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Space Technology 5 Launch and Operations

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NASA Goddard Space Flight Center

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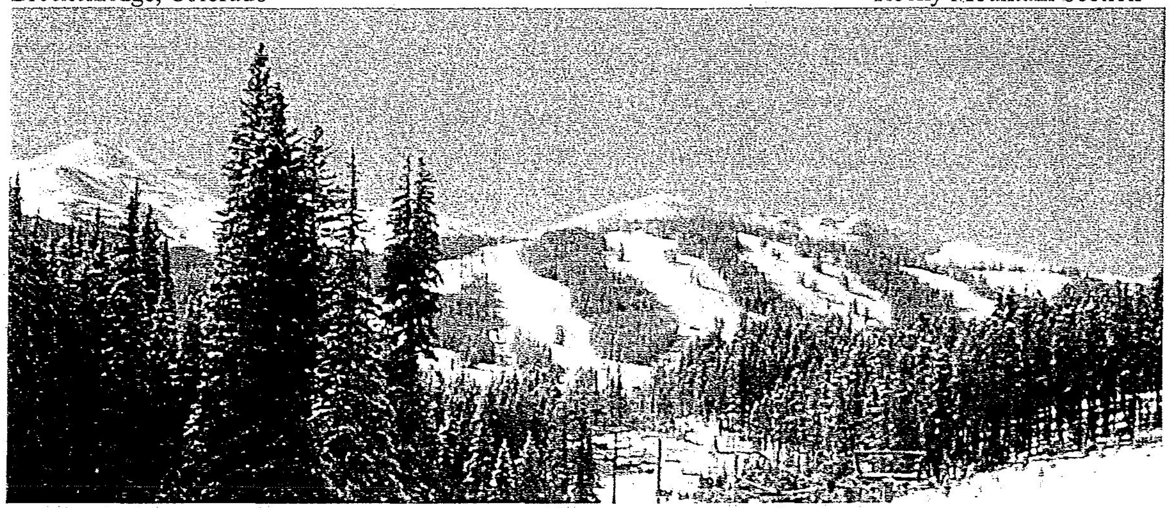

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SPACE TECHNOLOGY 5 LAUNCH AND OPERATIONS

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The three spacecraft that made up the Space Technology 5 (ST5) mission were successfully launched and deployed from their Pegasus launch vehicle on March 22, 2006. Final contact with the spacecraft occurred on June 30, 2006, with all Level 1 requirements met. By the end of the mission, all ST5 technologies had been validated, all on-board attitude control system (ACS) modes had been successfully demonstrated, and the desired constellation configurations had been achieved to demonstrate the ability of small spacecraft to take quality science measurements. However, during those 100 days (ST5 was planned to be a 90-day mission), there were a number of anomalies that made achieving the mission goals very challenging.

This paper will discuss: the chronology of the ST5 launch and early operations, work performed to diagnose and work-around a sun sensor anomaly, spacecraft tests devised to demonstrate correct operation of all onboard ACS modes, the maneuver plan performed to achieve the desired constellation, investigations performed by members of the ST5 GN&C and Science teams of an anomalous spin down condition, and the end-of-life orbit and passivating operations performed on the three spacecraft.

INTRODUCTION

The Space Technology 5 (ST5) mission is part of the NASA New Millennium Program (NMP), which validates technologies for future science programs. The purpose of ST5 was to validate the ability to design and manufacture multiple small spacecraft and operate these spacecraft as a system. The project's goal was to dramatically reduce the weight, size, and cost of missions while increasing their science capabilities. Results from ST5 will be used to design future missions using constellations of spacecraft such as the Magnetospheric MultiScale (MMS) mission currently under development. The NMP technologies that ST5 validated were a miniature communications transponder, a cold gas micro-thruster, variable emittance coatings, CMOS ultra-low power radiation tolerant logic, and a low voltage power subsystem. Other new technologies and hardware flying on the ST5 spacecraft included a miniature science-grade magnetometer, a miniature spinning sun sensor, the spacecraft deployment mechanism, the magnetometer deployment boom, an in-house designed, fluid-filled, passive nutation damper, and an X-band antenna. The three defining Level 1 requirements for ST5 were that the project: shall design, develop, integrate, test, and operate three full service spacecraft, each with a mass less than 25 kg, through the use of breakthrough technologies; shall demonstrate the ability to achieve accurate, research quality scientific measurements utilizing a nanosatellite with a mass less than 25 kg; and shall execute the design, development, test and operation of multiple spacecraft to act as a single constellation rather than as individual elements.

ST5 MISSION OVERVIEW

Figure 1 shows one of the ST5 spacecraft with its magnetometer boom in the stowed position (the structure on the left of the spacecraft is part of the Pegasus Support Structure that holds the spacecraft within the launch vehicle) along with a schematic of spacecraft showing the important components and with the boom deployed. As is shown on the schematic, the body of the spacecraft is just over half a meter long; fully deployed, the magnetometer boom adds another three-quarters of a meter to the spacecraft's length. The spacecraft components that will be discussed in this paper include the magnetometer, the sun sensor, the cold-gas micro-thruster, and the passive nutation damper. The three ST5 spacecraft were denoted spacecraft 094, 155, and 224, based on the transponder spacecraft identification numbers of each.

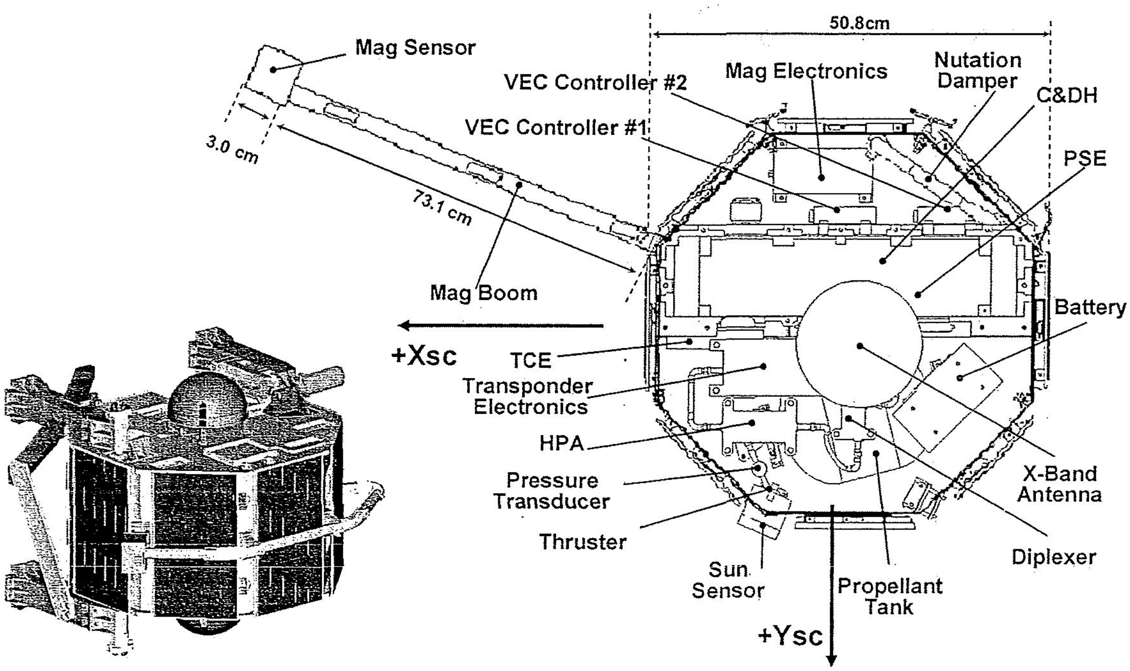


Figure 1: ST5 Spacecraft Schematic

Figure 2 shows the three ST5 spacecraft as mounted in the Pegasus Support Structure (PSS). In the nominal deployment sequence, just after the Pegasus launch vehicle third stage cutoff (3SCO), the launch vehicle would perform a maneuver in order to deploy the spacecraft in their nominal operational attitude; spin axis pointed towards the ecliptic poles and solar arrays normal to sun. The spacecraft would then be deployed after third stage cutoff in the following order: the forward spacecraft (sc155, top of stack, leftmost in figure) at 3SCO + 200 seconds, the middle spacecraft (sc094) at 3SCO + 390 seconds, and the aft spacecraft (sc224) at 3SCO + 530 seconds. The release springs on the PSS provided separation ΔV on each spacecraft as well as the initial spin. Even though the deployment order of the spacecraft was sc155, sc094, and then sc224, because of the dynamics of the three planned orbits and firing of the Pegasus third stage attitude control thrusters, the expected rise order was sc155, sc224, and finally sc094.

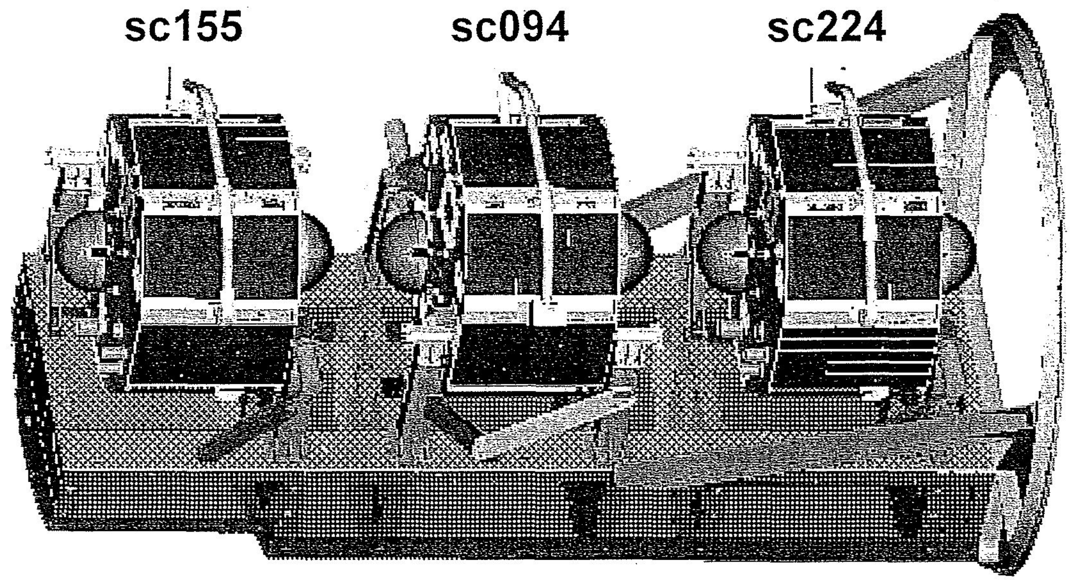


Figure 2: The Three ST5 Spacecraft in the Pegasus Support Structure

Just after deployment, it was planned that the spacecraft would be only meters apart, but nominally spin-stabilized and power positive. The initial constellation separation would then passively increase due to the relative velocities imparted by their orbit position (true anomaly) at deployment and the velocity imparted by the deployment mechanisms. The post-separation and two science validation "string-of-pearls" constellations planned for ST5 are shown in Figure 3.

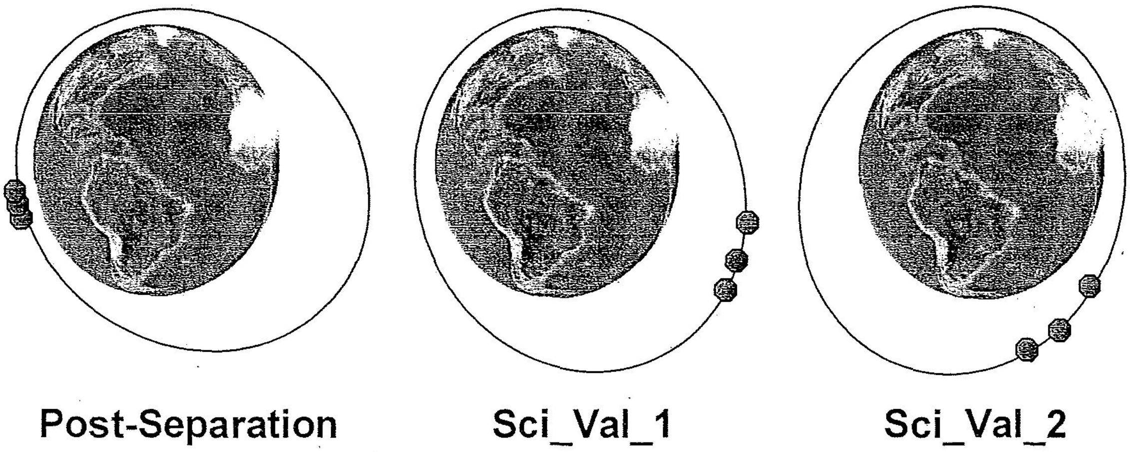


Figure 3: Planned Spacecraft Constellations

SEPARATION, DEPLOYMENT, AND INITIAL ACQUISITION

The three ST5 spacecraft launched successfully at 14:03:52.69 UTC on March 22, 2006. The Orbital Pegasus XL with the ST5 spacecraft aboard was dropped over the Pacific Ocean from its carrier L-1011 aircraft and its 1st stage solid motor ignition commenced 5 seconds after drop. The Pegasus ascent to orbit insertion occurred nominally with the sc155 separating from the PSS at 14:13:33.97 UTC, sc094 separating 14:16:43.92 UTC and sc224 separating at 14:19:54.02 UTC. The insertion orbit was determined to be 67 km high in apogee, resulting in a first acquisition time for the first pass 25 seconds later than originally planned.

In the last few weeks prior to launch, there had been a fair amount of concern both about possible recontact between the three spacecraft and about our ability to acquire and track each spacecraft using their X-band tracking information. Also, ST5's 300 × 4570 km orbit meant that its dynamics were considerably faster than most missions that use the DSN network, raising additional acquisition and tracking concerns. On launch day therefore, there was much relief in the control room when initial contact was made with each spacecraft as planned. However, as each was contacted, it was obvious that an unexpected anomaly was affecting all three. The telemetry point showing the delta time between sun sensor pulses was alternating between a smaller and a larger value. Both of these values were smaller than the expected spin period, though the sum of the two was approximately equal to the expected spin period for each spacecraft. Additionally, there was a flood of downlinked "buffer overrun" event messages from each spacecraft.

In addition to the anomalous conditions apparent from downlinked telemetry, analysis of the launch vehicle separation vectors for the three spacecraft showed that the spacecraft deployment from the Pegasus third stage was completely out of bounds with what was expected, resulting in the forward spacecraft being significantly ahead of the other two and moving away from them at high velocity. The rise order was different than expected; sc155 was still in front, but sc094 was now the middle spacecraft and sc224 bringing up the rear. Further complicating the situation, it was subsequently discovered that the Pegasus third stage had ended up within the ST5 constellation, between sc155 and sc094.

In spite of all of this, however, there was no immediate danger to any of the spacecraft. The orbit rates of the three spacecraft left no fear of recontact, and while the sun sensor sun pulse telemetry was suspect, the elevation angles seemed to be accurate and were showing each spacecraft oriented as desired. Being spin-stabilized and power and thermal safe with no other time-critical operations, there was time to diagnose the anomalies.

SUN SENSOR SPURIOUS PULSES

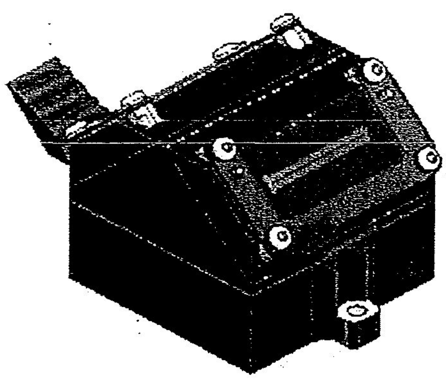
Approximately four hours after the initial contact with the ST5 spacecraft, the "ST5 Anomaly Team" that had been set up before launch was activated. This team, along with members of the flight operations and flight support team, began to assemble data and information to identify the cause of the anomaly. This process was somewhat complicated by the need to monitor the continued operation of each of the *three* spacecraft during each of the real-time contacts. However, because they were spin-stabilized and in a safe configuration, there was no urgency to prepare for time-critical operations.

A fault tree was put together to attempt to diagnose the anomaly. Based on the frequency and the contents of the telemetry packet coming down from the spacecraft with sun sensor information, it was clear that two packets were being sent down each spin cycle. The sum of the sun pulse delta times of two adjacent packets was equal to the expected spin period. The conclusion from this was that the sun sensor was producing two sun pulses each spin period. The potential causes for the additional sun pulse included in the fault tree included the following:

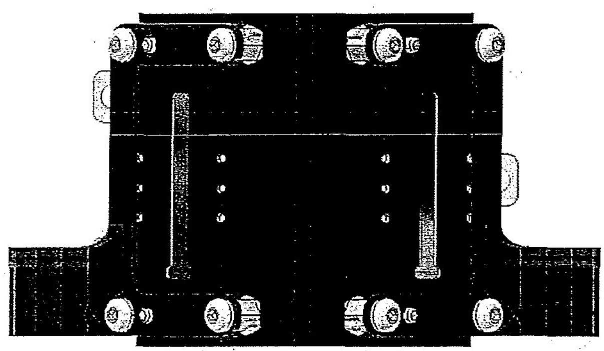
- Reflection of bright object(s) (localized to respective spacecraft),
- Reflection of bright object(s) including Moon, planets, other spacecraft in the string-of-pearls formation,
- Hardware electronics; sun pulse timing and threshold related circuit operation,
- Ground support software, and
- Flight Software, experiencing buffer overflows related to sun sensor packet accumulation.

Sun Sensor Spurious Pulse Characterization

Figure 4 shows two views of the Miniature Spinning Sun Sensor (MSSS) used on the ST5 spacecraft, manufactured by the Adcole Corporation. Each unit consisted of a dual reticle and solar cell assembly and processing electronics packaged in one unit. Its outputs consisted of Sun pulse, an eleven bit parallel data output, and a thermistor signal. The Sun pulse output was nominally generated when the Sun crossed the command plane normal to the sensor. The parallel data output, latched upon generation of the Sun pulse, consisted of a gray-coded digital output word that is a measure of the angle of the Sun line in the command plane (elevation angle). The ST5 flight software (FSW) and electronics used the MSSS data for spacecraft spin rate determination, triggering of thruster firings during attitude precession maneuvers, and closed-loop sun acquisition maneuvers. MSSS data was also used in support of ground-based spacecraft attitude determination and science data processing activities.



MSSS Side View



MSSS Top View

Figure 4: Miniature Spinning Sun Sensor

The search for the cause of the sun sensor anomaly ended the day after launch. As a part of the initial fault tree generation on launch day, Adcole was informed of the spurious pulses anomaly that were seen. Without providing exact details on the timing of the pulses, by the next morning, Adcole reported back to the ST5 anomaly team that they had reproduced the spurious sun pluse effect. This was achieved through test on a spare MSSS Engineering Unit built for ST5. Adcole's results showed a spurious pulse occurred approximately 70° in the spin plan in advance of the true sun pulse, which was identical to the phenomenon being experienced on each spacecraft MSSS. The MSSS was designed to generate a Sun pulse output when the sun crossed through the command plane of the sensor, which is nominally perpendicular to the MSSS mounting surface. A Sun Pulse is generated when the Automatic Threshold Adjust (ATA) signal amplitude is above a preset threshold, and the command signal amplitude exceeds the ATA signal amplitude.

For the true Sun pulse, the ATA signal exceeds its preset threshold by a large margin, and the command signal spikes sharply over the ATA as the sun crosses the sensor. The circuit in the MSSS that performs the comparison of the command signal to the ATA contains hysteresis, so a clean Sun pulse output is generated. However, in testing at Adcole, they discovered that at angles 70° in advance of the true Sun crossing, the ATA threshold signal is low enough that the ATA signal exceeds it and further, that the command signal exceeds it as well. As the command signal is illuminated and reaches a level that is greater than the equivalent threshold on the ATA and the ATA signal then rises, an anomalous Sun pulse output would be generated. Because the circuit that compares the ATA amplitude to its threshold does not contain hysteresis, multiple spurious pulses can be generated. This is because the minimum threshold set for the ATA amplitude was very low and the command signal also exceeds it. Without hysteresis in this circuit, the Sun pulse outputs could chatter, showing multiple spurious pulses within a very short period of time.

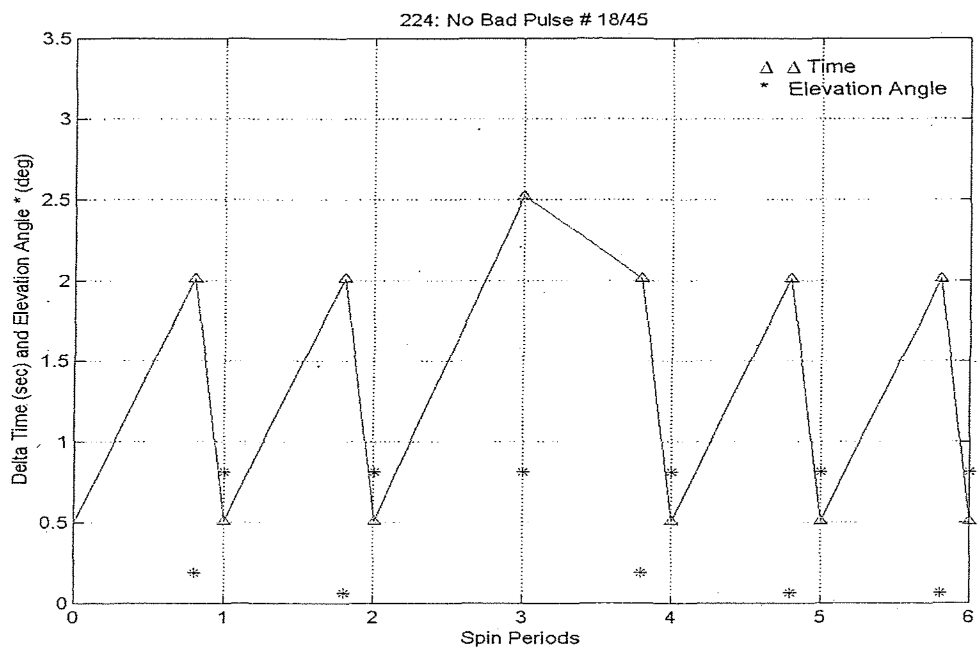


Figure 5: Sun Pulse Delta Times and Elevation Angles

Figure 5 presents flight data from sc224, showing the Δ time, defined as the time between adjacent pulses, and the corresponding elevation angles reported by the sun sensor at those times. Because the spurious pulses are approximately 70° in advance of the true Sun pulse, the signals triggered by the Sun correspond to the smaller Δ time. Figure 5 shows a "short" Δ time of ~0.5 sec and a "long" Δ time of ~2 sec, meaning that the sc224 spin period is ~2.5 sec. The elevation angle corresponding to the good pulses is approximately 0.7°. The elevation angle reported as a result of a spurious pulse is the result of whatever light happens to be illuminating the sun sensor at the time of the pulse. In subsequent testing at Adcole, they were able to produce data showing the elevation angles that would be reported from spurious pulses, depending on the true elevation of the spacecraft (see Figure 6). It should be noted that the elevation angle reported during spurious pulses was much more vulnerable to contamination by other light sources, particularly Earth albedo. On sc094, which was oriented upside-down compared to sc155 and sc224, as the orbit precessed to bring perigee to higher and higher latitudes, the elevation angle reported during spurious pulses began to jump to high values whenever the spacecraft flew over the north pole.

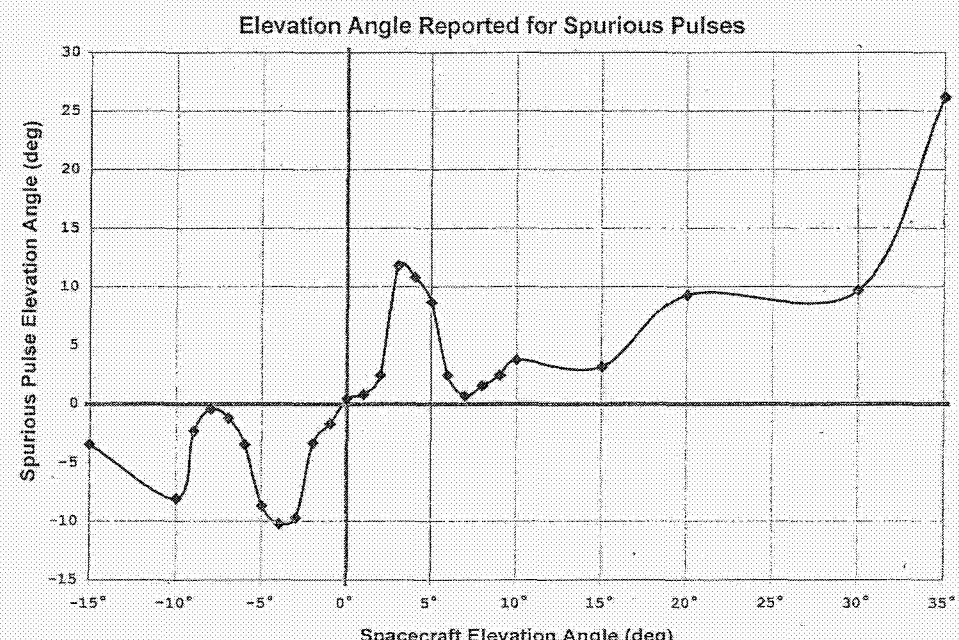


Figure 6: Elevation Angles from MSSS Spurious Pulses

There is one other "feature" of the sun sensor as shown in Figure 5 that would become important to dealing with the anomaly in actual operation. Notice that during the third spin period there was no spurious pulse. As a result, for that particular spin cycle, the Δ time reported was the actual spin period of the spacecraft.

Once the ST5 team had fully characterized the MSSS anomalous behavior with the full cooperation and support of Adcole, a meeting was held to give lessons learned to members of the THEMIS project, as they were using the Adcole MSSS on their five spacecraft as well. The MSSS used on ST5 was a new build for Adcole, using heritage optics but a new electronic packaging. To support the THEMIS project MSSS procurement, which had particular concerns about glint or reflection from the other THEMIS spacecraft in close proximity, design refinements were made to the MSSS. These refinements included a slightly higher ATA threshold level, an advanced baffle design, and the addition of hysteresis to the ATA threshold compare circuit. With these refinements implemented, the chance of the ST5 anomaly occurring on THEMIS was eliminated. Further, GSFC ST5 Anomaly Team recommended that Adcole modify their test program to include testing of the sun sensor over the full range of azimuth and elevation angles. For past procurements and as procured on ST5, the testing of the sun sensor's response was focused to an output window of ±4–5° degrees in spin phase (azimuth) for a zero elevation angle (sun line lying in the "equator" of the MSSS) about the zero crossing of the Sun, i.e., where the sun pulse is issued. Had a wider range of test window about the zero crossing been established for the acceptance test program for the ST5 sensors, the spurious pulses would have been discovered well before launch.

Effects of Spurious Pulses on Spacecraft Operations

For each time the MSSS produced a pulse, whether a true Sun pulse or a spurious one, it triggered several actions on the spacecraft. Two of these actions, one in software and one in hardware, were of primary concern to the guidance, navigation, and control (GN&C) flight support and anomaly teams. On the software side, each sun pulse would trigger a software interrupt service routine (ISR) which would then start the flight software attitude control (AC) task. The AC task would then read the elevation angle of the spacecraft and calculate a filtered spin rate based on the Δ time since the last pulse. In the event that that the spacecraft had been commanded into its sun acquisition or attitude precession modes, the AC task would also calculate the timing and duration of the next cold gas thruster pulse. Note that the buffer overrun messages flooding the downlinked event messages were a result of the multiple, closely-spaced spurious pulses generated during most spin cycles; while the software was still processing the interrupt generated by the first pulse, additional interrupts would occur that the software was not able to process and so a buffer overrun would result.

The hardware action triggered by the MSSS pulses is related to operation of the cold gas thruster. For sun acquisition and attitude precession mode, the timing of the thruster firing was based on receipt of the Sun pulse. When this pulse is correctly received once per spin period at the time of the sun crossing, the pulse provides a very good way of timing thruster firings to achieve the desired reorientation of the spacecraft spin axis.

Of these two actions, the hardware-based action was the one that was the most problematic. The ISR routine that responds to MSSS pulses and the AC task that was used to implement the desired actions could be rewritten and patched on-orbit, if necessary. However, there was no way to change the hardware action related to thruster firings. When the spacecraft was placed in sun acquisition or attitude precession mode, it would calculate thruster firings as a delay and a pulse width and write those commands to the thruster hardware command registers. When the next MSSS pulse was received, whether it be a true Sun pulse or a spurious one, the thruster hardware would then wait out the commanded delay before firing the thruster for the desired pulse width. Note that this was not an issue for thruster firings associated with orbit maneuvers, as those were fired strictly on a separate 2 Hz clock. If another pulse was received during the delay time of a thruster command, processing of that command would stop in favor of the next one (if any).

The combination of these two actions results in a "keep-out" zone for thruster firings twice the size of the ~70° between the spurious pulse(s) and the true Sun pulse (see Figure 7). The reason that this keep-out zone is twice the size is not completely obvious. On most spin cycles, the AC task would be run twice, once at the time of the true Sun pulse and once at the time of the first spurious pulse. If in sun acquisition or attitude precession mode, any thruster pulse commanded would be executed at the time of the next pulse. So, a command generated at the time of the true Sun pulse would be executed (after the commanded delay) at the time of the spurious pulse, while a command generated by the spurious pulse would be executed at the time of the true Sun pulse.

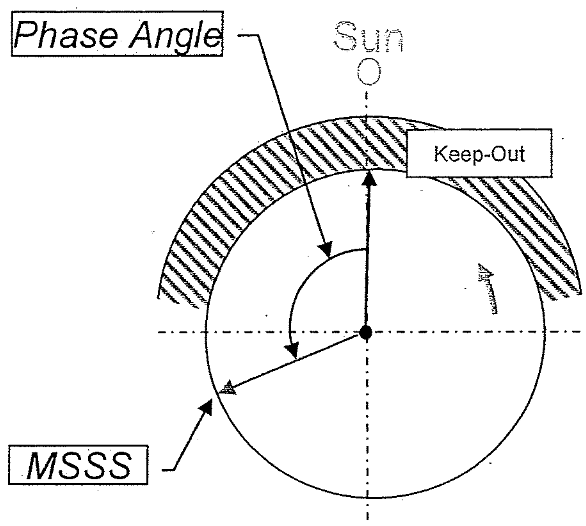


Figure 7: Attitude Precession Maneuver Restrictions Due to MSSS Anomaly

First of all, any command generated at the time of the true Sun pulse would probably not be executed. The thruster hardware would attempt to execute it at the time of the first spurious pulse, but since there are usually multiple spurious pulses, the command would be stopped before it could actually fire. This eliminates the first half of the keep-out zone. However, because the number of spurious pulses varies, it is possible that such a command could be executed. To ensure that thruster commands occur when they are desired, it is necessary to use a minimum delay time corresponding to the amount of time between the spurious and true Sun pulse. This means there will be no undesired firing triggered by the spurious Sun pulse, but it doubles the size of the keep-out zone needed to get desired movement of the spacecraft spin axis. While this created a sizable zone during which thruster firings were not permissible, it did not create any absolute restrictions on moving the spacecraft spin axis, since it would be possible to orient the axis as desired using multiple maneuvers, if necessary.

On-Orbit and Operational Tests and Mitigation

A number of on-orbit fixes were contemplated to deal with the spurious pulses, though only two were finally implemented. Had the ST5 mission been of greater duration, it is possible that additional fixes would have been performed. However, through the fixes that were used and operational methods, all three spacecraft were operated successfully during the duration of the mission.

The first on-orbit fix that was implemented was designed to eliminate the flood of buffer overrun messages generated by the spurious pulses. While this did not change the functionality of the spacecraft, it did allow the event message buffer to become more useful since other messages being sent down from the spacecraft could be seen.

The second on-orbit software change enacted was not implemented as a fix but was designed to collect more information and statistics to characterize the spurious pulses. Figure 8 shows a graphic of the some of the data extracted from this test on sc224. It shows the collection of one or more spurious pulses received each spin cycle in advance of the true Sun pulse. In most cases, there were multiple spurious pulses received within a very short amount of time; as many as ten pulses within 8 millisec were observed.

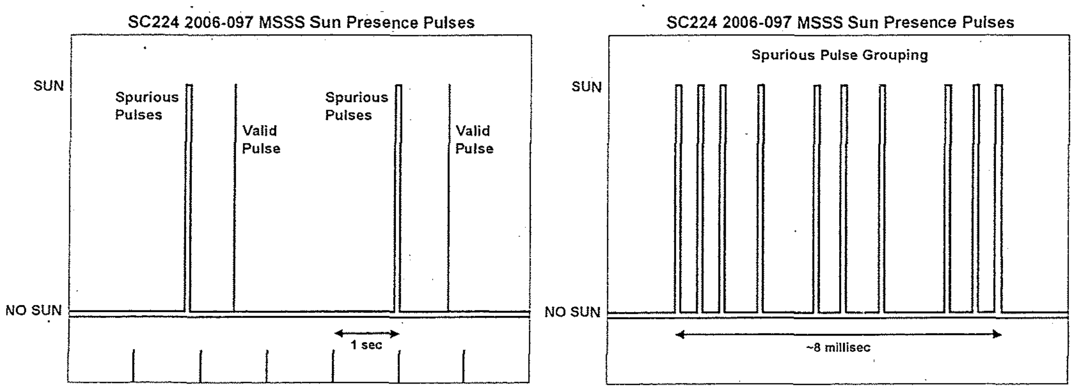


Figure 8: MSSS Sun Presence Pulse Characterization

The only other on-orbit fix applied related to the MSSS anomaly was designed to allow for correct spin rate calculation on the spacecraft. Because the spin rate calculation was based on the Δ times between adjacent pulses, the spurious pulses caused the calculated spin rate to be higher than actual. Given that the Δ times for the true and spurious pulses were approximately 20% and 80% of the true spin period, the calculated spin rates would be higher by a factor of 5 and 1.25, respectively. Existing software limits on the calculated spin rate eliminate the calculation that was five times too high, so the resultant spin rate calculated onboard was 25% too high. Because the onboard spin rate is used in sun acquisition and attitude precession modes to calculate the timing of thruster commands, it was important to have this number be accurate.

Recall from Figure 5 that pulses are occasionally dropped. Looking at data from each of the three spacecraft shows that five or ten times a day there is a spin cycle with no spurious pulses at all. This fact leads to the method used to calculate a correct spin rate with nothing but simple table changes to the flight software. The three steps needed were:

- Change the software limits on minimum and maximum spin rate to be ±10% of the expected value,
- Change the software first-order filter coefficients used to filter the spin rate to effectively disable the filter, and
- Wait.

By implementing the first two changes, the calculated spin rate "freezes" at the current value, which is 25% high, because the calculated value when there is a spurious pulse is always too high. However, when a spin cycle occurs where there is no spurious pulse, the spin rate calculation gives a correct spin rate within the allowable range, and the spacecraft begins to report and use the correct spin rate. This change was implemented on all three spacecraft, and within a day each spacecraft began reporting the correct spin rate.

CONSTELLATION REPLANNING & MANEUVER PERFORMANCE

More detail of the ST5 constellation planning and maneuvers is shown in Ref 1; it will be covered briefly here to point out the effect of the MSSS anomalies on the ST5 constellation. Figure 9 shows the nominal constellation planned and timeframe for maneuvers for the three ST5 spacecraft. The plan was for the middle spacecraft in the string-of-pearls constellation to be the reference spacecraft, with planned in-track separation between it and the leading and following spacecraft. After the initial post-deployment separation, the spacecraft would be maneuvered into two science validation constellations for approximately 30 days each.

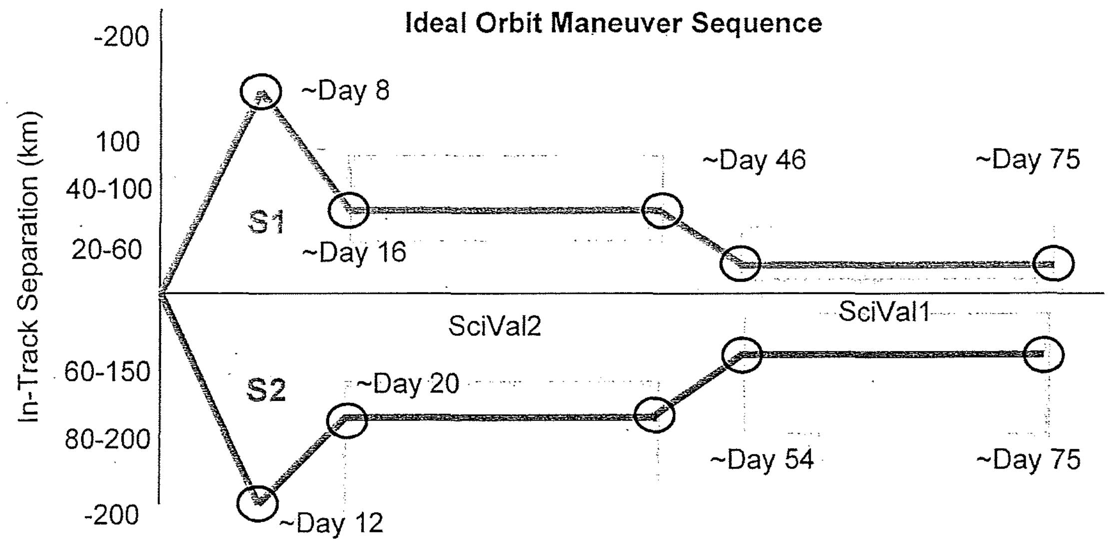


Figure 9: Nominal Constellation Plan

A number of things happened that prevented the original constellation plan from being implemented. As mentioned previously, the rise order of the constellation was different than expected, with sc094 the middle spacecraft instead of sc224. The forward spacecraft, sc155 was separating much faster than expected with respect to sc094 and sc224. To further complicate matters, the Pegasus rocket body ended up within the ST5 constellation, between sc155 and sc094. For the first week of the mission, there were tracking data processing issues that resulted on lost passes and a loss of quality of the orbit solution. Finally, the sun sensor anomaly raised early concerns about the ability of the spacecraft to perform sun acquisition and attitude precession operations. Until that capability was demonstrated, the project was reluctant to use the thruster for orbit maneuvers.

All told, this situation resulted in a three-week delay in maneuvering sc155 and resulted in a longer time to achieve formation than originally planned. Further, the presence of the Pegasus rocket body within the constellation greatly complicated the planning needed to establish the science constellations. Figure 10 shows the in-track separation derived from the definitive orbit determination of the forward (sc155) and aft (sc224) with respect to sc094. The width of each swath shows the relative separation dynamics of a lead-trail formation in an eccentric orbit. Also noted are all of the attitude precession (ATT) and orbit (ORB) maneuvers actually performed on each spacecraft.

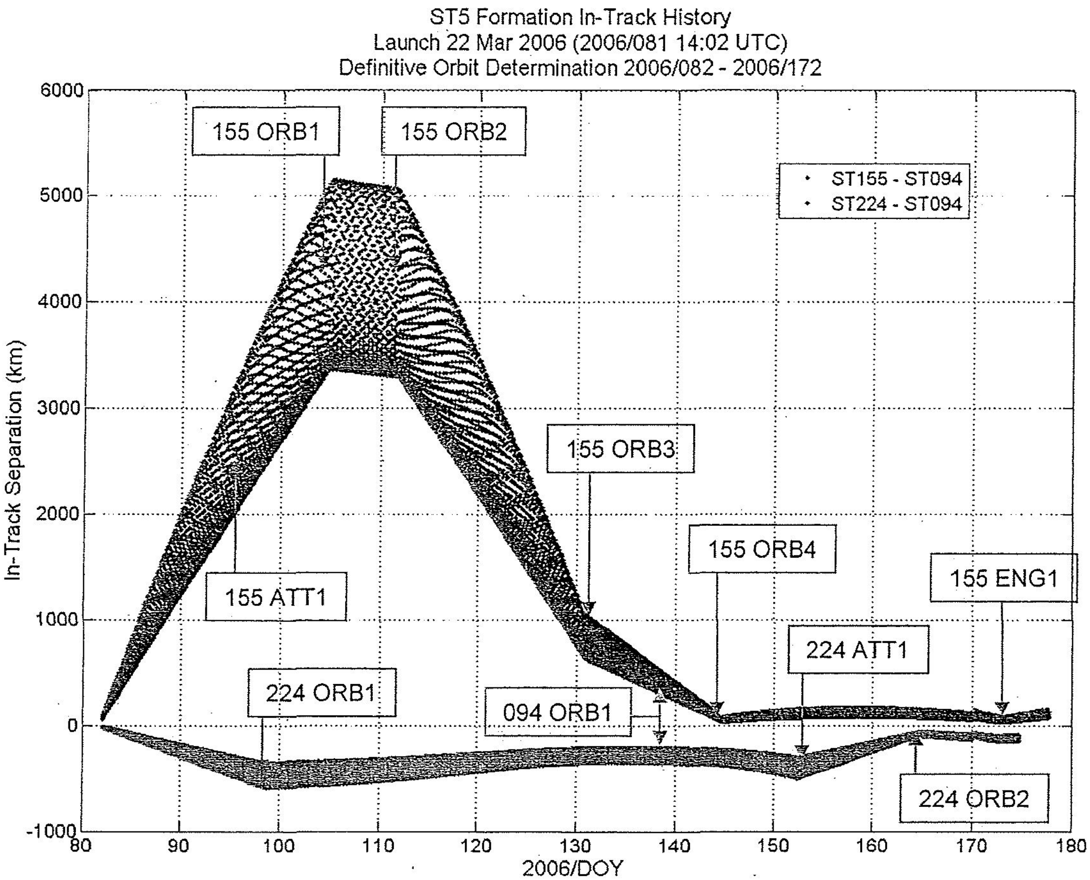


Figure 10: Achieved Constellation History

SPACECRAFT SPINDOWN

During the first week of operations of the ST5 constellation, the spacecraft flight operations, flight support, and anomaly teams were deeply involved in diagnosing and determining how to work with the spurious pulses being produced by the sun sensor and by operating the three spacecraft. It was during that time, however, that it was first noticed that the spacecraft were all spinning down at a faster rate than expected. Once the sun sensor anomaly investigation and operations worked out, more attention was paid to the spacecraft spin rate decrease and how it would affect future operations.

Initial Discovery

During the investigations of the sun sensor anomaly, the spacecraft data from the sensor were looked at and processed based on "short" and "long" Δ times. While the information that was received from Adcole helped to quickly determine the cause of the anomaly and whether the "short" or "long" Δ times corresponded to the true Sun pulse, the anomaly team continued to look at the data from both the true and spurious Sun pulses, as both were needed to calculate spacecraft spin rate, elevation, and other parameters.

Figure 11 shows a plot of the "short" and "long" Δ times for the first four days of the mission. It was at this point that it was first noticed that the trend of both sets of Δ times was increasing. Since adding adjacent Δ times gives the spacecraft spin period, it became clear that the spacecraft spin period was increasing and so their spin rates were decreasing.

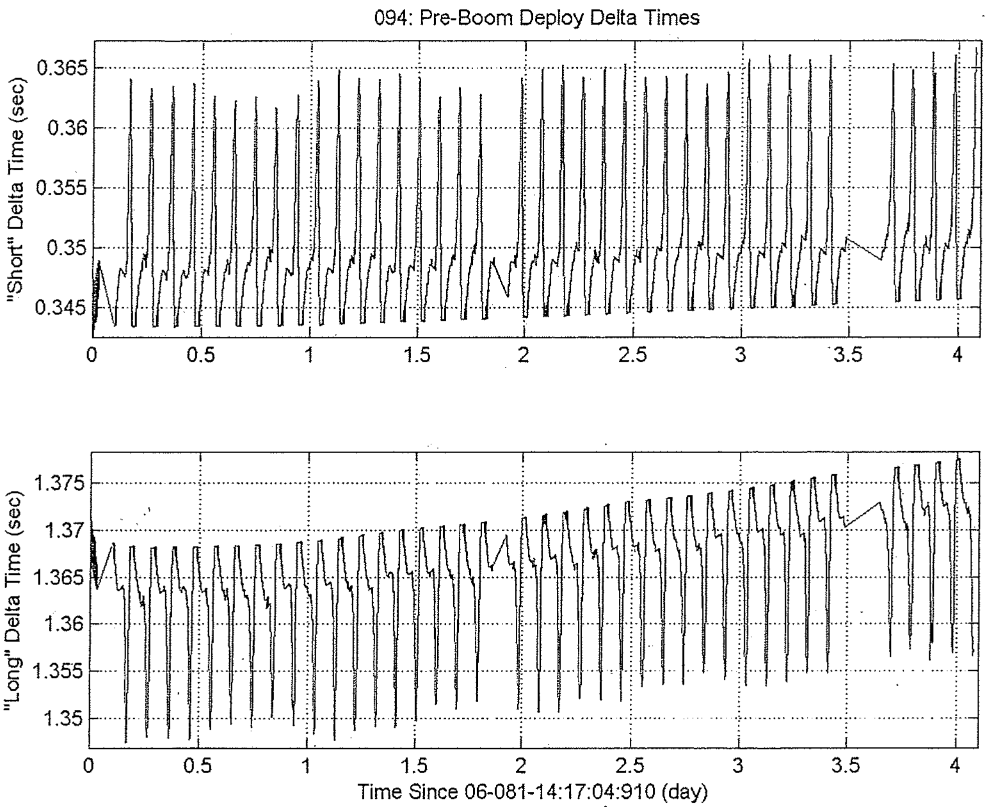


Figure 11: "Short" and "Long" Pulse Delta Times

Figure 12 shows the instantaneous spin rate of sc094 calculated at the time of each true Sun pulse over one orbit, centered at perigee. As can be seen, the spin rate does not uniformly decrease over the entire orbit. For this particular orbit of sc094, the spacecraft reaches its highest latitude around 1000 sec before perigee. Note that the spin rate is increasing around spacecraft apogee, with a sharper decrease around its point of highest latitude and perigee crossing.

Cause Investigation

Once the spin rate decrease was confirmed on all three spacecraft, and shown to be between approximately 0.05 and 0.1 rpm decrease/day, the team began investigating the possible causes. Because it was observed on all spacecraft, it was unlikely to be the result of a leakage from the cold gas thruster propellant system, and it was confirmed via telemetry that there were no unexpected pressure drops from the propellant tanks. Pre-launch analysis of environmental torques—atmospheric drag, solar pressure, gravity gradient, and magnetic from spacecraft residual dipole—was revisited and it was confirmed that they were too small to produce the observed effect. However, one additional environmental torque was examined to explain the spacecraft spin rate decrease that was not analyzed prior to launch. This additional environment disturbance was torque on the spacecraft caused by the interaction of the Earth's magnetic field with induced eddy currents.

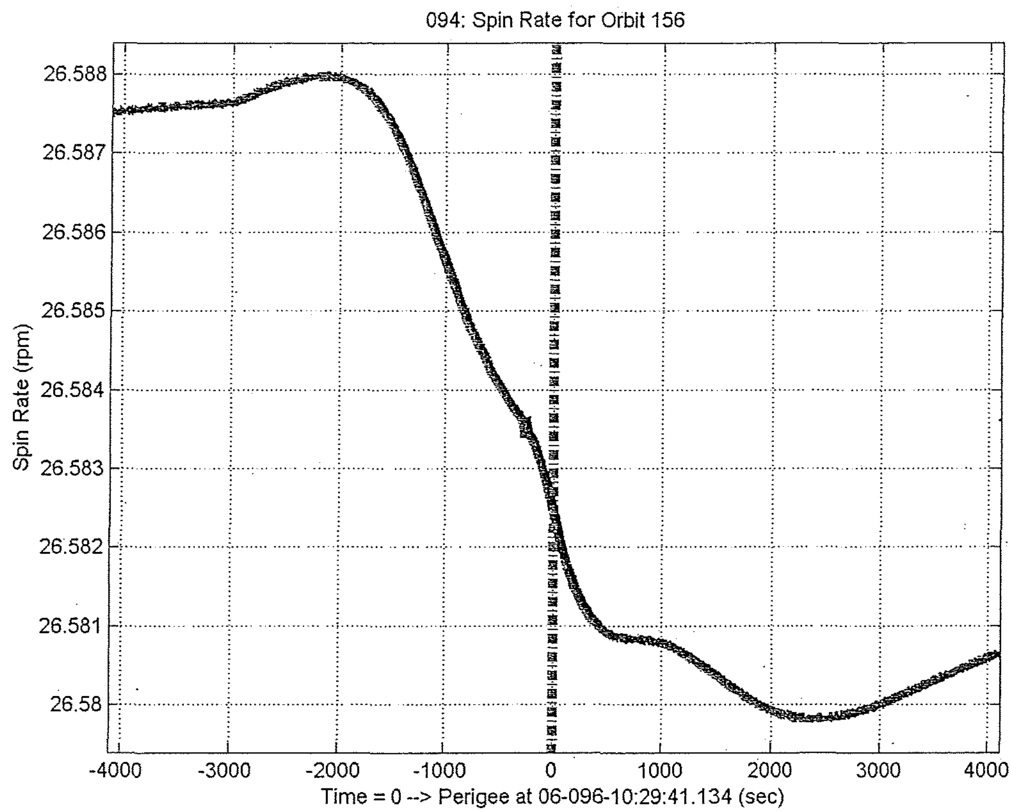


Figure 12: Spacecraft 094 Spin Rate Orbit 156

Using Ref. 2, Dean Tsai of the GN&C flight support team conducted a simulation of the spacecraft over several orbits using the 2005 IGRF-10 magnetic field model. The model included the residual magnetic moment of the ST5 spacecraft, as measured in ground testing, along with an effect due to eddy currents in the spacecraft structure taken from Ref. 2 and scaled to the size of ST5. Figure 13 shows the results of this simulation, with the red dashed line representing the results of the eddy current simulation and the solid blue line showing representative data from sc094 (this plot shows spin rate decrease). This analysis is especially interesting because it provides an explanation for both the general downward trend of the spin rate as well as the periods during the orbit in which the spin rate is increasing: the increase is caused by the residual magnetic dipole while the overall decrease is caused primarily by the induced eddy currents. Guan Le and Jim Slavin of the ST5 Science Team also provided another potential explanation of the observed spacecraft spin down, also caused by interaction of the Earth's magnetic field with the spacecraft. The mechanism, which is similar to eddy currents within the spacecraft structure, is instead carried by leakage current from the spacecraft structure being closed through the surrounding plasma of the Earth's magnetic field. While the project did not have the resources to do enough investigation to identify the exact mechanism for this spin down, the analysis that was performed pretty conclusively showed that it was primarily caused by interaction of the spacecraft with the Earth's magnetic field.

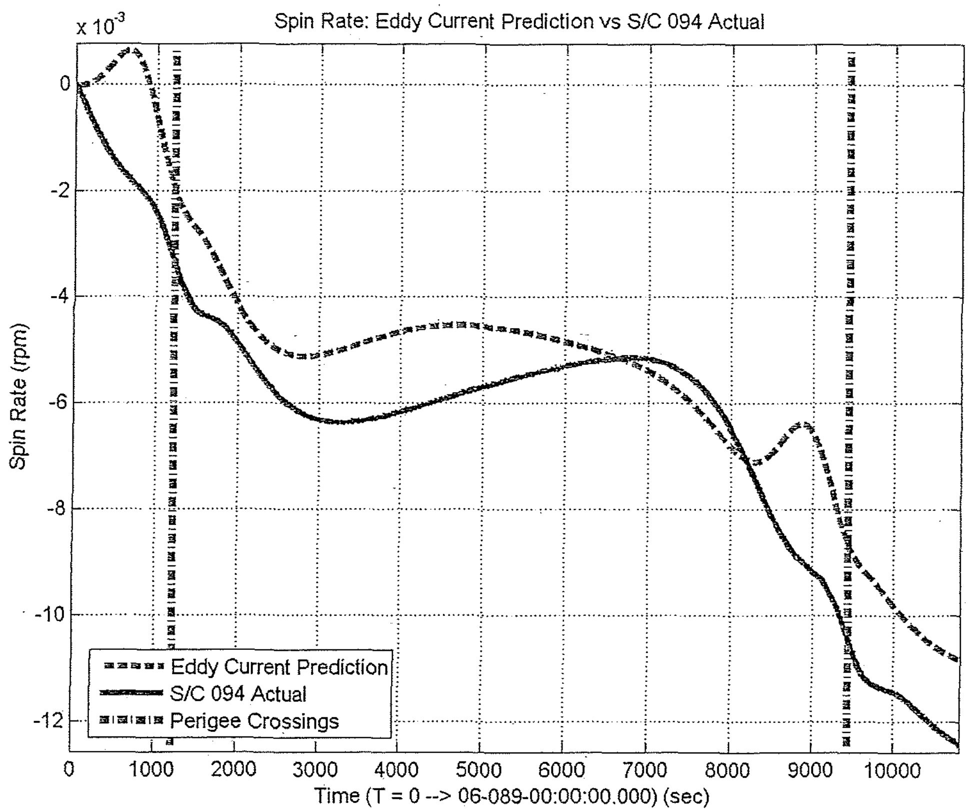


Figure 13: Eddy Current Spindown Prediction vs Spacecraft Data

Mitigation

Because the three ST5 spacecraft each had only one thruster nominally aligned with the spin axis, there was very little that could be done to mitigate the spin rate decrease. Trending analysis showed that the spin rates would not get too low to have a negative effect on spacecraft stability within the planned 90 day mission, but there was some concern for an extended mission. It was hoped that with luck, misalignments of the one thruster with respect to the spin axis might result in a spin rate increased when maneuvers were performed. As shown by Figure 14, which plots the spacecraft spin rate for each spacecraft around its first orbit maneuvers, it was true that with sc094 and sc224, the spin rate increased during the maneuvers. For sc155, however, the spin rate further decreased. In no case, however, was the spin rate change very significant, being the same order of magnitude as roughly one day of spin rate change.

During the investigation of the cause of the spacecraft spin down, it was noted that one of the reasons that the eddy current effect was unexpectedly large might be due to the fact that the ST5 spacecraft had a large metal card cage that ended up being perpendicular to the Earth's magnetic field as the spacecraft approached its point of highest latitude and perigee. Normally, small low Earth orbiting spacecraft will be designed to avoid this, but when ST5 was designed it was planned to be in a higher orbit. This suggested one possible experiment that could be made to both confirm the influence of the Earth's magnetic field as a cause of the spin down and to mitigate its effect. The experiment was to reorient the spin axis of one of the spacecraft about the sun line so that when the spacecraft went through perigee the magnetic field lines would not be as perpendicular. This experiment was performed by reorienting sc224. The results of this maneuver on the rate of spacecraft spin rate decrease are shown in Figure 15. These results provide further confirmation of torque caused by the Earth's magnetic field as the primary contributor to the spacecraft spin down.

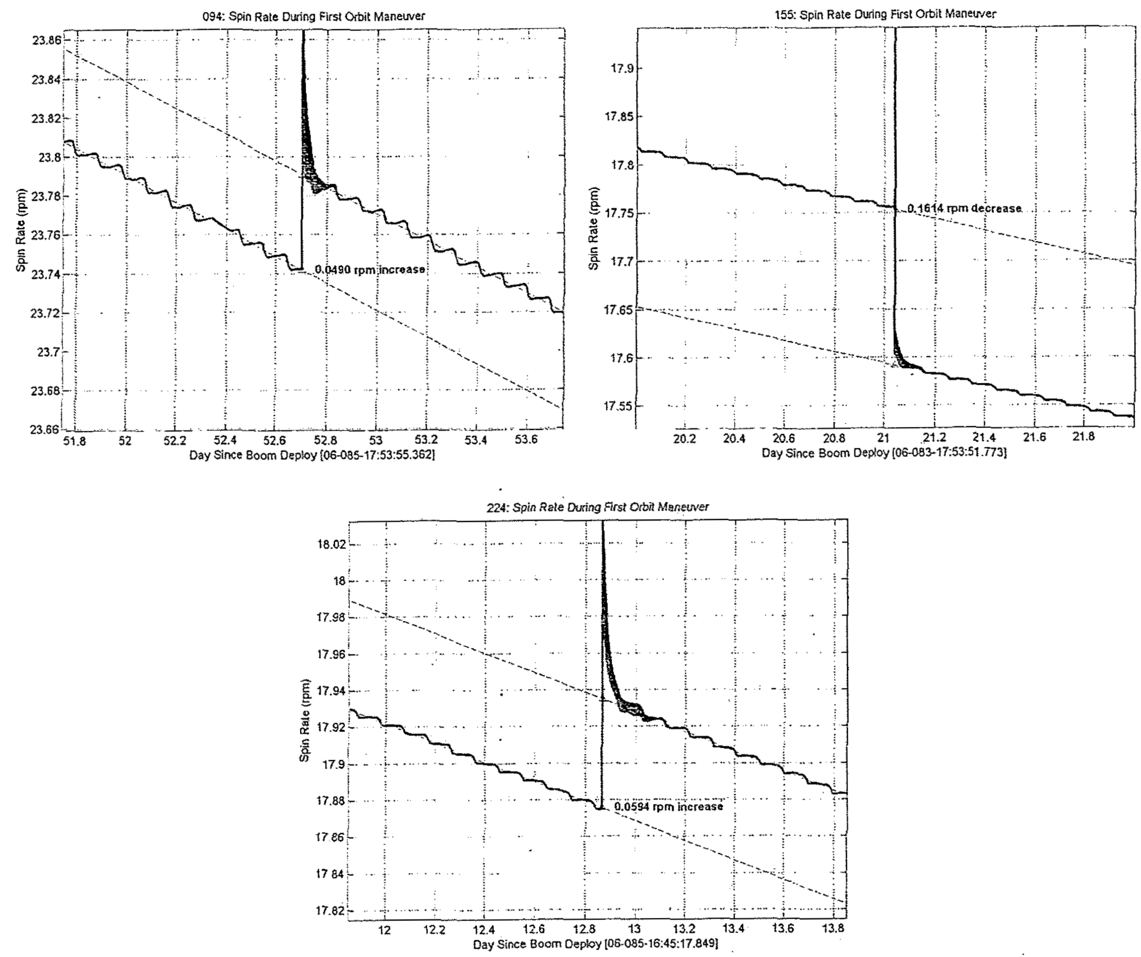


Figure 14: Effect of Orbit Maneuvers on Spacecraft Spin Rate

PASSIVE NUTATION DAMPER PERFORMANCE

Figure 16 shows a picture of one of the passive nutation dampers designed and built at Goddard Space Flight Center for use on the ST5 spacecraft. The device passively damps out spacecraft nutation, which is defined as the angular displacement between the spacecraft's major principle (spin) axis and the spacecraft's angular momentum vector and occurs on ST5 following spacecraft separation from launch vehicle, magnetic boom deployment, and thruster firings. The dampers were fully filled with a viscous silicone fluid. No bellows mechanism was used, thereby giving the damper a high internal pressure as well as reducing mass and complexity. The ST5 dampers were designed to operate between spin rates of 15 and 40 rpm, and to have a time constant of 45 minutes at a nominal spin rate of 28 rpm.

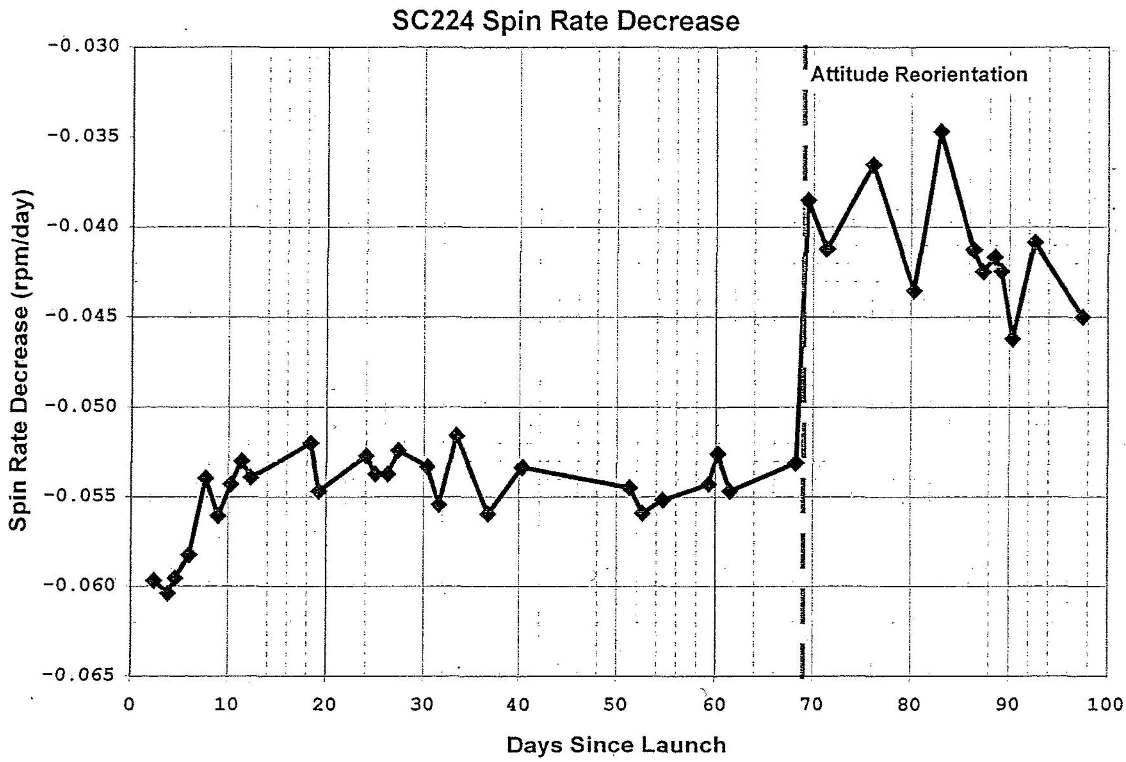


Figure 15: Spacecraft 224 Spin Rate with Attitude Reorientation

Figure 17 shows an example of the nutation damper performance, plotting the elevation angle of sc155 after deployment of the magnetic boom. The red dashed line shows the envelope of the nutation damping and can be used to calculate the nutation time constant. Table 1 shows a tabular listing of each "disturbance" event on the three ST5 spacecraft, with the corresponding spin rate and calculated nutation time constant. This information is depicted graphically in Figure 18. The nutation damper performed as designed.

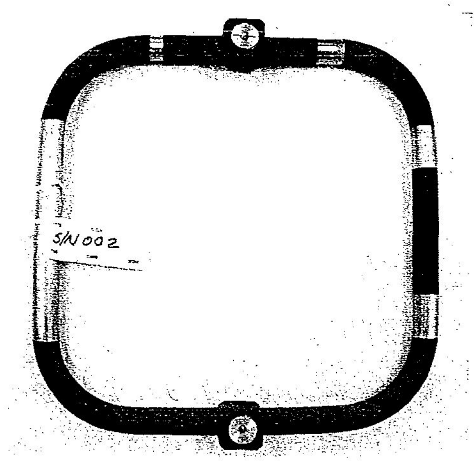


Figure 16: Nutation Damper

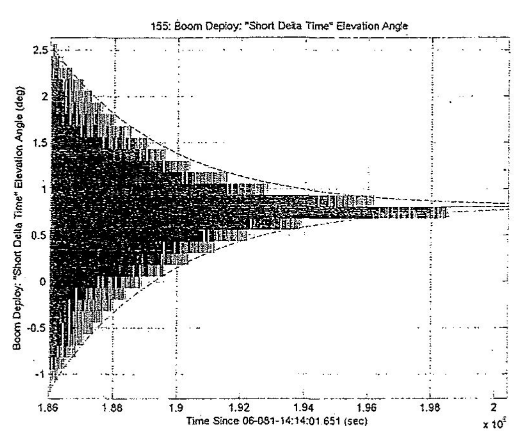


Figure 17: Damper Performance for S/C 155

Table 1: Nutation Damper Performance

Spacecraft	Event	Post-Event Spin Rate (rpm)	Time Constant (min)
094	Initial Release	35.0	32
094	MAG Boom Deploy	27.4	30
094	20 pulse ΔV	23.8	39
094	600 pulse ΔV	20.8	47
094	1500 pulse ΔV	21.7	46
155	Initial Release	24.6	50
155	MAG Boom Deploy	19.2	51
155	6 pulse Att Mnvr	18.2	59
155	155 pulse ΔV	17.6	64
155	140 pulse ΔV	17.1	68
155	130 pulse ΔV	15.7	70
155	76 pulse ΔV	15.0	79
155	400 pulse ΔV	13.0	103
155	1500 pulse ΔV	11.8	124
224	Initial Release	23.8	51
224	MAG Boom Deploy	18.6	64
224	47 pulse ΔV	17.9	66
224	76 pulse Att Mnvr	15.1	85
224	20 pulse ΔV	14.6	87
224	600 pulse ΔV	13.6	100
224	1500 pulse ΔV	13.0	115
224	1500 pulse ΔV	12.7	112

SPACECRAFT DISPOSAL

On June 1, 2006, the ST5 Project received a Notice of Intent to Terminate Operation of Space Technology 5 Mission letter from the NASA Science Mission Directorate, directing the project to complete end-of-mission activities and cease operations at completion of primary mission no earlier than June 30 and no later than July 7, 2006. These dates allowed ST5 to complete its planned 90 day mission. The end-of-mission (EOM) activities for ST5 were fairly simple: disable the onboard spacecraft failure detection and correction, empty the propulsion tanks, conduct follow-up passes, and turn off transmitters by not scheduling any further contacts. After conducting these operations, the spacecraft would re-enter the atmosphere well within 25 years and with no debris field, meeting NASA orbital debris requirements.

The objectives of the EOM maneuver plan were to fire all of the remaining propellant, configure the spacecraft orbits for eventual disposal (via orbit decay), and minimize recontact probability. The constraints were on spacecraft attitude, available propellant, and operating condition limits on the propulsion system. Between June 26 and June 29, 2006, a series of 11 EOM thruster firings were successfully conducted on the three ST5 spacecraft and final contact with them was on June 30, 2006.

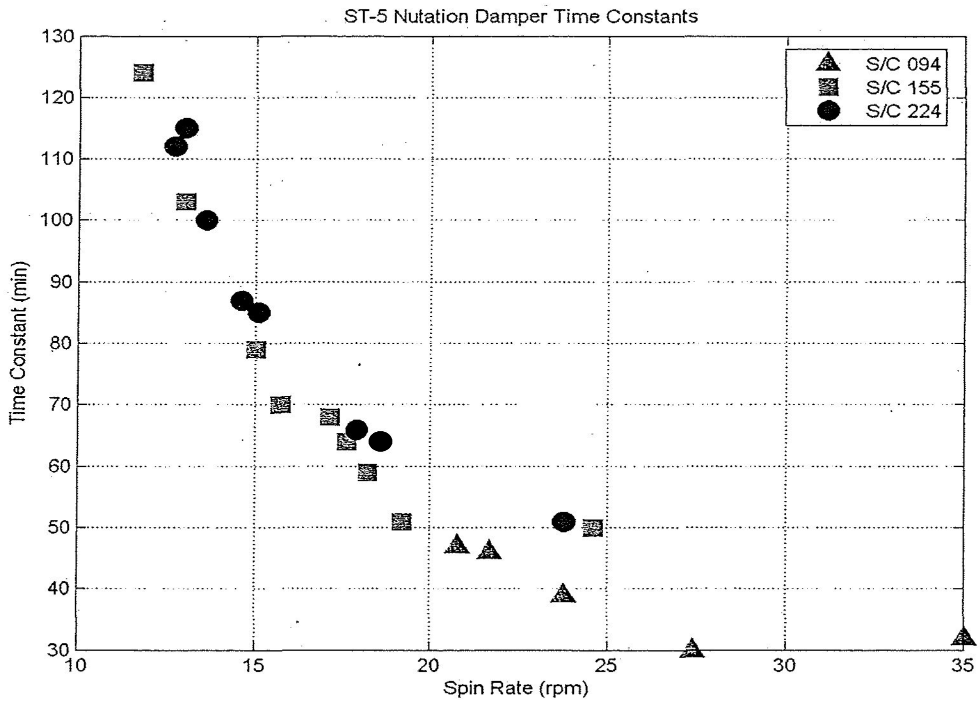


Figure 18: Nutation Damper Performance

CONCLUSION

During a relatively brief 100 day mission and in the face of a number of anomalies, the three ST5 spacecraft were able to successfully satisfy the mission's level one requirements, primarily to design, build, launch and operate three small spacecraft as a constellation in order to achieve accurate, research-quality science measurements. Additionally, ST5 successfully demonstrated a number of new technologies including a miniature communications transponder, variable emittance thermal coatings, a cold-gas micro-thruster, CMOS ultra-low power radiation-tolerant logic, a low voltage power subsystem including LiIon battery, and software tools for autonomous ground operations. The project held the ST5 Technology Symposium to present their results at Goddard Space Flight Center on September 13, 2006.

REFERENCES

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2. N. W. Tidwell, "Modeling of Environmental Torques of a Spin-Stabilized Spacecraft in a Near-Earth Orbit", *Journal of Spacecraft and Rockets,* Vol. 7, No. 12, pp. 1425–1433, December 1970.