrization to accommodate the differences in the spacecraft dynamics between missions.

The remainder of the EKF algorithm is dependent on the sensors that are available for attitude information. The update step involves the linearization of the measurement equation, which relates the measurements to the states of the system, about the propagated state estimate. From the linearized measurement equation and the propagated state error covariance matrix, the Kalman filter gain is produced. In the state update step, the Kalman filter gain is multiplied by the innovations of the system to produce an update to the state estimate. The innovations, the difference between predicted measurements and actual measurements, rely on the ephemeris models that were discussed in the previous section. These ephemeris models are combined with the state estimates to produce an expected measurement.

Finally, the Kalman gain is applied to update the covariance matrix.

It is important to note that the Kalman gain is a function of the sensors that are available, but it is not necessary to have all sensor information at all times. The state and covariance propagation steps in no way rely on measurement information. Therefore the state and covariance can be propagated regardless of sensor availability. In the extreme this allows the measurement process to be asynchronous. The Kalman gain can be calculated whenever a sensor measurement is available, and can then result in an update of the state and error covariance. In practice measurement updates are required frequently to allow a moderate bandwidth in the control system. However, the asynchronous nature of the update step allows for a multirate update and is in no way impaired by sensor drop out. This means that a variety of sensors can be sampled at different rates and their information is naturally fused. Sensor drop out can occur for a number of reasons, not all of which are unexpected. For example, sun sensor information is not valid when the sun is not in the field of view of the sun sensor either because the spacecraft is not pointing close enough to the sun or the spacecraft is eclipsed by the Earth or possibly the Moon. If sensor drop out occurs, the EKF algorithm can march along without its input.

Simulations

To demonstrate the effectiveness of the EKF-based attitude estimation, simulation results for CHIPSat and MOST will be shown. Two simulations are shown, in particular, to demonstrate key features of the attitude estimator.

The first simulation looks at the accuracy of the coarse pointing mode for HPAC. 3000 seconds, roughly half an orbit, is simulated for the CHIPSat mission that takes the spacecraft from sunlight, through an eclipse and back into sunlight. This simulation is designed to demonstrate operability of the attitude control system through the transition from sunlight to eclipse. The attitude control torques are applied by three reaction wheels. The attitude sensor suite consists of the two axis sun sensor, three axis magnetometer, and three single axis rate gyros. During the sunlit portion of the orbit, all three sets of sensors are used to provide attitude information. When the eclipse is entered, a sun presence indicator in the sun sensor shows that the sun sensor data is invalid, and the sun sensor data is not used in the Extended Kalman Filter. When the sun reappears, the sun sensor again produces valid data, and its information is reincorporated into the EKF. Throughout the simulation, the attitude states and rate gyro bias are estimated.

Figure 2 shows the pointing error for the satellite through the simulation. The solid line shows the Yaw axis pointing error; the dashed line shows the Roll axis pointing error, while the dash-dot line shows the Pitch axis pointing error. The simulation was initialized with a 20 degree estimation error in the Pitch axis. The EKF solution converges very quickly and the feedback control is able to control the pointing error to within 1 degree within 60 seconds. During the first sunlit portion of the simulation (0-500 seconds), the pointing error is controlled to within 0.1 degrees on each of the three axes. In this simulation the pointing error is governed by the noise levels in the sensors which are assumed to be 0.14 deg RMS from the sun sensor and 20 nT RMS from the magnetometer.

The eclipse is entered shortly after 500 seconds. For a short period of time, the pointing error remains very small, but by 700 seconds, the attitude error begins to increase on all three axes. Throughout the remainder of the eclipse period the attitude error on each of the three axes is kept to within 0.6 deg, though the direction of the error changes throughout. The direction of the error is not random. It has a very precise and well known direction. At

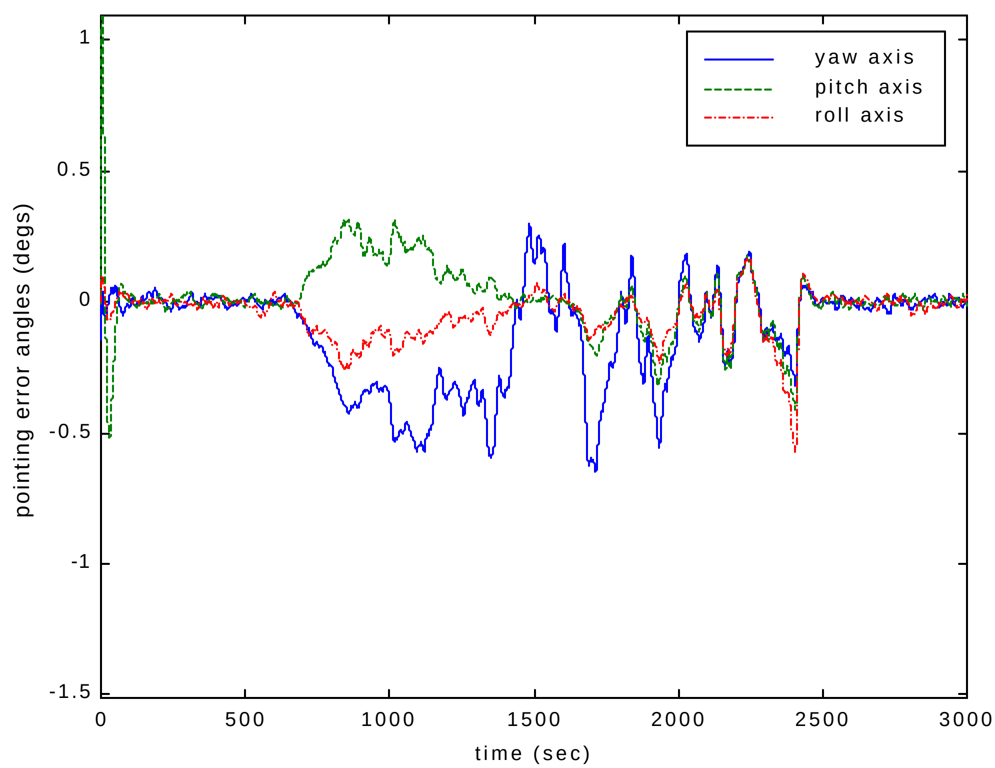

Figure 2: Pointing accuracy for CHIPSat through eclipse

any given time, the magnetometers provide only two pieces of attitude information. Magnetometers measure the direction of the Earth's magnetic field. There is, however, no means of measuring the spacecraft roll around the magnetic field line, and therefore only two axes of information are available. The direction of the error in the simulation is roll about the magnetic field line. The only attitude sensors that can provide information on this third axis are the rate gyros which measure the spacecraft attitude rates, but are subject to bias drift. This simulation has been fed with measured bias drift to represent the true response of the rate gyros. With only two pieces of true attitude measurement, as the bias drifts, the EKF has no means of distinguishing between bias drift and true spacecraft body rates. Therefore, the spacecraft follows the bias drift that corresponds to roll about the magnetic field line. However, because the spacecraft is orbiting the Earth, and to a much lesser degree because the Earth itself is rotating, the magnetic field direction does not remain constant inertially. This means that no individual axis is relying solely on rate gyro information for long, and also that information is available to the EKF to allow for an estimation of rate gyro bias drift. It can be seen in Figure 2 that the nature of the pointing error changes from being predominantly pitch and yaw in the early stages of the eclipse, to an equal pointing error in all three axes by the end. This corresponds precisely to roll about the magnetic field line at these times in the eclipse.

About 2400 seconds into the simulation, the spacecraft comes out from eclipse. As soon as the sun is visible in the field of view of the sun sensor, the EKF attitude estimate is corrected. The feedback control can then correct the pointing error. This happens within about 30 seconds of exiting the eclipse. Once this transient is over, the pointing error returns to the 0.1 deg level that was achieved prior to entry into the eclipse.

It should be noted that this simulation shows the stability of the pointing system on this satellite, not necessarily the accuracy. There are a number of other errors that cause the absolute pointing accuracy to degrade, including the accuracy of the sensors themselves, the mounting accuracy of the sensor to the spacecraft, and particularly ephemeris errors in the models of the spacecraft orbit, and the Earth's magnetic field. However with these errors included the attitude control is capable of meeting the 2 degree pitch and 4 degree yaw accuracy requirements.

The second simulation looks at fine pointing mode for MOST. 6000 seconds, roughly one orbit, is simulated for the MOST mission. This simulation is designed to demonstrate both the level of accuracy that can be obtained in the fine pointing mode, and the operability of the attitude control system with multirate data. Again, the attitude control torques are applied by three reaction wheels. The attitude sensor suite consists of the MOST star-tracker and three single axis rate gyros. The star-tracker is sampled at a 1 Hz rate while the rate sensors are sampled at 10 Hz.

Figures 3 and 4 show, in different ways, the pointing accuracy that is achieved by the HPAC system on MOST. Figure 3 shows the pitch and yaw pointing errors as a function of time. The roll error is not shown because the telescope is not very sensitive to roll about the boresight. This means both that the pointing requirements are not as stringent in the roll direction and that the star tracker is not as sensitive in measuring about the roll axis, and consequently errors are much larger. However, the roll errors that result are much less than the 1 degree requirement. Figure 4 shows the pitch and yaw pointing errors as a

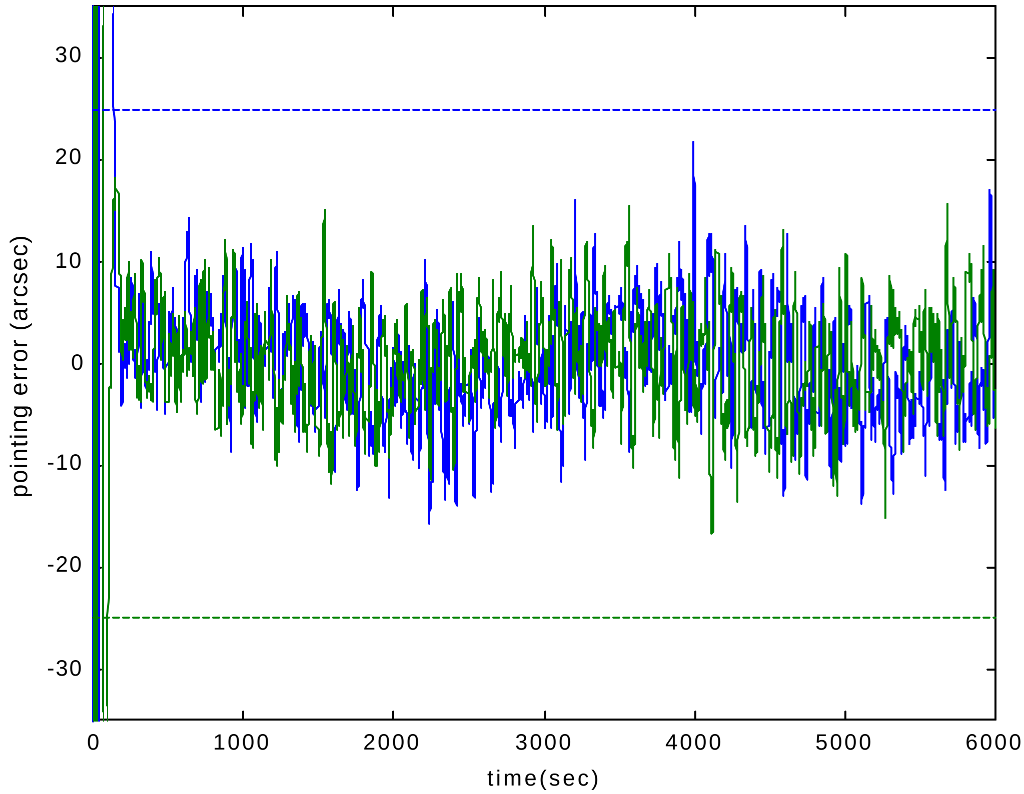

Figure 3: Pointing errors in single orbit simulation for MOST fine pointing

function of one another. In essence this shows where the boresight of the telescope is pointing in the pitch-yaw plane.

This simulation was initialized with an error of 1 degree, and the attitude is quickly corrected to within the 25 arcsec requirement for the spacecraft which is delimited by the dashed lines in Figure 3 and the dashed circle in Figure 4. Note that in Figure 4, the line entering from the upper right shows the initial transient, and from that point on pointing error remains within 25 arcsec. Figure 3 shows two significant error components a very low frequency twice per orbit oscillation, and a higher frequency error. The high frequency pointing error is produced from the noise and quantization error in the star-tracker, while the low frequency error component is the result of the disturbance torques on the spacecraft. The largest disturbance torque is due to residual magnetic moment on the spacecraft which interacts with the Earth's magnetic field to produce a disturbance torque on the spacecraft. The direction of the magnetic field changes with a period of twice per orbit, and thus the pointing error has a period of twice per orbit. While the gain of the controller is very high at low frequency, it is not infinite. Therefore it cannot eliminate the errors due to torques on the spacecraft.

Unlike the previous simulation, this simulation does represent the absolute accuracy of the attitude control system. There are no additional error terms to consider because no ephemeris data is used by the star-tracker, and most importantly the star-tracker and the instrument share the same boresight.

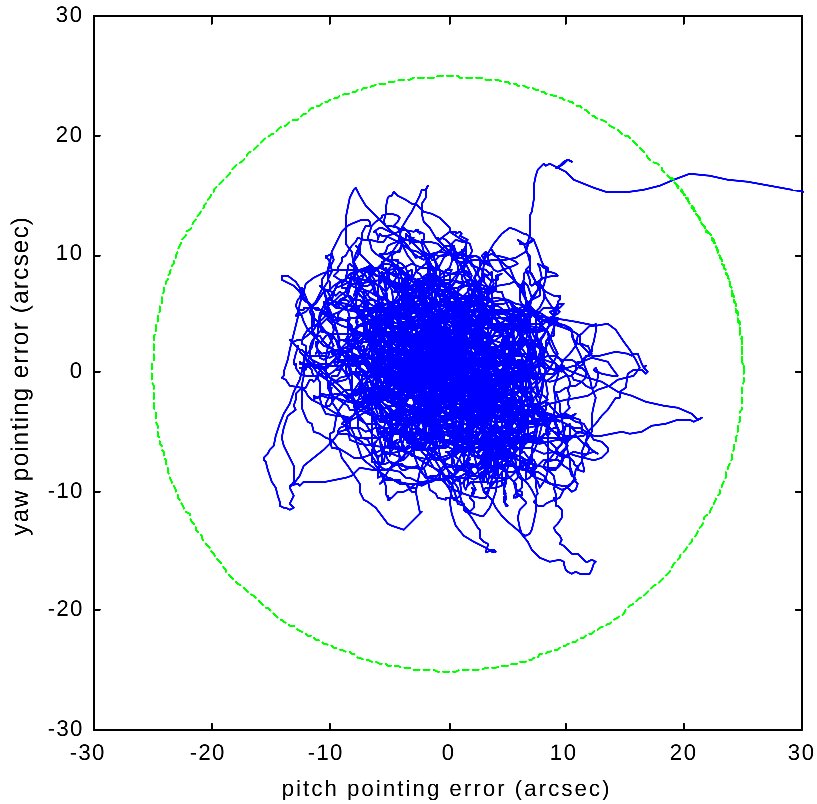

Figure 4: Telescope boresight pointing error for MOST

Conclusions

The simulations of the HPAC system have shown the capability of HPAC to produce highly accurate pointing control. Furthermore, this capability is very flexible in terms of the number and type of attitude sensors that can be used on the spacecraft mission. Accurate three axis pointing stability has been shown for a system which relies on only magnetometers and rate gyros through eclipse, yet has the flexibility to simply incorporate sun sensor information when the spacecraft is not in eclipse conditions. Multirate data capability has also been shown.

Most importantly, the HPAC system has meets the pointing requirements for three separate and very different spacecraft. This demonstrates the ability to save cost through modularity of the algorithms and software, allowing for reuse between satellite programs.

Acknowledgements

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