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GN&C Lessons Learned From Multiple Missions

Chuck Weyandt
Maryland Aerospace Inc.
2145 Priest Bridge Drive, Suite 15, Crofton, MD 21114 | Phone: 410.451.2505
chuck.weyandt@maiaero.com

ABSTRACT

Guidance, navigation and control (GN&C) or attitude determination and control (ADACS) can vary from very simplistic systems that simply minimize tumble rates of a vehicle, to complex 3-axis pointing systems that can provide a variety of pointing modes to ground and space objects. Recent experience with a variety of missions provides some view into just how sensitive the sensors and actuators can be in a real space environment. Flight experience coupled with high fidelity dynamic simulations and detailed analysis show how seemingly simple devices such as sun sensors, magnetometers and infrared Earth horizon sensors IREHS) have provided challenges to meeting desired pointing.

The challenges of a CubeSat or nanosat command and telemetry capability in evaluating performance and resolving minor on-orbit issues is also reviewed. The closed loop nature of satellite control can provide extreme challenges in determining actual performance and pin-pointing minor sources of error. Multiple experiences of using limited on-orbit data, lack of real-time control, and challenges of providing on-orbit software or scripting changes for a time limited mission are reviewed and can provide valuable lessons learned for future missions.

INTRODUCTION

Attitude control of a spacecraft is required for most space vehicle missions. The majority of CubeSat/nanosat programs with very small budgets along with the volume, mass and power constraints such vehicles present, prevent the direct use of existing flight proven attitude sensors. This, and the compressed schedules relative to typical spacecraft programs, has produced a paradigm of borrowing equipment from other disciplines or developing new hardware on very low budgets and with less testing and review. This introduces a variety of potential issues that can significantly impact the performance or reliability of guidance, navigation and control (GN&C) or attitude determination and control subsystems (ADACS), especially for early generation flights. With less flight heritage performance from which to draw on, the expected performance of these systems is based on limited ground testing and simulation. While simulation is necessary for analyzing expected system performance, the lack of heritage flight data to develop high fidelity models can result in overly optimistic performance estimates.

Determining the attitude of the vehicle is the first part of spacecraft pointing, and is the focus of this paper. There are a variety of ways that sensors can be used to drive the attitude pointing: some missions can use direct sensor input for operations such as sun pointing. Other operations require full 3-axis attitude knowledge, which requires multiple vectors from either multiple sensors (for example, the sun and magnetic field measurements combined with inertial models) or multiple objects from a single sensor, such as with a star tracker.

As CubeSat/nanosat experience is increased, so are the demands on sensor/attitude determination performance. A review of recent flight experience together with extensive development and flight experience from larger, more traditional vehicles is leveraged to develop reliable systems and realistic expected performance that can be used in support of payload missions under development.

Recent flight experiences, as well as decades of design and flight operations with a variety of different sensors and actuators have provided us with insight into a series of lessons learned that is applicable across the GN&C discipline, but especially to the design and anticipated performance of CubeSat control systems. Examples of some recent CubeSat on-orbit attitude control performance are evaluated and compared with simulation results. Performance flaws have been identified, evaluated and traced to different performance errors that can provide insight into actual performance. These types of errors can also be extrapolated into GN&C hardware simulation models and used to derive more realistic performance expectations for other missions.

ON-ORBIT PERFORMANCE EXAMPLES

Recent on-orbit telemetry provides samples of the impact of small to moderate performance variations affecting attitude determination. A review of five different sensors - magnetometer, sun sensors, gyros, IREHS and star trackers - can provide insight into the potential performance errors and relatively large impact that these variations can cause. The performance is based on experience using Maryland Aerospace's MAI-400 ADACS and experience from larger space vehicle sensors.

Magnetometer

Magnetometers are a common sensor used for CubeSats due to the small sizes available and their ability to be included inside the vehicle. Magnetometers measure the Earth's magnetic field to support attitude control, whether that be to support a simple de-tumble or create a tumble/spin, used to unload momentum, or as an attitude sensor. As an attitude sensor, magnetometers are typically used with another sensor to create 3-axis attitude knowledge by combining the sensed vectors and modeled measured vectors. With two vectors in both the vehicle (body) frame and two associated vectors in the inertial frame, the attitude can be determined using a cross-product attitude determination method or any of a variety of other options.

One of the attractions to using a magnetometer for attitude determination is the full-time availability of the sensor. Unlike most other sensors, a magnetometer cannot be used for a direct pointing of the vehicle (no known missions align the vehicle with the Earth's magnetic field). Thus, a magnetometer only provides two axes (a single vector) of attitude knowledge, which are relative to the magnetic field of the Earth, and not likely to be of much benefit by itself. It must be combined with another sensor to provide 3-axis pointing.

When used as a sensor for de-tumbling or detecting the magnetic field to enable magnetic torquer momentum unloading, low level noise, misalignments and small scale factors are usually small enough to not affect basic performance. Of course, the use of magnetic torquers in concert with the magnetometer readings requires that the magnetic torquer effect on the local magnetic field is compensated for, either through powering off the torquers before measurements are made, or through calibrated correction of the measured magnetic field. Residual dipoles can be estimated through magnetic calibration and are generally reduced or determined to be small in comparison to the magnetic torque dipoles.

The small form factor of CubeSats also requires very careful placement of the magnetometer relative to other equipment. The compact nature of these vehicles results in possible magnetic interference from residual dipoles and localized magnetic field from solar arrays, EPS systems, electronics cards, reaction wheels and other equipment.

When used as an attitude sensor, these factors can have a significant impact on the performance of the magnetometer. As a sensor, the measured magnetic field is compared to a predicted magnetic field model in software. Moderate level magnetic field models are typically accurate to less than three degrees. Higher order models may improve this accuracy to less than a degree in most locations over an orbit. Even higher order models cannot adequately model the magnetic field variation with high precision over the magnetic poles (for highly inclined orbits) and the South Atlantic Anomaly (SAA), which most orbits intersect at some point during a day. Magnetic storm activity generally does not affect the magnetic field performance in LEO, except over the poles and SAA where it can cause multiple degrees of vector angle error.

Ground calibration can be used to estimate alignment and scale factor errors of the magnetometer as well as dipoles. However, precise measurements are difficult without high precision test magnetometers, proper procedures and a very low magnetic environment. A Helmhotz coil system simplifies ground calibration by cancelling the ambient Earth field and damping external field variations. The small form-factor of CubeSats and nanosats places the magnetometer very close to equipment that may be difficult to totally eliminate, and may require state-based magnetic field compensation equations in software.

Figure 1 shows the effect of electronics that are located close to the magnetometer turning on and off. These examples lasted from seconds to minutes, and caused angular variations of up to 25 degrees. Equipment may affect the magnetometer as a bias when it is turned on or off if magnetic calibration was performed with it on (especially from higher powered units such as downlink communications), during solar array operation or based on battery charge state (EPS). Magnetic interference from magnetic torque actuators also needs to be compensated for, either with equations or by cycling when the torquers are used and magnetometer readings are taken. In this case, enough time needs to be provided between powering off the torquers and reading the magnetometer to allow the magnetic effects to reduce before readings are started. In Figure 2, an attitude slew caused an apparent change in the magnitude of the magnetic field (which does vary over an orbit, but moderately slowly). This is traced primarily to incorrect biases that were applied to the magnetometer. This caused an even larger vector angular error.

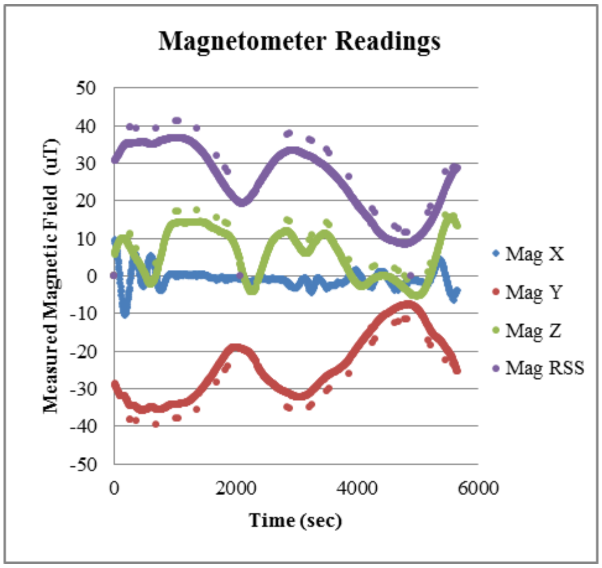

Figure 1: On-Orbit Magnetometer Jumps

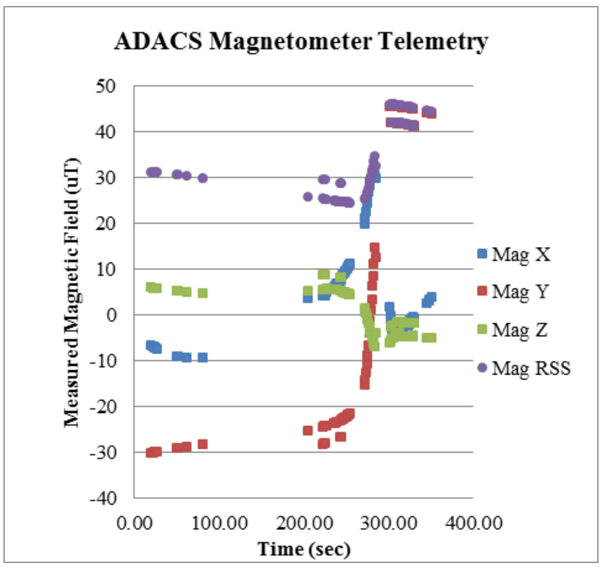

Figure 2: On-Orbit Mag. Field During Slew

On-orbit calibration of the magnetometer can also be performed. This can be affected if the calibration is performed over areas where the magnetic field is more variable than the model, such as over the magnetic poles or SAA. Figure 3 shows an example of post-calibration for a low inclination orbit which passes over the SAA regularly. The International Geomagnetic Reference Field (IGRF) is a low (4th) order field calculated in the on-orbit software compared to the measured field to assist in attitude determination.

The 4th order model is selected for on-orbit use because it is accurate relative to a higher 8th or 10th order IGRF model or high order World Magnetic Model over most of the orbit. In locations where the lower order field does not match the higher order models, the field is also affected by variations in the magnetic field caused by magnetic storms which are caused by variations in the solar flux, but short term which are typically caused by solar flares and longer term variations. Thus, the benefit of the higher order models are not significant compared to the processing required.

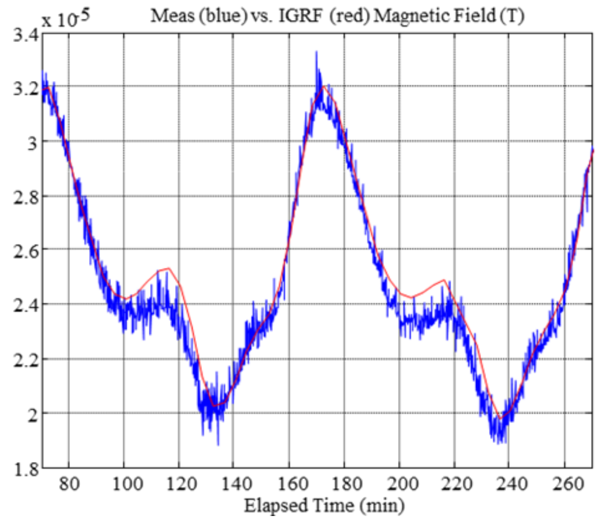

Figure 3: Post Mag. Field On-Orbit Calibration

This figure shows a larger error between the measured field and the modeled filed over the SAA period. The difference between the measured field and the model changes over the 2 orbits shown, providing some indication that the field over the SAA was varying. It also shows noise levels and periods where the noise is higher than nominal, potentially due to nearby equipment on the satellite powering at a lower level than seen in Figure 1.

Analysis of the magnetometer noise shows similar short term noise comparable with ground measurements. This does not include identified jumps from localized fields by electronic equipment. The effect of the noise on attitude determination is based on the magnitude of the magnetic field, and therefore has a larger effect in space where the magnetic field magnitude is lower.

If simulations are used to predict the magnetic field, the simulation will use a modeled field similar (potentially with higher order) than performed with the on-orbit software. If the on-orbit field varies from the models used, the attitude performance will of course be different than that predicted.

Sun Sensors

A group of simple sensors needs to be combined to provide a sun vector. This can be as few as 3 sensors, but there are a myriad of options that can be used depending on the coverage desired, accuracy, etc. ranging from simple groupings of coarse sensors, slit sensors with masked sensor patterns, and high accuracy specialized sensors. There are some issues that need to be considered with sun sensors for use as an attitude sensor. These include effects of shadowing and reflections from appendages or other sensors, Earth albedo, alignment differences, etc.

In the continual effort to reduce volume, usable surface area and cost, a variety of efforts have been used to find alternatives for sun sensors. As with larger spacecraft, silicon cells prove to be a good approximation of a cosine effect of the Sun, making them good “coarse” sun sensors, accurate to within a few degrees without calibration (plus potential alignment errors).

Vehicles equipped with multiple solar array panels, such as body mounted panels and even multiple angled deployed arrays can use the current derived from the panels as sun sensors as well. However, the panels need to be used as a direct energy transfer EPS system rather than a peak power tracking system which changes the voltage point of the arrays to reduce the power output of the cells based on current battery charge. A peak power EPS system might be possible to be used but would require (most likely calibrated) software compensation of the data to correct for the voltage point effect.

Maryland Aerospace performed a test comparing silicon cells to a variety of photo-cells, which are generally smaller than the silicon cells. Figure 4 shows a comparison of these sensors. This shows the variation between photocell types. Some of the photocells are much more of a bi-mode sensor, making it difficult to use as an accurate angular sensor. None of the photocells compare favorably with the silicon cell (IXYS) for uncompensated use and full hemispherical coverage.

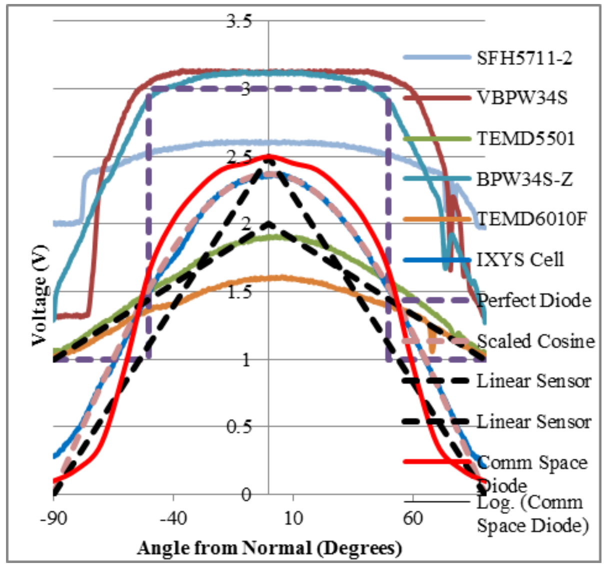

Figure 4: Sample Sun Sensor Performance

Several of the photocells (TEMD5501 and the commercially available space diode) provide sufficient coverage and variation over angle to be used with software compensation to be comparable to the silicon cells, however, neither are adequate without software compensation to provide a sensor less than 20 degrees over its field of view.

Limited evaluation of ground test and on-orbit performance of the commercial space photo-diode also shows that there is likely moderate variation from sensor to sensor, requiring calibration, perhaps for performance better than 10 degrees. On-orbit data of this photocells indicates that it (and likely many other photo-cells) are more sensitive to reflections and Earth albedo. Data from peak albedo seems to suggest a peak signal of 30% of solar input vs. approximately 10% for a silicon cell. This may be due to the spectrum sensitivity of the photocells relative to the silicon cells.

Figure 5 shows an example of the measured sun vector using a series of photocells. This data was taken from a slow tumble without attitude control, thus the expected sun vector should be smooth. The data shows jumps and variations which are significantly different than expected motion should create. With large jumps eliminated, the angular variation seen over the curves is still well in excess of 10 degrees.

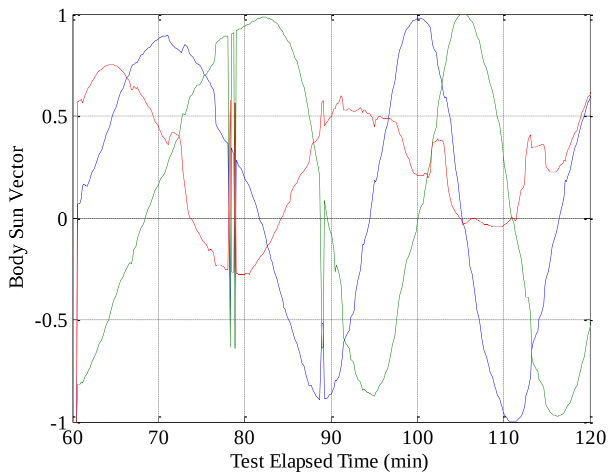

Figure 5: Sun Vector Derived from Photocells

There are a variety of different error sources that could be driving the performance. The variation with respect to an assumed cosine performance (See Figure 4) is a source of error for a single sensor and a grouping of multiple sensors with different error values could result in some of the relative error seen. While a minimum of 3 sensors is required to estimate the sun vector, overlapping fields of view can result in several more being added. Reflections off of solar arrays, other small appendages and even the sensors themselves can create the types of performance observed. Earth albedo (sunlight reflected off the Earth and in some wavelengths, light emitted from the Earth) is an additional source of error. Some methods can be used to reduce or eliminate Earth albedo. However, such methods do have limitations based on relative Sun angle versus Earth angles of the sensors.

Most photocells have been observed to change output levels as a function of temperature. Silicon cells exhibit this feature as well, but have been measured to have small variation relative to the photocells tested. Because photocells are mounted externally and have a variable illumination to the sun, they will experience a wide range of temperature change. Unless calibrated versus temperature, this could be a significant source of error.

Finally, the small size of the sensors makes them hard to align properly to a spacecraft axis. Affixing sensors on corners or to appendages increases the likelihood of larger alignment errors that, without detailed testing and calibration, can create significant errors in determining a sun vector.

In addition to individual sensor performance, one aspect that is not covered by performance analysis is the selection of the number of sun sensors used on a vehicle. This will depend on a number of factors, such as the FOV of the individual sensors and the intended use. Sun sensors are traditionally used as a back-up or safe-mode sensor, used to provide a method to point the vehicle towards the sun to collect power and provide a known thermal condition. In this case, a limited net field of view, such as that provided with a 4 cell pyramid sensor can be used, combined with a vehicle slew driven with gyros (see gyro performance below).

If the goal is to provide 3-axis attitude knowledge, a larger set of sensors is generally needed, ideally providing full spherical coverage. In this case, the impact of reflections can be minimized by selecting sun sensor orientations that minimize sensitivity of possible reflections and multiple sensors to provide overlapping coverage. Minimizing the number of sensors may result in the inability to correct for reflections and poor sensor accuracy at times.

Except for highly inclined sun synchronous orbits (that also require propulsion to maintain), low Earth orbit (LEO) vehicles need to accept that if a Sun sensor based attitude determination system is used, that a large percentage of the orbit will not be available for 3-axis control. The time available must include the time spent slewing to the desired attitude once the sun sensors acquire the sun again. For missions that use non-spherical coverage, time must be allocated for a sun search slew or random tumble acquisition of the sun.

Gyroscopes

Gyroscopes (gyros) are typical spacecraft equipment to directly measure rate or relative angular position change of the vehicle. Gyro packages for larger spacecraft often rely heavily on gyros for attitude control, and significant development effort has been used to create high performance gyros. However, gyro packages that are appropriately sized for CubeSat/nanosat are limited to lower performance (relative to typical spacecraft performance) gyros that still use significant power (>1W per axis) or MEMS "gyro on a card". The available MEMS gyros have poor performance for providing attitude control, but can be used for functions such as damping rates. Figure 6 shows some typical on-orbit performance of MEMS gyros (in degrees per second, dps) used by Maryland Aerospace without reaction wheel control that can add to the effective noise.

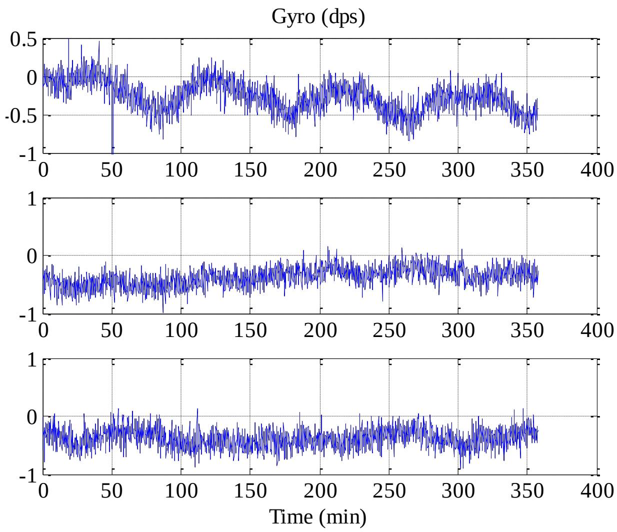

Figure 6: Sample MEMS Gyro Performance

MEMS gyros have moderately large bias specification drifts (1 deg/sec), vary from turn-on to turn-on, and can be sensitive to temperature changes. In comparison, unfiltered rate derived from the on-board magnetometer and sun sensor array have much smaller noise and show effective rate despite the noise levels of these sensors, as shown in Figure 7 for the same time period. Note that in this case, the rates cannot be derived during eclipse periods, although if another sensor is used, such as Maryland Aerospace's IREHS or a star tracker rates could be determined during eclipses (potentially with even lower noise).

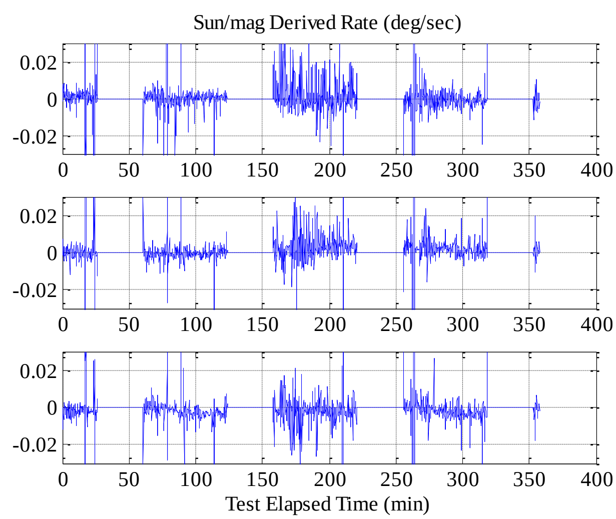

Figure 7: Derived Body Rates

The gyro rate biases could be determined with ground calibration and corrected on-orbit with additional calibration relative to the sensors. However, the noise level of the gyros is still roughly 10 times the level of that that can be derived, which would significantly affect attitude control capability. This noise level is especially critical for trying to achieve small (<1 deg) attitude control capability and attitude control for jitter/drift sensitive payloads.

If used primarily for slewing, it should be noted that the MEMS gyros are not accurately aligned to the board, and may have moderate misalignment angles of a few degrees. This can cause additional error over a quick slew, for example, a 2 degree misalignment can create an additional 6 degree of error over a 180 degree slew. Scale factor errors are also likely and may vary with temperature, creating additional errors for slews.

Earth Sensors

Earth sensors on larger spacecraft were typical up until the last two decades. The reduced cost, improved performance and recent increased lifetime of star trackers have nearly eliminated the use of Earth sensors for modern spacecraft. Indeed, the Maryland Aerospace IREHS is the only currently available commercial tracker sized for CubeSats/nanosats, although others may be being developed. This sensor now has flight history and is working through early flight performance "bugs".

Initial flights in 2014 and 2015 provided data that showed the IREHS sensors worked, although the two vehicles flown were not able to achieve nadir pointing due to issues with the spacecraft. Flights in 2016 have provided flight heritage for this sensor that show significant promise in maintaining attitude knowledge in both sun illuminated and eclipse periods.

As with most initial sensor flights, telemetry data revealed that the sensors were more temperature sensitive than originally believed. The sensor temperature (included in the thermopile sensor) was required to be used to correct relative readings. The Earth temperature appears to be close to predicted value, but the sensor also showed to be sensitive to the much colder temperatures of deep space (~4° K) than initially estimated. This required gain changes in the sensor (which were commandable on-orbit). The temperature sensitivity required software corrections to adequately provide control.

Figure 8 shows the nadir vector calculated from the IREHS sensors. In addition to the temperature sensitivity (compensated for in the plot), the sensors were found also to have occasional jumps that are still being investigated but has been fixed with future software upgrades and hardware connection improvements. Average performance is about 0.6°, which includes control performance, with 1 sigma noise level of 1.0°. Additional variations in performance are still being studied and evaluated.

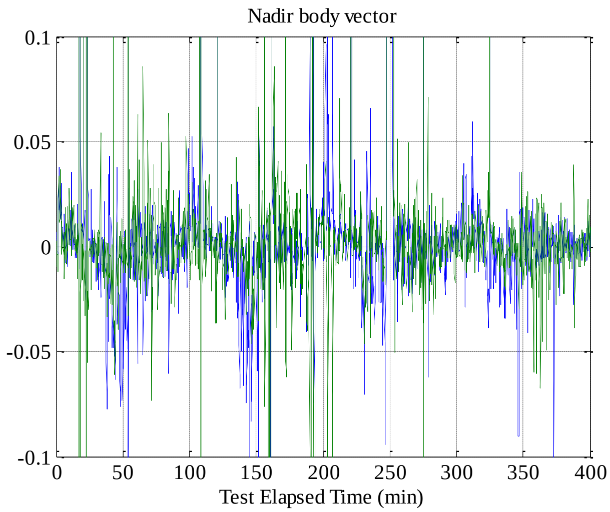

Figure 8: IREHS Derived Nadir Vector

The sensors are now being used and providing 98+% of orbit time nadir pointing (2% outage time includes the time to recover from faults that are created from all causes, including several that are not linked to the IREHS).

Star Trackers

Several CubeSat/nanosat sized star trackers are now commercially available. Ground and ground based night-sky tests have been used to predict estimated attitude performance. Some sensors are claiming performance rivaling that of the best star trackers available for larger vehicles.

However, decades of experience with flight trackers on larger vehicles have shown that the performance achieved, both for attitude knowledge, rate capability and sensitivity to stray light from the Sun, Moon and Earth is not as easily achieved as initially thought based on the same types of tests being performed with the CubeSat/nanosat lines of star trackers.

Stray light is of essential performance to being able to detect low magnitude stars. Here, decades of flight experience have shown that adequate baffle design is critical in providing a working tracker. Out of 6 different flight star trackers/cameras that the author has had extensive experience with (cameras are defined as trackers where the algorithms for star detection, ID and attitude determination are performed outside of the electronics provided with the sensor), 3 were tested on the ground for stray light rejection with their baffles, only to have on-orbit performance shown to be much worse than predicted (up to 90 degree solar stay-out angles vs. 30 or 45 degree predicted by test).

Operational sensitivity to radiation of the focal planes of the trackers also has been an issue for many trackers, causing issues with noise or creating excessive false stars to cause processing issues. Many focal planes also are sensitive to thermal variations, causing additional noise on the focal plane to affect minimum star magnitude capability.

There are a variety of other issues that new star tracker vendors have experienced issues with, such as lens darkening (due to radiation), lens coating darkening or degradation, and lens distortion.

Another area requiring significant performance impacts is thermal deformation. With star tracker providers claiming performance capability greater than 0.01°, thermal distortion of the tracker lens assembly, focal plane, and mounting relative to the payload requiring tight attitude knowledge needs to be addressed. As with the other CubeSat/nanosat sensors, the funding limits of such programs generally do not allow for high performance testing across environments that will match on-orbit conditions. Thus, methods will need to be developed to address such issues, such as the use of on-orbit calibration. The complexity of on-orbit calibration requires a significant amount of data processing and experience and should not be underestimated both for the level of detail required and the time to perform such operations.

ON-ORBIT DATA COLLECTION AND ANALYSIS

Recent flight experience with multiple CubeSat/nanosat developers raises an issue that is not specifically sensor related, but rather a system issue. The communication systems of a CubeSat/nanosat is limited in size and power much like the attitude determination system. The result is that there is likely a significant reduction in telemetry provided to analyze the attitude determination/control performance of the vehicle.

Data rates of once every 10 seconds, once per minute or even much lower may be adequate for validating that the system is operating as expected. However, if the system is not operating as expected, such as shown for the sensors described above, the slow data rates provide very little data to define what the issues may be.

It can be very difficult to determine the source of error for a closed loop system with sparse telemetry. For example, it can be impossible to determine whether a system is showing high sensor noise and variations in reaction wheel speeds to determine if the source of the error is sensor noise, attitude control instability or anomalous variation in wheel speeds. If they can be supported, specific on-orbit tests may be defined to evaluate sensor performance without actuator operation, or reaction wheel speed tests without input from sensors. However, implementing tests that were not initially prepared for or designed into an operation plan can be difficult and time consuming.

Even with additional tests, sparse telemetry may make understanding the cause of unexpected performance difficult. Collecting data at the rate of data use can provide insight into the nature of the sensor or actuator performance that would otherwise appear as random noise. Often, anomalous performance will have rise and fall times that describes the actual source of error, or it can show whether actuator or sensor performance is initially affected that in return causes the larger problem being evaluated.

If the anomalous behavior is infrequent, collecting enough data to determine a correlation to operational or environmental conditions can also be difficult. It may require copious amounts of sparse data or sets of high speed data to find the correlation of events and environment. However, if the system is not designed to collect high speed telemetry and transmit it to the ground, it may be impossible to correct the on-orbit situation, and highly daunting to mimic the performance on the ground for future hardware or software modifications to correct the root cause of the problem.

Attitude determination systems, in general, are part of a closed loop control system. Some data, such as Earth magnetic field measurements and gyros may be able to be operated with the control system off in order to collect data, while others may give intermittent data or limited performance information that may still prove useful. If designed with testing in mind, the actuators should be able to be commanded in an open-loop fashion to facilitate performance evaluations apart from the sensors or closed loop control complexities.

The additional data collection capability may still be constrained by the downlink capability of a low cost system. However, without the capability to provide higher rate telemetry in anomalous conditions, it may not be possible to correct problems that may prevent the current asset from meeting mission goals or providing valuable insight that may allow future program reliability and/or performance improvements.

LESSONS LEARNED AND GUIDELINES TO FOLLOW

Lessons learned from the experience of other programs can reduce repeating failures of the past. With larger, traditional spacecraft, this has led to more and more testing, using expensive test equipment and following strict conformance to practices that prevent ingenuity. The test programs associated with “space qualified” hardware and software is a significant driver of the system cost. In addition, the cost of initial development and testing is amortized over an anticipated product build. Rarely is the size or material cost of GN&C hardware the actual driver for cost and schedule. Expectations of testing and qualification for low cost systems needs to be tempered with the knowledge of how much typical space qualification testing costs to perform. The GN&C component vendors (as well as vendors for many other CubeSat components) must find new ways to provide some level of testing to support flight expectations while not driving up component costs that could drive the components out of the budget for the majority of CubeSat/nanosat programs.

The CubeSat/nanosat paradigm that allows these vehicles to be developed and fielded for a small fraction of the cost of larger vehicles is primarily due to avoiding the drivers of traditional spacecraft cost. Providing the same level of testing and build rigor that enhances the reliability of traditional space vehicles cannot be performed. Therefore, several “lessons learned” and guidelines are developed to enhance the reliability and performance of low cost systems.

1) Do not expect limited ground testing or “wishful thinking” to provide a reliable attitude determination/control subsystem. Attitude determination and control systems are complex systems that may not be adequately tested with limited budget test set-ups. In addition, simulation performance based on simplistic models may show performance that is better than may actually be realized on-orbit. There is very limited oversight in the current CubeSat market to provide for an “apples to apples” comparison of performance, and therefore the onus is on the vehicle builders to try and do a relative comparison of available hardware or systems. In general, as with most businesses, get what you pay for. A mission requiring high accuracy ADACS performance may find it irresistible to select hardware that is unproven but that has shown with preliminary testing to provide capabilities that rivals state of the art traditional space hardware.

2) Rely on heritage based sensors and performance where possible. The same applies for attitude control hardware and ADACS systems. Using detailed sensor models based on on-orbit telemetry can provide a realistic representation of actual performance, but all aspects of performance should be understood. Of course, part of the CubeSat paradigm is quick turn-around improvements and design. As with initial hardware, relying on previously flown designs and making moderate improvements to address flaws or enhance performance will reduce the amount of risk compared to regularly trying the newest hardware available.

3) Expect to go through a period of evaluation and potentially calibration and correction for any new hardware and/or algorithms fielded on-orbit. Even proven systems may have minor changes in build or environment that may provide an adverse change in capability. In many cases, it may be more cost effective to perform on-orbit calibration than trying to accurately calibrate systems on the ground. A good example of this will be star trackers, where the desired performance is exacting enough that minor alignment and thermal/mechanical distortion errors will dominate the net performance. Providing high accuracy ground alignment and tracker performance test equipment is likely to exceed the cost of the tracker or possibly the entire spacecraft, and may still be affected by launch and on-orbit conditions.

4) Develop a system that can collect higher rate sensor, actuator and ADACS telemetry to help debug a system on-orbit. The same capability may also assist in potential ground testing. Provide a capability of selecting non-standard telemetry that may be required for analyzing anomalous behavior that may not be possible to collect normally due to communication constraints. This requires fore-thought into the telemetry system and on-orbit data storage requirements that may not otherwise be required. However, due to the complexity of the ADACS system, the minor outlay of cost into the telemetry system is likely to pay off by enabling root cause investigations, future ADACS improvements and potentially on-orbit fixes that can overcome unexpected flaws in performance.