Pulsed Plasma Thruster (PPT) Validation Report

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1. INTRODUCTION

The Earth Observing-1 (EO-1) Pulsed Plasma Thruster (PPT) is a new generation advanced propulsion technology that was flown to demonstrate, for the first time, the PPT technology's ability to serve as a precision attitude control actuator for spacecraft. PPTs can offer spacecraft significant mass saving by replacing the combinations of reaction wheels, torque rods, and chemical thrusters.[1,2] They have the advantage of being simple to integrate into spacecraft because of their limited mechanical mounting hardware and electrical requirements. PPTs also eliminate safety and component layout complexities associated with fluid-propellant propulsion systems.

The EO-1 PPT is the first flight PPT developed in more than 10 years. It incorporates significant improvements over the previous generation of PPTs. The EO-1 PPT flight experiment was designed to validate this new generation of PPT in three key areas: 1) to demonstrate the ability of the PPT to function as a precision attitude control actuator, 2) to confirm benign interaction with other spacecraft subsystems and instruments, and 3) to verify performance parameters in flight. The flight validation of this technology was the collaborative effort of NASA Goddard Space Flight Center (GSFC), NASA Glenn Research Center (GRC), General Dynamics Ordnance and Tactical Systems (formerly Primex Aerospace Corporation), and Swales Aerospace in partnership with the Hammers Company.

The PPT technology validation consisted of operating the PPT as a replacement for the pitch wheel in EO-1's attitude control loop to counter spacecraft disturbances. The spacecraft disturbance forces include solar pressure torques, aerodynamic drag, gravity gradient torques, and bearing friction and torques generated by the solar array drive.

This report will describe the results from three on-orbit test opportunities. The initial testing took place between January 4, 2002, and January 7, 2002, the second test opportunity was on March 14, 2002 and the third was on April 25, 2002. During these tests, the PPT provided pitch axis torque after the pitch axis momentum wheel and magnetic torquer bar had been disabled. This substitution occurred in the nadir pointing mission mode of the spacecraft.

2. TECHNOLOGY DESCRIPTION

2.1 PPT Hardware Description

The EO-1 PPT is a small, self-contained electromagnetic propulsion system that utilizes solid Teflon propellant. It can deliver high specific impulses (650 – 1400 sec), very fine impulse bits (90 – 860 micro N-sec) at low power levels (12 - 70 W) and an estimated total impulse of 460 N-sec. The total mass is 4.95 kg. Figure 1 shows the PPT mounted to the spacecraft while the spacecraft is attached to the launch vehicle.

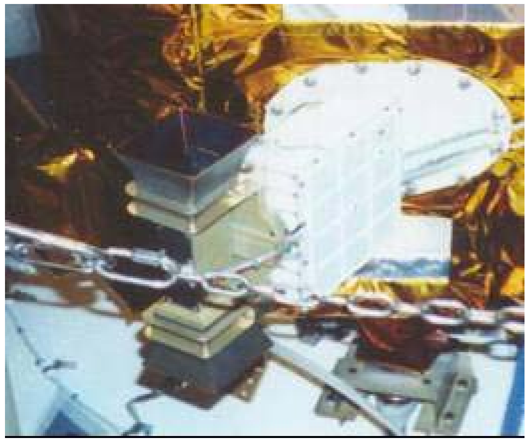

Figure 1. EO-1 Spacecraft With PPT

The important performance characteristics are given in Table 1.

Table 1. EO-1 PPT Characteristics

Impulse Bit (per pulse)	90 – 860 micro N-sec
Specific Impulse	650 – 1400 sec
Pulse Frequency	1 Hz
Main Capacitor Energy	8.5 – 56 J
Overall Thrust/Power Efficiency	8%
Total Mass	4.95 kg
Fuel Mass	0.07 kg per side
Total Impulse (estimated)	460 N-sec
Orbital Average Power (estimated)	12.6 W

The operation of the PPT is inherently simple. Referring to Figure 2, the main capacitor is initially charged to the desired level and then discharged across the face of a Teflon fuel bar. Two fuel bars are located between separate and opposing electrode pairs to provide thrust in the positive and negative Z-axis directions (the PPT location with respect to the center of mass results in +/- pitch torque). The discharge of the main capacitor occurs when the spark plug on the desired electrode pair is commanded to fire. A minute amount of charged particles is ablated into the electrode gap when the spark plug is fired. These charged particles provide a conductance path that initiates the main capacitor discharge across the gap. The main capacitor discharge ablates a small amount of Teflon. A small percentage of the Teflon is ionized to form plasma. A Lorentz force accelerates the plasma and thus produces thrust. Charged-particle to neutral-particle collisions and pressure forces from resistive heating produce additional acceleration of the neutrally charged, ablated Teflon plasma.

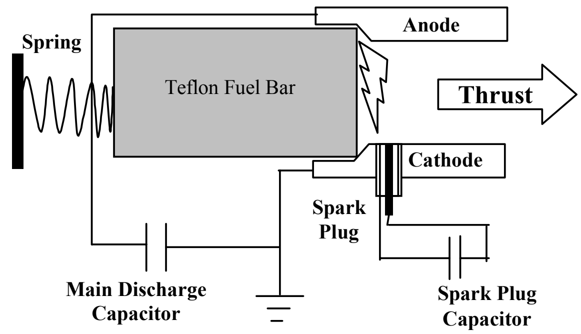

Figure 2. PPT Diagram

To modulate the thrust, the magnitude of the impulse bit is varied. Varying the charge time of the main capacitor changes the magnitude of the impulse bit. The length of time the main capacitor charges directly affects the amount of energy in the capacitor and, consequently, the amount of thrust produced during the discharge. The relationship between the impulse bit per firing as a function of capacitor charge time is shown for each firing side of the PPT in Figure 3.

The differences in performance between the two sides are most likely a result of slightly different spark plug characteristics and electrode properties. The most significant source of shot-to-shot variability comes from the PPT temperature change. The maximum and minimum impulse bits obtained at the maximum and minimum PPT operating temperatures are also plotted in Figure 3.

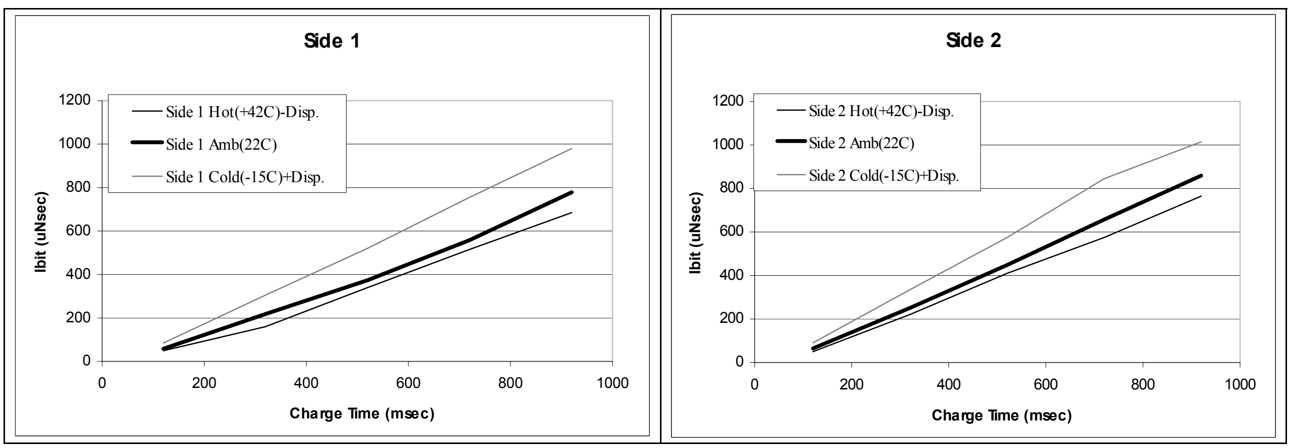

Figure 3. PPT Performance

2.2 PPT to Spacecraft Interface

The mechanical and electrical interfaces between the PPT and the spacecraft bus are relatively simple. The mechanical mounting and orientation of the PPT on the spacecraft are shown in Figure 4. Two harnesses, one for power and one for command and telemetry, are used for the electrical interface. The power harness connects the PPT to a power subsystem output module providing 28 +/-7 V from the unregulated spacecraft power bus. The command and telemetry harness connects the PPT to the Attitude Control Electronics (ACE) Input/Output card. The services provided are given in Table 2:

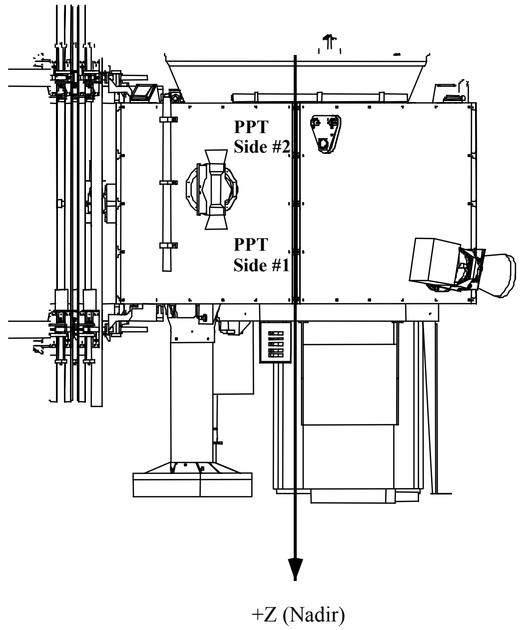

Figure 4. Orientation on Spacecraft

Table 2. PPT Command and Telemetry

PPT Commands:
Charge Main Capacitor (0-5V logic)
Fire Spark Plug # 1 (0-5Vlogic)
Fire Spark Plug # 2 (0-5V logic)
PPT Telemetry:
Main Capacitor Voltage
Spark Plug #1 Capacitor Voltage
Spark Plug #2 Capacitor Voltage
Main Capacitor Temperature
PPT Transformer Temperature
Fuel Bar #1 reading (length of bar)
Fuel Bar #2 reading (length of bar)

The PPT commands are sequenced by the attitude control subsystem that operates at a 1 Hz cycle rate. The ACE operates at a faster 25 Hz rate. The ACE rate allows PPT commands to be quantized in 40-msec increments within each one-second attitude control software cycle. Therefore, one ACE cycle is equal to 40 msec. The charge time length was initially set to be between 160 msec (4 ACE cycles) and 920 msec (23 ACE cycles). The minimum charge length is determined by the minimum energy at which the PPT will reliably discharge. The maximum time is set by the need to charge the main capacitor, fire the appropriate spark plug, and discharge the main capacitor to 0 V within one cycle of the 1-Hz cycle rate. Since 80 msec are required to fire a spark plug and discharge the main capacitor, the maximum allowable time for main capacitor charging is limited to 920 msec or 23 ACE cycles.

2.3 PPT Closed Loop Control Design

To achieve the minimum objective associated with flight validating the advanced PPT technology for performing the attitude control function within the cost and schedule constraints of the EO-1 mission, the impacts to the baseline attitude control subsystem were minimized. This approach led to the fundamental experiment design choices of using a single PPT unit to replace the function of the pitch axis momentum wheel for a limited time during the mission and operating the PPT at a fixed firing frequency (1 Hz) while varying the PPT power level to achieve thrust variability.

For PPT closed-loop operation, the on-board control algorithm is a simple derivative of the controller used for normal 3-axes reaction wheel operation. This algorithm is described in detail in Reference 1. The pitch torque commands from a Proportional-Integral-Derivative (PID) controller are processed for PPT actuation instead of wheel actuation. The pitch integral control gain constant is the only adjustment to the PID controller. During PPT control mode, the roll and yaw axes continue to be controlled by their associated reaction wheels and torquer bars.

The spacecraft is sent into PPT pitch control by a single command. The transition occurs in two phases. In the first phase, the pitch reaction wheel momentum is compared to the momentum limit for PPT transition. If the pitch reaction wheel momentum exceeds the transition momentum limit, the appropriate PPT is selected to fire full-on to provide a torque on the spacecraft in the same direction as the reaction wheel. The PPT torque will cause the reaction wheel to spin down, and this scenario will continue until the reaction wheel momentum drops below the transition threshold. In the second phase, once the pitch reaction wheel drops below the transition threshold, the PPT will automatically take over closed loop pitch control of the spacecraft. A zero torque command will be sent to the pitch reaction wheel so it will continue to spin down until it stops

3. TECHNOLOGY VALIDATION

As stated above, the validation objectives of the EO-1 PPT technology are to:

1) Demonstrate precision attitude control capability
2) Confirm benign interaction with the spacecraft bus and instruments
3) Confirm performance parameters in flight

This interim report will describe the ground test verification and the to-date on-orbit testing. Subsequent on-orbit testing is planned to complete PPT validation objectives.

3.1 Ground Test Verification

3.1.1 Component Testing

The EO-1 PPT component flight unit underwent extensive ground protoflight hardware validation and development testing. The major tests are summarized in Table 3. A more detailed description can be found in Reference 3.

Table 3. Ground Validation Tests

Functionality Tests	Performed at the bench top and vacuum chamber level to verify and map the electrical characteristics of the unit. The PPT was throttled through the range of charge durations (160 - 920 msec).
Performance	For details see Reference 3. Measured thrust and impulse bit as a function of PPT charge time before and after life testing. Evaluated off-axis thrust components. Characterized shot-to-shot repeatability.
Vibration	Acceptance level vibration tested to Delta II levels. (Random vibration tested to 14.1 grms in 3 axes).
Thermal Vacuum	4 thermal cycles demonstrated survival and operations across required temperature range -32 to +42°C survival range -15 to +42°C operating range Verified performance at temperature plateaus. Characterized sensitivity in main capacitor charge rate to temperature.
Life/Contamination	Demonstrated thruster minimum experiment life (100,000 pulses/side). Evaluated plume contamination effects on spacecraft surfaces. – Spacecraft mock-up with surface samples (X-band antenna surface, radiator, MLI).
EMI/EMC	Characterized conducted and radiated emissions – Consistent with previous electric propulsion devices – RE01, CE01 and CE07 results within spec – CE03 limits (conducted emissions) exceeded by up to 12 dB below 4 MHz – waiver accepted. RE02 broadband radiated emissions exceeded levels below 100 MHz.

Results from the electromagnetic interference/electromagnetic compatibility (EMI/EMC) tests eventually led to major concerns about the effect the PPT-radiated emissions would have on the EO-1 spacecraft and instruments. The radiated emissions were 60-80 dB above specifications for a significant portion of the specification frequency range. Initially the Advanced Land Imager (ALI) instrument, and then the Hyperion instrument teams became concerned that these emissions might harm sensitive electronics. Because of these concerns, the on-orbit tests of the PPT were delayed until after the completion of the validation of all the EO-1 instruments. A thermal vacuum firing test of the PPT at the spacecraft level was also eliminated. To allow the PPT to be functionally tested at the spacecraft level at ambient conditions before launch, an EMI shield was implemented. The details of the EMI tests results and instrument concerns are given in Reference 4.

At the completion of the EO-1 baseline mission for all of the other EO-1 technologies, an extensive review of PPT EMI effects on the spacecraft and instruments was performed. Analysis and additional ground testing with a breadboard PPT unit demonstrated that the PPT-radiated emissions posed minimal risk to the spacecraft and instruments and permission was given to proceed with the on-orbit testing of the PPT. Due to the lack of resources to analyze existing plume contamination data, the Hyperion instrument team requested that the Hyperion be powered off and put into an out-gassing mode during all PPT operations prior to operation of Hyperion at the end of EO-1 mission life.

3.1.2 Spacecraft Level Test

The PPT was tested at the spacecraft level in essentially three different test configurations. In the initial set of tests, the PPT was electrically integrated to the spacecraft while it was located in a vacuum bell jar located in the proximity of the spacecraft bus. At the time of these tests no instruments where integrated to the spacecraft. The second test configuration consisted of having the PPT electrically and mechanically integrated into the spacecraft with PPT Ground Support Equipment (GSE) attached to the thruster electrodes to allow simulated firings in ambient atmosphere. Because of concern raised about possible

PPT EMI effects on the spacecraft instruments, a third test configuration was implemented in which a EMI shield was placed around the PPT and the PPT GSE as mentioned above.

The spacecraft-level tests included sending discrete fire commands to the PPT as well as closed-loop autonomous commanding of the PPT by the EO-1 attitude control system. Hundreds of shots were accumulated over the entire range of PPT charge levels. The only anomalies to occur during these tests were associated with PPT GSE. These problems were corrected and the PPT and spacecraft performed as expected. The results of all of the PPT spacecraft-level tests confirmed proper command and telemetry to and from the PPT and PPT electrical functionality.

3.1.3 Attitude Control Simulations

A comprehensive nonlinear analysis was performed using a high-fidelity MATLAB-based simulation to evaluate the EO-1 attitude control performance with the PPT as the pitch axis actuator. The simulation included mass property variation, ephemeris models, and torques due to solar pressure, aerodynamic drag, gravity gradient, solar array movement, magnetic torquer bar momentum management, and momentum wheel friction. It should be noted that the movement of the ALI telescope door prior to imaging was not part of these simulations.

Shown in Figure 5 are the results of a standard EO-1 orbit. The plots represent pointing errors from all three axes during the course of a complete orbit, derived from the difference between the desired attitude and integrated Inertial Reference Unit (IRU) measurements. The largest disturbance in the simulator model, and therefore the event requiring the largest output from the actuators, is the acceleration and deceleration of the solar array during orbit night rewind activities. For the entire orbit, the worst-case roll, pitch, and yaw errors were found to be 1.25 (260), 0.8 (165), and 1.1 (210) x 10-3 rad (arcsec). Shown in Figure 6 are the corresponding PPT charge commands for the simulated orbit. The torque required from the PPT is below its maximum capability, which corresponds to 23 ACE charge cycles, for the majority of the orbit. The maximum torque is only required at the initial transition into the PPT control mode and briefly during solar array rewind. The simulations showed that the PPT has adequate capability to perform attitude control during science data collection for purposes of hardware validation.

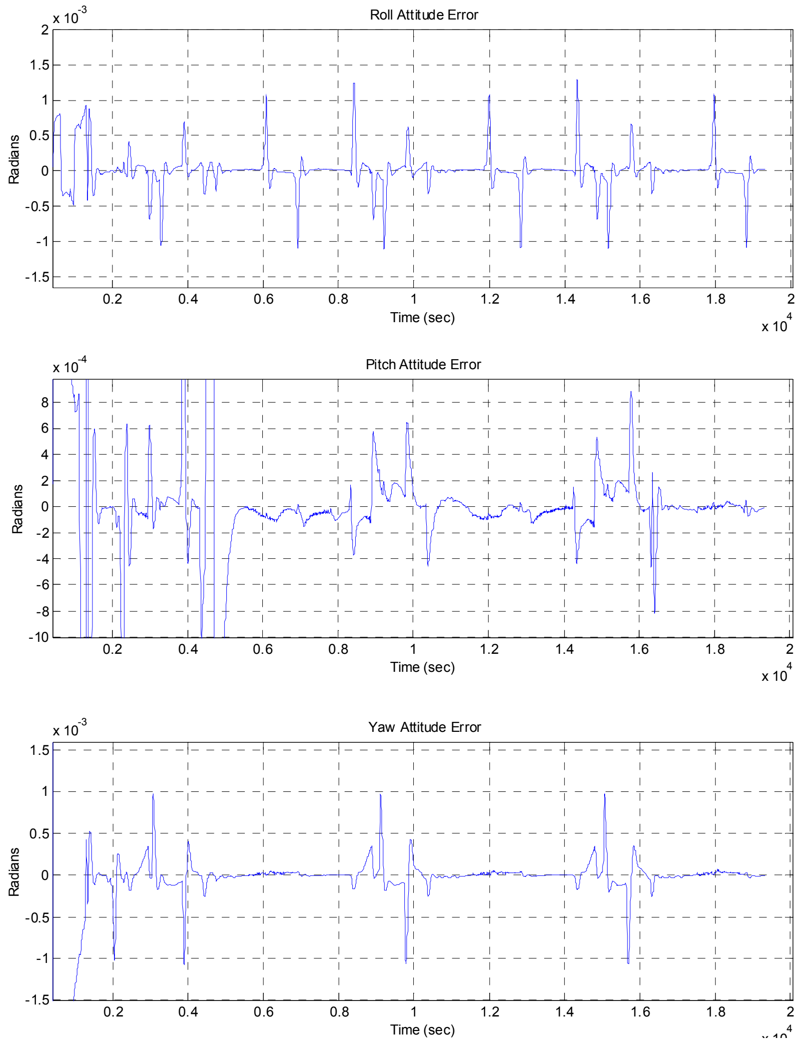

Figure 5. Simulator Predictions of Attitude Errors for PPT Control Mode

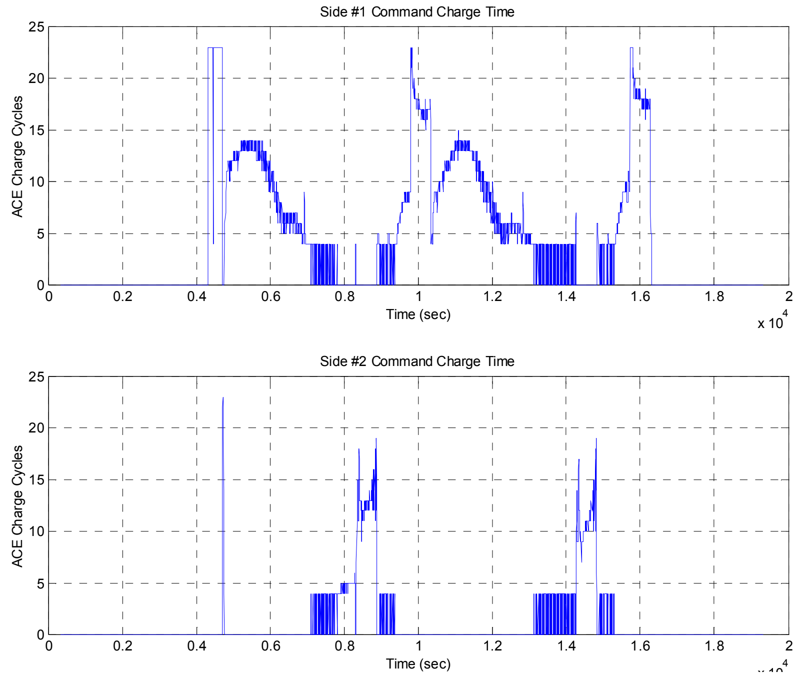

Figure 6. Simulator Predictions of PPT ACE Charge Commands for PPT Control Mode

3.2 On-Orbit Test Validation

The initial validation of the PPT was performed January 4-7, 2002. Additional testing of the PPT occurred on March 14, 2002 and on April 15, 2002. The results of the initial and second series of tests as well as preliminary results from the third series of tests will be covered in this report. The major events of each test period are outlined below and discussions of the major validation elements are presented in the following subsections.

3.2.1 Initial PPT Testing, January 4-7, 2002

These initial PPT tests were conducted according to the steps outlined in the PPT Verification Plan. The tests consisted of incrementally verifying PPT functionality as well as spacecraft and instrument health. Table 4 provides a brief chronology of the events. The test firings and short duration closed-loop control activities culminated in four hours (2.4 orbits) of continuous PPT pitch axis control on January 7. During this time period, the attitude commanding was nadir pointing with yaw steering enabled. Three Data Collection Events (DCEs) were performed with the ALI under PPT control to validate pointing performance for science observations and to evaluate whether there were any adverse effects on the instrument. It should be noted that the Hyperion instrument was powered off and remained off and in an out-gassing state upon request of the Hyperion instrument team. Tests with the PPT operating during normal Hyperion imaging are planned at the end of EO-1 mission life.

Table 4. PPT Test Period #1 Activity Sequence

Day	UTC	Activity	Comment(s)
Wednesday Jan 2	002-1918	PPT Power ON	Nominal
Friday Jan 4	004-1459	Spark Plug 1, 4 cycles, 1 sec	Good Discharge
	004-1501	Spark Plug 2, 4 cycles, 1 sec	No Discharge
	004-1503	Spark Plug 2, 4 cycles, 1 sec	Good Discharge
	004-1534	ALI Lamp Cal	Nominal

Day	UTC	Activity	Comment(s)
Saturday	005-1427	Spark Plug 1, 23 cycles, 1 sec	Good Discharge
Jan 5	005-1427	Spark Plug 2, 23 cycles, 1 sec	Good Discharge
	005-1427	Spark Plug 1, 4 cycles, 1 sec	Good Discharge
	005-1441	ALI Lamp Cal	Nominal
	005-2355	Spark Plug 1, 23 cycles, 10 sec	Nominal
	005-2355	Spark Plug 2, 23 cycles, 10 sec	Nominal
	006-0129	Spark Plug 1, 23 cycles, 60 sec	Nominal
		Numerous NOOP commands	Validated Uplink
	006-0301	Spark Plug 2, 23 cycles, 60 sec	Nominal
		006-0303	ALI Lamp Cal
Sunday	006-1408	PPT control Enabled	Nominal
Jan 6	006-1415	PPT control DISabled	Nominal
	006-1524	PPT control Enabled	Nominal
	006-1535	ALI Lamp Cal	Nominal
	006-1555	Eclipse Entry, SAD Rewind Start	(Day to Night)
	006-1628	Eclipse Exit, SAD Rewind End	(Night to Day)
	006-1632	PPT control DISabled	Nominal
Monday	007-1403	PPT control Enabled	(Yaw Steering)
Jan 7	007-1428	ALI Image – Chile	PPT DCE #1
	007-1550	ALI Image - Florida	PPT DCE #2
	007-1726	ALI Image - Colorado (blind)	PPT DCE #3
	007-1801	PPT control DISabled	Nominal
	007-1802	PPT power OFF	Nominal

3.2.2 Second PPT Test Period, March 14, 2002

During the second test period of operations with the PPT, the pitch axis of the spacecraft was under PPT control for a total of 10 hours over a 12-hour period. Four science-instrument DCEs were performed with the ALI as well as additional ALI calibration events with PPT firings. As part of the planned operations of the day, the attitude scenario included yaw steering, reaction wheel speed biasing and solar array stop/start activities in order to investigate the limits of the PPT control authority. During this testing, the following activities were planned and executed to further characterize the EO-1 PPT:

- Long term operation to evaluate thermal performance during steady-state conditions
- Reaction wheel speed biasing, for the roll and yaw Reaction Wheel Assemblies (RWAs), as part of the science DCEs
- Solar array Ramp to zero and return to Track mode as part of the science DCE
- PPT maximum charge operation during an ALI Dark Calibration event, with the ALI cover in both the OPEN and CLOSED states
- PPT operation with a minimum discharge level of both two ACE cycles (80 msec) and three ACE cycles (120 msec) instead of the launch value of four ACE cycles (160 msec).

Table 5 provides a chronological summary of the PPT events on March 14, 2002 (2002-073). During PPT control, the Failure Detection and Correction (FDC) subroutine #25 automatically switched the spacecraft from PPT to RWA pitch control on two occasions. This FDC routine prevents the PPT from being continuously fired at the maximum energy level for longer than a specified number of times (100 cycles in this case). The first FDC #25 trip occurred at UTC 14:22 and was caused by the response of the PPT to the solar array movement after the DCE. The transition of the array from a parked mode back into sun tracking mode was not modeled in simulations. If necessary it would be possible to allow for continuous PPT control by increasing the allowable number of continuous fires at maximum charge during this event or changing the acceleration profile of the solar array as it transitions back to a sun-tracking mode.

The second FDC #25 trip occurred at UTC 17:46 and was caused by excessive pitch momentum build-up due to cross coupling from roll and yaw wheel bias operation mode. A more extensive description of this event and analysis of its cause can be found in Reference 8. By resetting the wheel-biasing speed to zero, the spacecraft was prevented from repeating this event during the rest of PPT operations. This second FDC trip is not considered to have any significant implications for implementing PPTs on future spacecraft because the trip was an artifact of controlling one axis with the PPT and the other two with wheels and torquer bars.

Table 5. PPT Test Period #2 Activity Sequence

UTC	Activity
12:06	Start Yaw Steering
12:40	ALI Dark Cal
12:55	PPT Power ON
12:56	PPT Control Enable
14:05	SA Ramp prior to DCE
14:12	ALI Cover OPEN
14:14	ALI DCE (Argentina)
14:17	FDC #25 Trip due to SA Track and Wheel Bias -> back to Pitch RWA control
14:22	ACS FDC Reset #25
14:26	Disable RTS #87 to prevent SA Ramp on subsequent DCEs
14:28	Zero out RWA Bias values
14:49	PPT Power ON
14:50	PPT Control Enable
15:18	Load ACS Table 93 to change PPT Min. Cycles to a value of 2
15:36	ALI DCE (East Coast)
15:44	RWA Bias change
16:27	ALI Dark Cal (Cover Closed) with 10 PPT SP1 Override Firings at Max Charge
16:33	ALI Dark Cal (Cover Open) with 10 PPT SP1 Override Firings at Max Charge
17:13	ALI DCE (Great Plains)
17:20	RWA Bias change
17:46	ACS FDC #25 Trip due to Wheel Bias operation -> back to Pitch RWA control
17:52	Zero out RWA Bias values
18:09	PPT Power ON
18:10	PPT Control Enable
18:49	ALI DCE (Seattle, West Coast)
18:50	Zero out RWA Bias values via ATS
19:07	Load ACS Table 93 to change PPT Min. Cycles to a value of 3
23:38	PPT Control Disabled -> back to Pitch RWA control
23:39	PPT Power Off

3.2.3 Third PPT Test Period, April 25, 2002

During the third test period of operations with the PPT, the pitch axis of the spacecraft was under continuous PPT control for over 9 hours. During this period, five science-instrument DCEs were performed with the Atmospheric Corrector (AC). Prior to the start of PPT operations, a flight software patch was uploaded to the spacecraft to allow zero-charge time commands to be processed when the calculated pitch control torque was below the minimum charge level of the PPT. The minimum PPT charge time remained set at 3 ACE cycles (120 msec).

3.2.4 Attitude Control Results

The primary purpose of the PPT attitude control experiment was to demonstrate the capability of the PPT to perform the function of a precision attitude control subsystem during a science imaging mode and to compare the performance of the PPT to the baseline reaction wheel design. This section describes the significant attitude control results for each PPT test period.

3.2.4.1 Initial Test Period

For the initial PPT testing sequence, all closed-loop PPT control operations showed that the PPT performed as expected and was able to maintain acceptable control throughout all the events of the spacecraft orbit. The period of greatest interest was the DCE sequence.

During the DCE sequence, there are transient pitch-axis torques generated by the opening and closing of the ALI cover. (As noted above, this was not modeled in the simulations.) Each of the following data plot sets has been chosen such that the ALI cover opening occurs near 300 seconds into the sequence. Figure 7 has been constructed by overlaying the Pitch attitude error from all four DCE data sets. The peak attitude error for the ALI cover open disturbance is 270 arcsec under reaction wheel control and 310 arcsec under PPT control. It is difficult to compare the attitude error for the ALI cover closed disturbance because, in the case under reaction wheel control, the ALI cover is closed at the same time the reaction wheel biasing is zeroed and the solar array begins tracking again.

Figure 8 provides the pitch rate error comparison for all four DCEs, and Figure 9 provides the PPT-commanded ACE charge cycles for the first DCE on January 7. Note that the control torque required to counteract the ALI cover opening causes the full, saturated value of the PPT (23 ACE cycles) to be commanded for about 15 seconds. Since the EO-1 installation of the PPT does not have as much torque authority as the reaction wheel, the peak attitude error in Figure 7 is slightly higher for the PPT cases as compared to the RWA case. However, since the ALI cover opening occurs about three minutes prior to the actual science data acquisition, the critical pointing performance occurs during the period referenced by 480-540 seconds on the plots. During this interval, the RWA-based error is 10-20 arcseconds and the three PPT-based errors are 0 to 40 arcseconds. Two of the three PPT cases demonstrate better performance than the RWA case during this interval. The attitude control error transient response for all cases is largely determined by the characteristics of the PID controller parameters. Note that the pitch integral control gain for PPT is about one-fourth the RWA value.

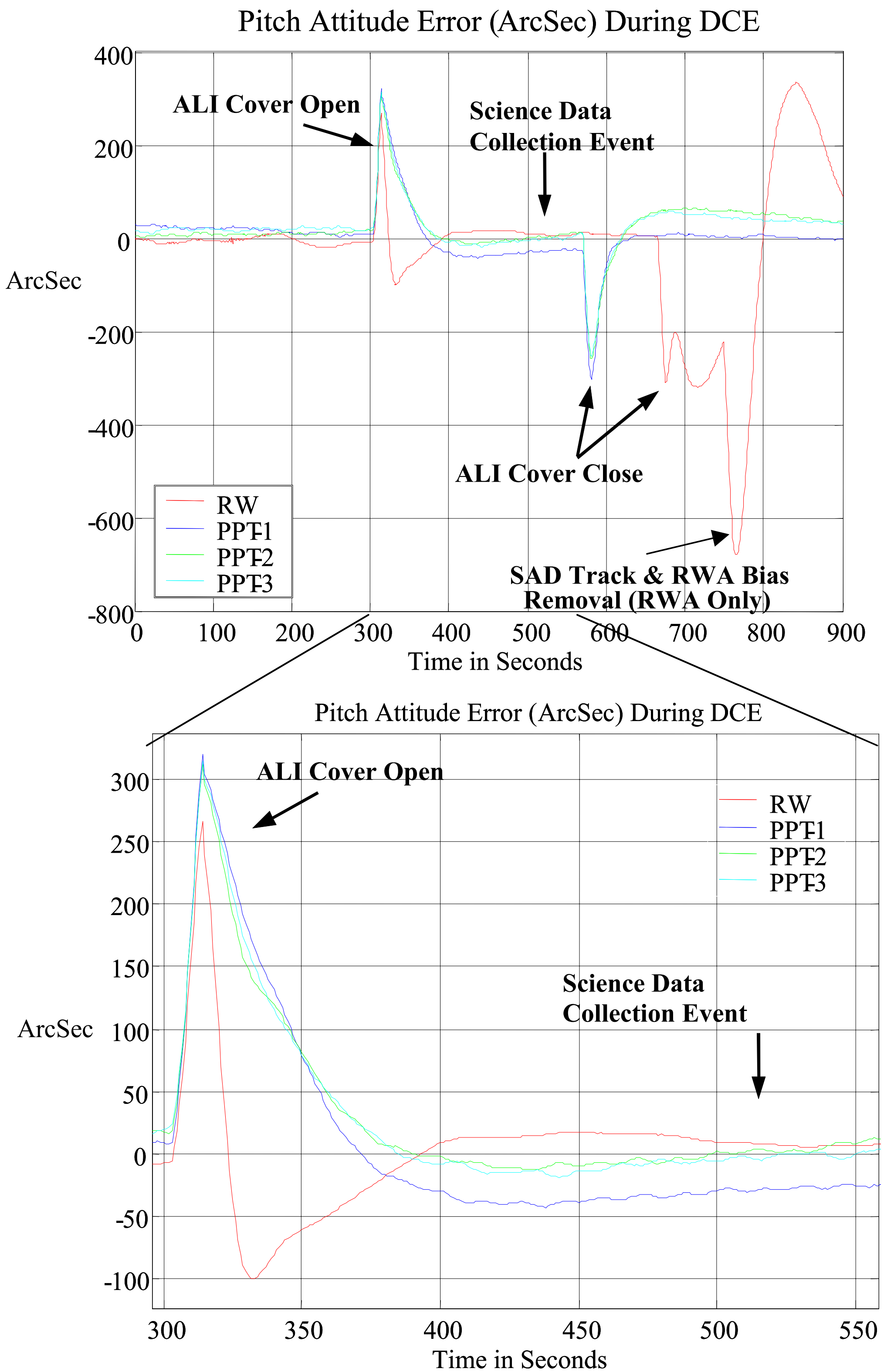

Figure 7. Pitch Attitude Error Comparison

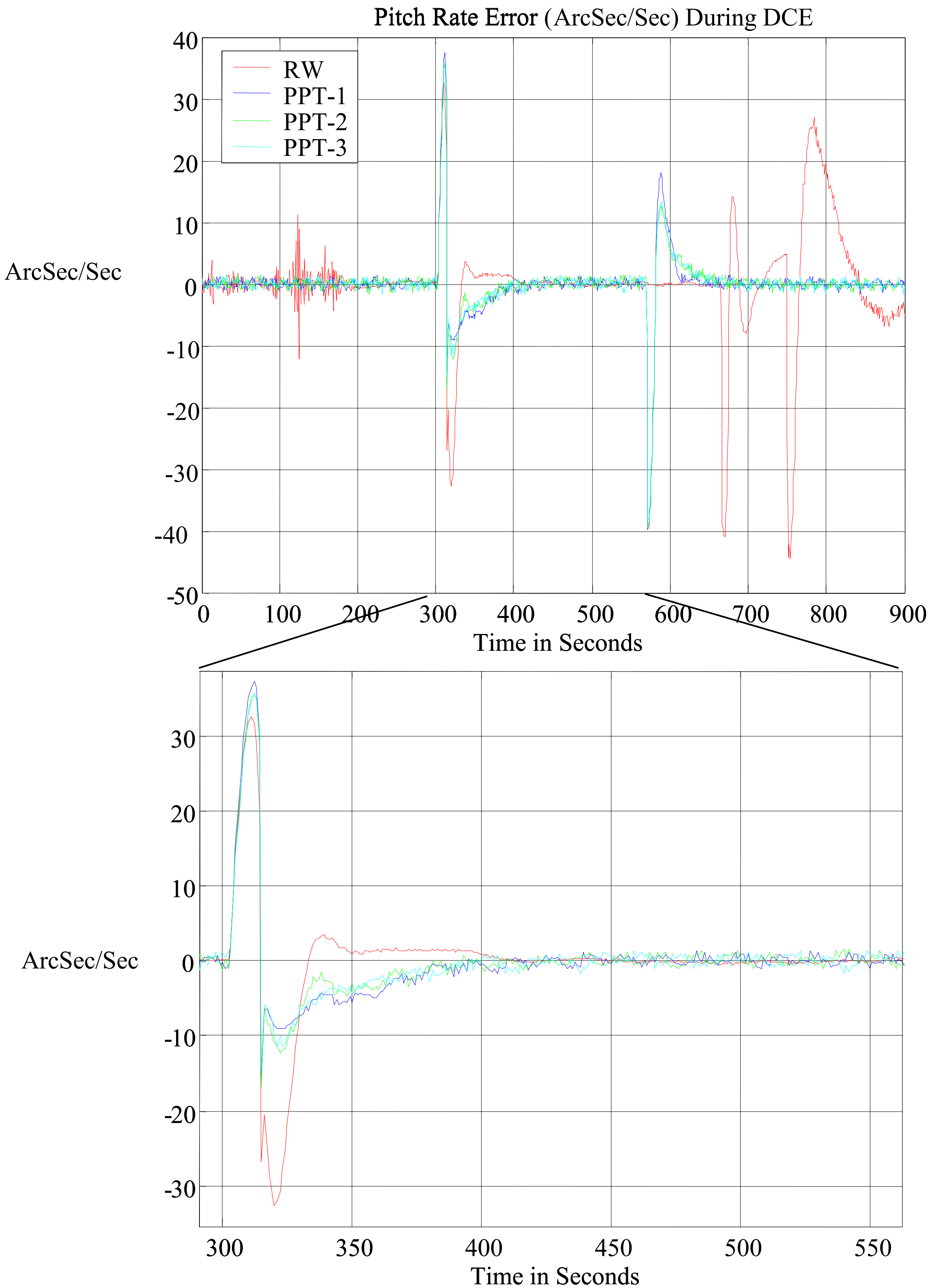

Figure 8. Pitch Rate Error Comparison

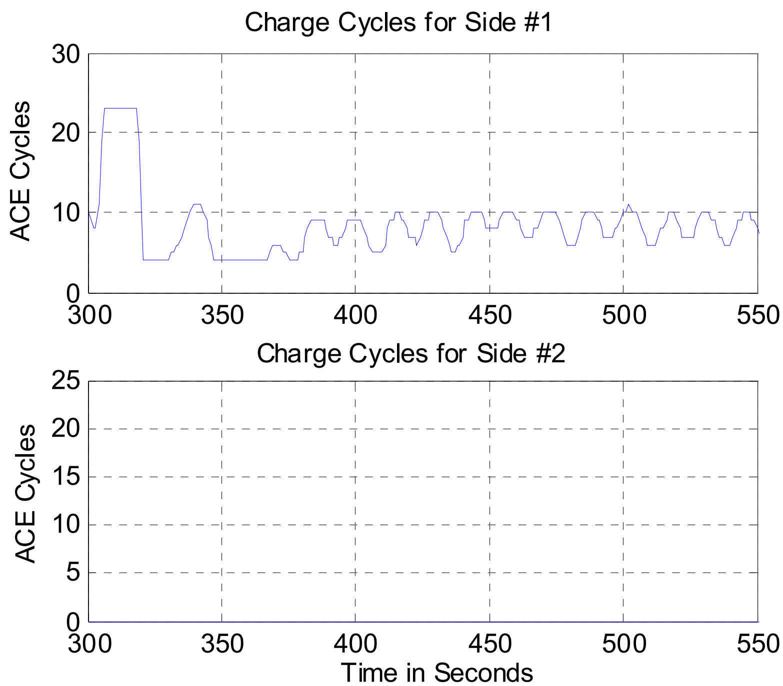

Figure 9. Command PPT ACE Charge Cycles

From these comparisons, it can be seen that the PPT pointing performance exhibited by the EO-1 spacecraft has been demonstrated to be comparable to that provided by the pitch reaction wheel. While the PPT has a lower torque authority, the lack of internal momentum accumulation and the precision impulse capability enable excellent pointing performance.

Figure 10 shows a comparison between PPT and reaction pitch wheel control for attitude error parameters for all three axes. The data shown covers one orbital period and is centered near when the DCE occurred. For these plots the attitude error, given in arcsec, is the controller position error between commanded attitude and estimated attitude. The rate error, given in arcsec/sec, is the controller derivative error between commanded angular rate and measured rate. The integral error, given in arcsec*sec is the controller integrated position error. It should be noted in comparing the PPT performance to the wheel performance that the vertical axis scales might be different. It can be seen, that for all three axes, that the PPT control mode compares favorably with the wheel control mode.

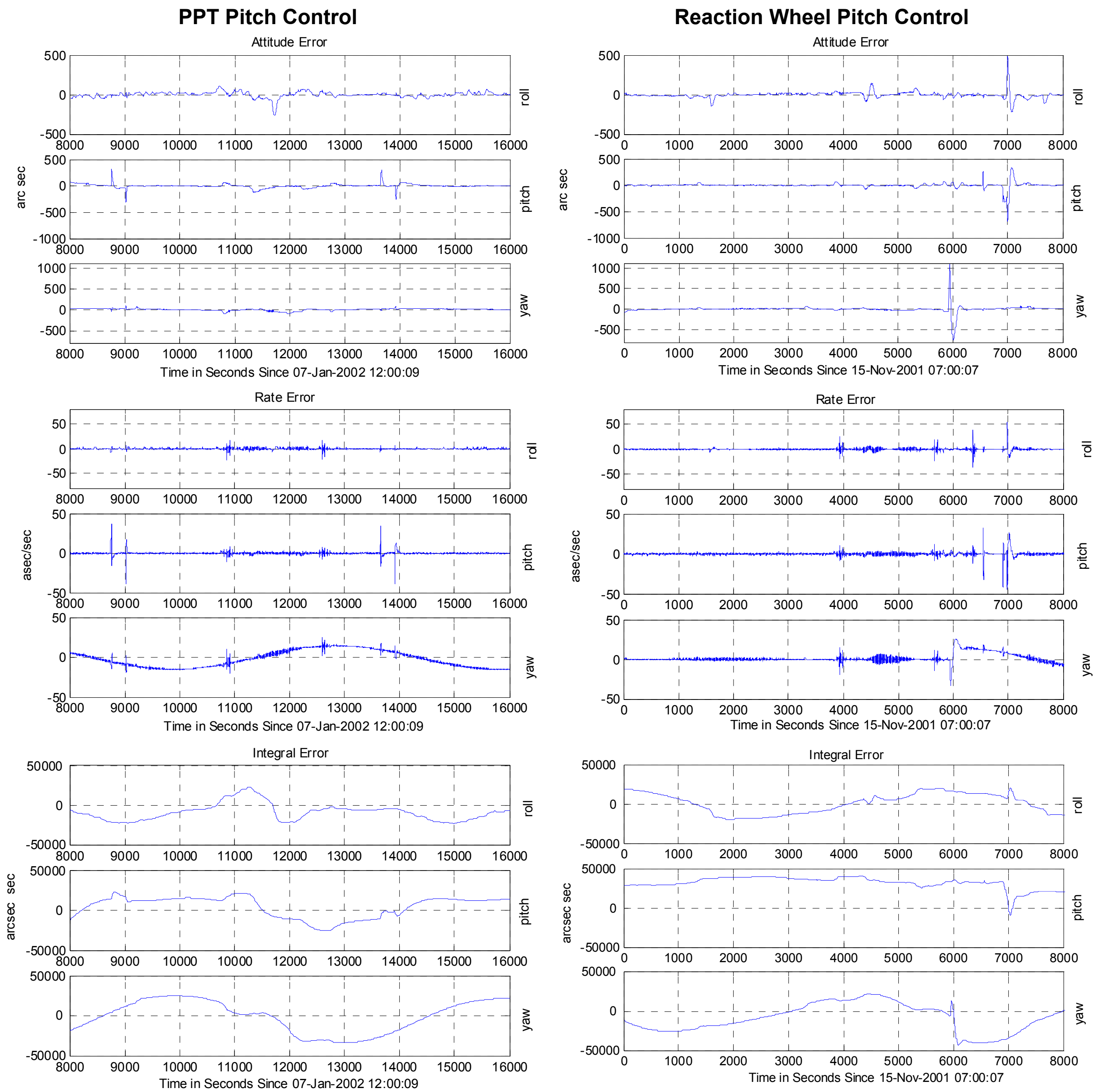

Figure 10. Comparison of PPT and Pitch Wheel Attitude Control Error Parameters

3.2.4.2 Second PPT Test Period, March 14, 2002

The second PPT tests enabled evaluation of PPT pointing performance over a longer duration as well as the evaluation of the effects of changing the PPT minimum charge cycle. As mentioned above, tests to evaluate the capability of the PPT to control step-change in torques associated with the returning the solar array to TRACK mode after a DCE showed that these torques were beyond the PPT control authority. Additionally, disturbances caused by the RWA speed biasing also caused saturation of the PPT and triggered FDC as a conservative response.

Figures 11–13 show attitude control parameters for the entire PPT test period that covered approximately nine orbits. Time zero corresponds to 08:00 UTC. To compare performance of the PPT with respect to pitch wheel performance it is useful to look at parameters from time 0 to 1.8 x 10^4 sec during which the spacecraft is in an all-wheel control mode and those same parameters from time 3.7x10^4 to 5.6x10^4 sec during which the spacecraft is in uninterrupted PPT control. From Figures 11 and 12 it can be seen that the attitude errors over the entire orbit for PPT control are very comparable to the attitude errors for all-wheel control and match the predicted PPT performance. Figure 13 shows that the command charge cycles for each spark plug also closely match simulation predictions as shown in Figure 6. Figure 14 is a plot of commanded PPT firings for both sides as a function of orbit angle. Again it is seen that the PPT charge commands match predictions and are reasonably repeatable from orbit to orbit.

Tests were performed to determine the minimum charge level at which the PPT would reliably discharge. Based on ground tests, the default minimum charge length had been set at 4 ACE cycles (160 msec). This minimum level was decreased to 3 and then to 2 ACE charge cycles. Analysis of ACE capacitor voltage telemetry for the 3 ACE charge cycle minimum level indicated that the PPT successfully discharged each time it was commanded to charge at this level. Analysis of the data taken for the 2 ACE charge cycle tests showed that the PPT would occasionally not discharge when only charged for 2 cycles (consistent with vendor expectations). During part of the period when the minimum charge time was set at 2 ACE cycles, it was determined that out of 211 times the PPT was commanded to fire at its minimum level, the PPT failed to discharge 123 times. Based on this testing, the minimum charge level of the PPT was reset from 4 to 3 ACE charge cycles for all additional PPT testing. Plots of the pitch attitude control performance for PPT operation with the minimum charge cycle set to 3 ACE cycles are shown in Figure 15.

Note that analysis of this data identified that the ACS flight software required a code patch to correct a command logic implementation error. The baseline logic did not implement a dead zone wherein the required thrust must be above the minimum value or else a zero value is commanded. Instead, the minimum charge cycle was always issued when the desired thrust is equal to or less than the minimum value. This code error was corrected for the third PPT test period.

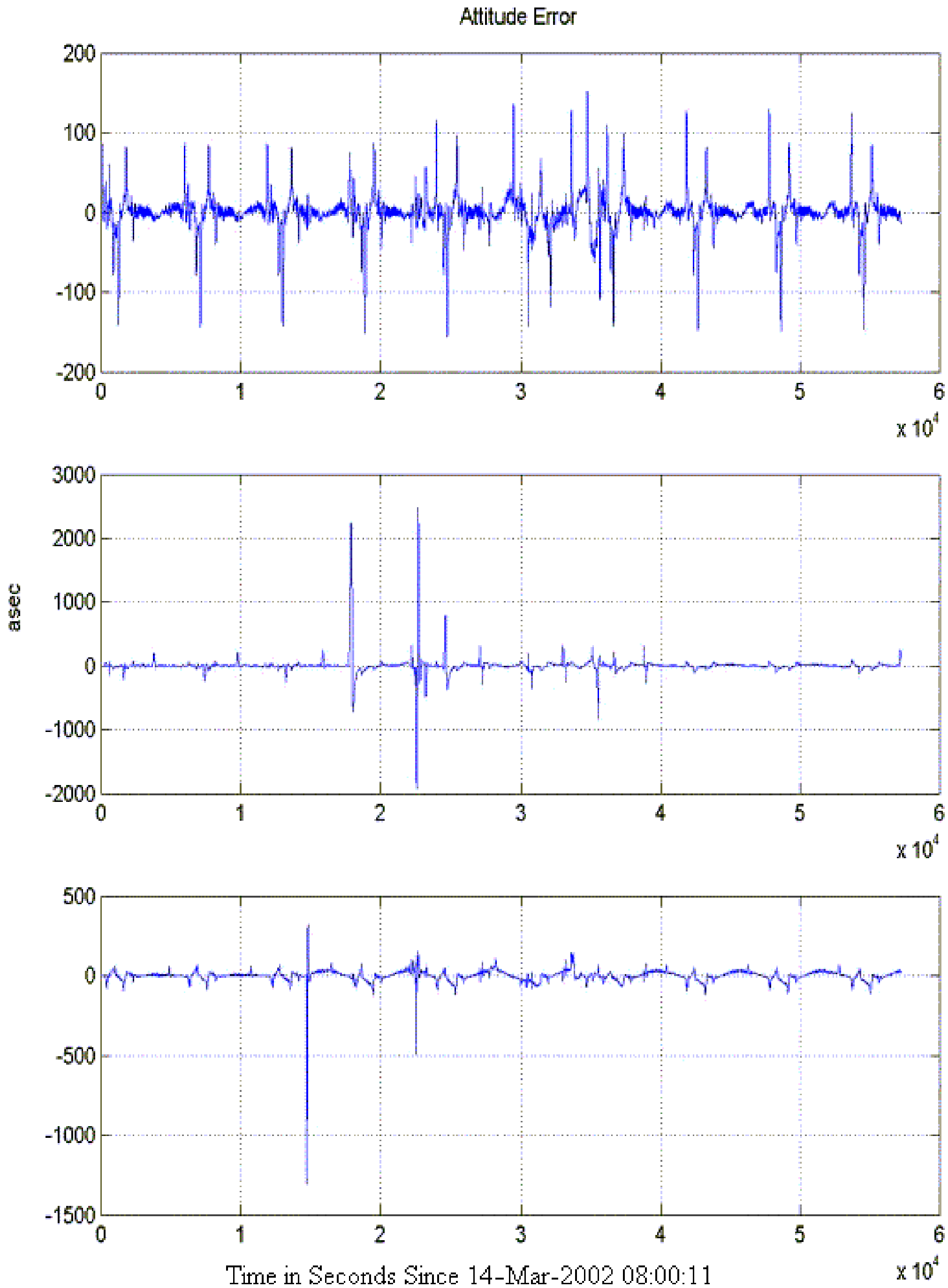

Figure 11. Test Period 2 Attitude Errors – Full Scale

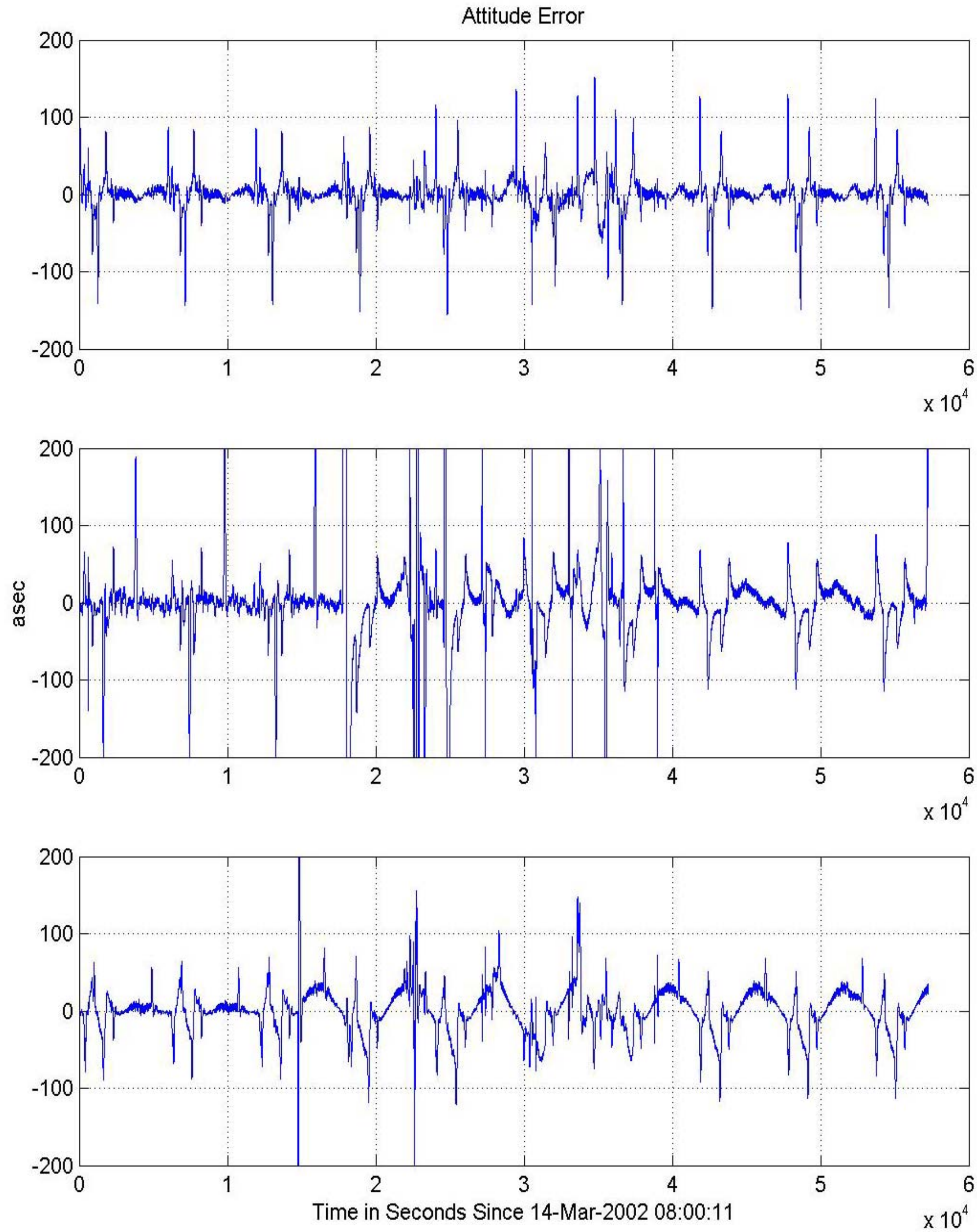

Figure 12. Test Period 2 Attitude Errors – 200 Arcsec Scale

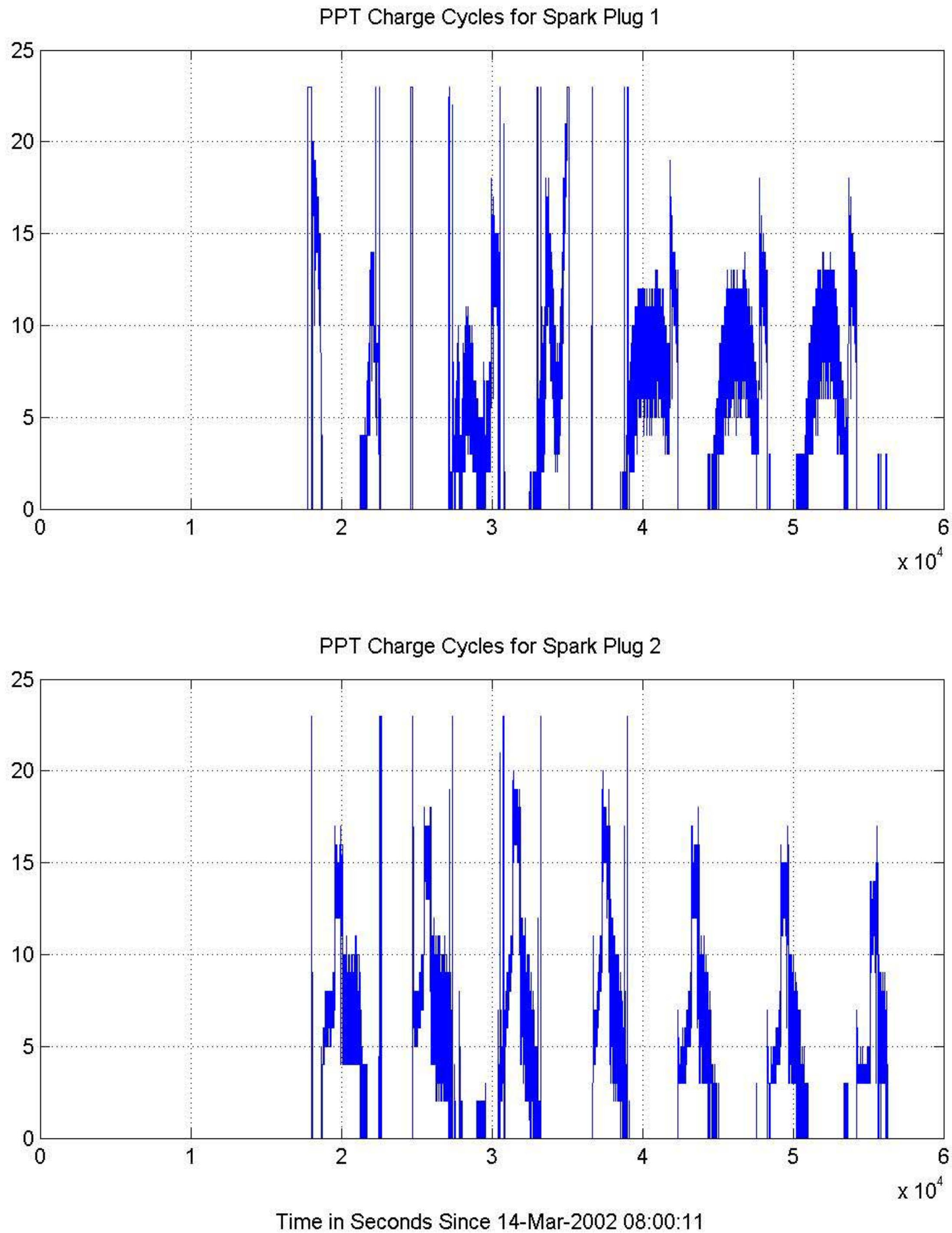

Figure 13. PPT Commanded ACE Charge Cycles (23 Cycles = Saturation)

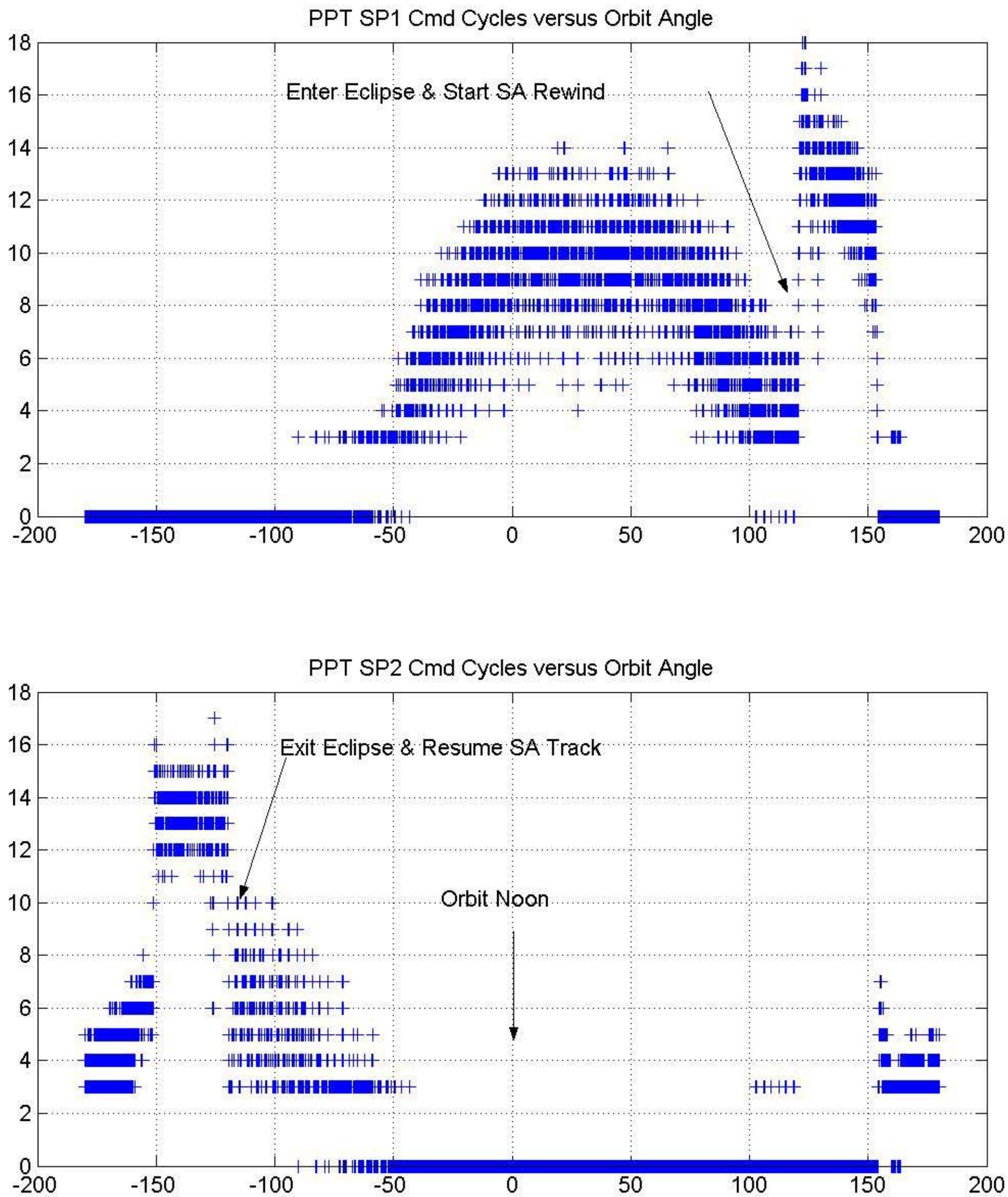

Figure 14. PPT ACE Charge Commands Versus Orbit Angle

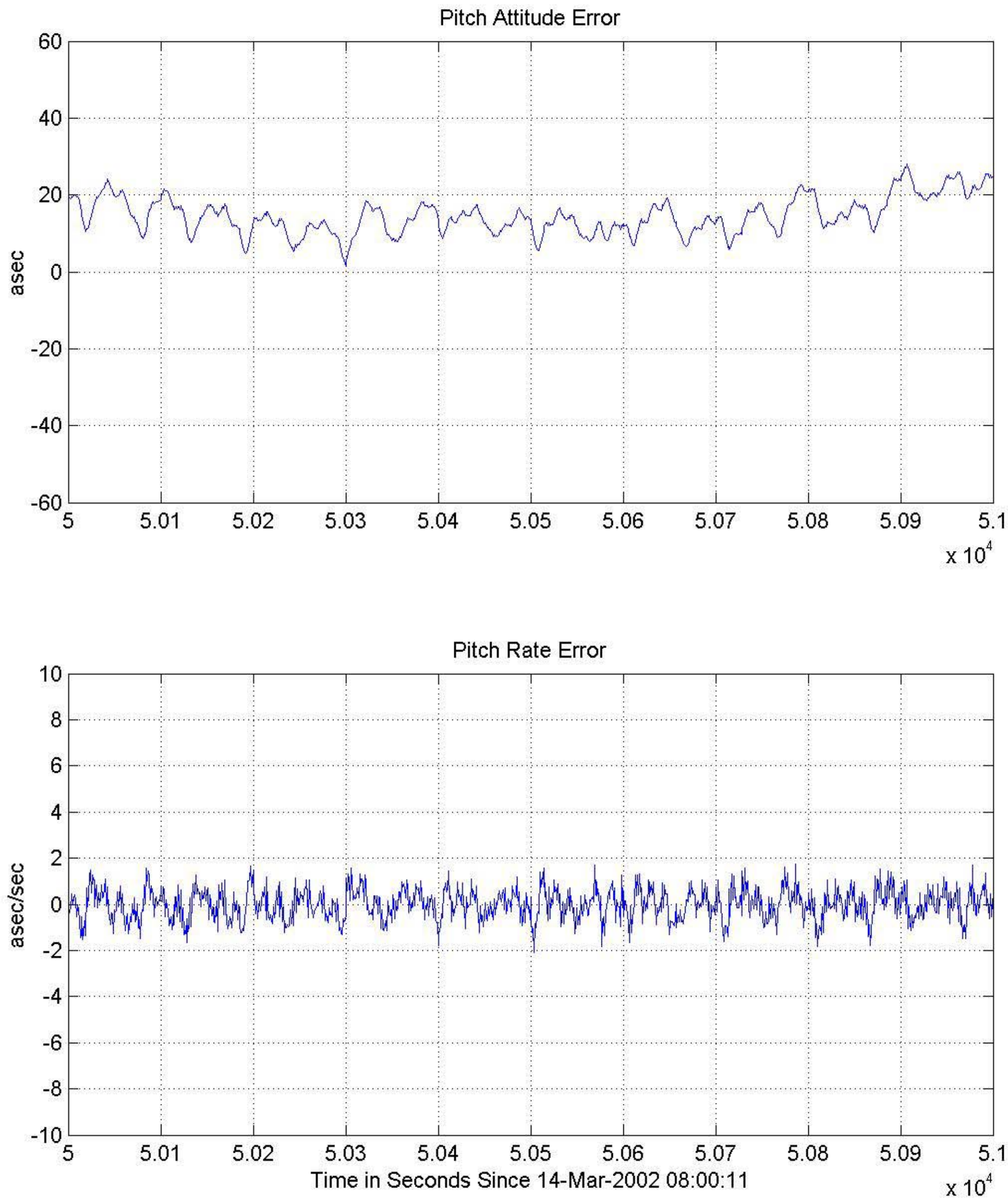

Figure 15. PPT Control Errors With Min Cycles = 3

3.2.4.3 Third PPT Test Period

Preliminary results from the third test period indicate nominal PPT attitude control performance for the entire test period. The software patch that allows for PPT charge commands to be set to zero appeared to function as expected. The spacecraft remained in PPT control mode for the entire period without experiencing any FDC trips. Analysis of the control performance is currently on going.

3.2.5 Spacecraft Interaction

All spacecraft subsystems and the ALI instrument operated nominally during and after all PPT operations. The PPT operated for more than 23 hours, accumulating over 84,000 pulses from the three PPT test periods. During PPT operation, there were no processor or other error flags generated on the spacecraft that could be linked to PPT operations. All telemetry and telemetry links appeared nominal. There was no evidence of EMI or plume effects on other subsystems or instruments.

3.2.5.1 ALI Instrument

As indicated in the timeline in Table 4, the PPT was operated incrementally as the ALI went through a verification/calibration sequence between each step. ALI's standard lamp calibration tests were used to verify the status and health of the instruments. All the lamp calibrations looked nominal and an analysis of the noise background showed no discernable difference between that seen with and without PPT operation.

<u>Images</u>

All ALI ground images looked nominal and showed no sign of distortion or blurring due to PPT operations. An example of part of one of these images is shown in Figure 16. Because of timing and pointing constraints during the PPT testing it was impossible to image a location imaged previously with 3-axis wheel control for a direct comparison. An attempt to achieve comparable images will be made during future PPT operations.

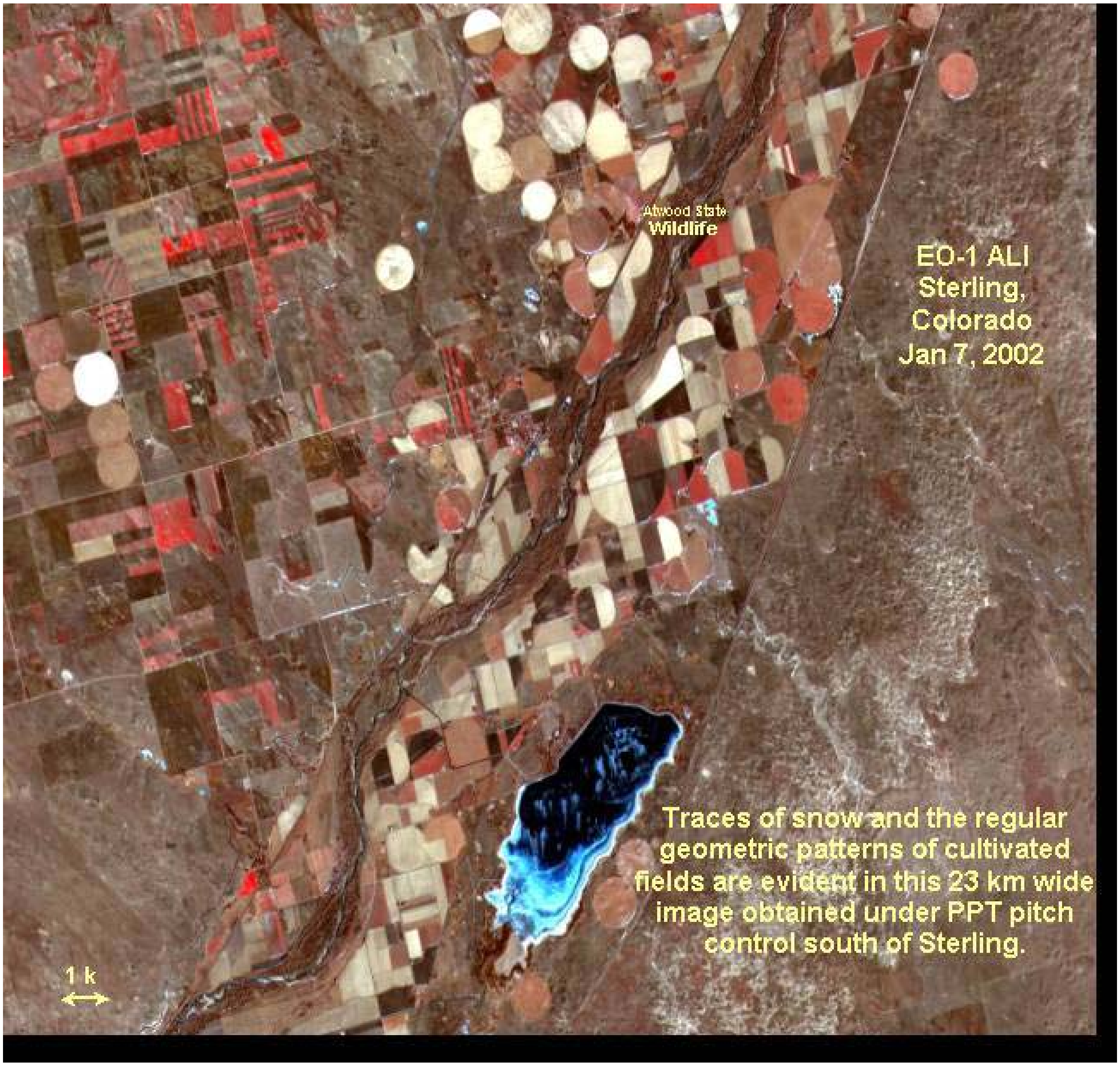

Figure 18. ALI Image Taken Under PPT Control

During the second PPT test period, a sequence of ALI DCEs were taken during orbit night over the Pacific Ocean to quantify the effect of the PPT operation on ALI noise and to search for stray light effects that might be caused by the PPT plume. Three 30-second images where taken, the first with the ALI cover closed and the PPT not firing, the second with the ALI cover closed and the PPT firing, and the third with the ALI cover open and the PPT firing. The two images with the PPT firing where centered on 10 seconds worth of override commands that forced PPT side one to fire at maximum charge.

The noise for each band of the ALI was calculated over the 30-second period and the image taken with the PPT not firing was used as a baseline. Figure 17 shows plots of the ALI noise for both the baseline case and the case of the PPT firing with the cover open. All dark data indicates that the level of ALI noise is not affected by PPT firing. To search for stray light effects the mean subtracted dark current images were plotted highlighting pixels with values greater than 5 times the noise. For these plots PPT stray light effects would appear as horizontal lines. Shown in Figure 18 are plots for each ALI band with the PPT firing with the cover open. All dark image data demonstrates that stray light is not introduced by PPT firing.

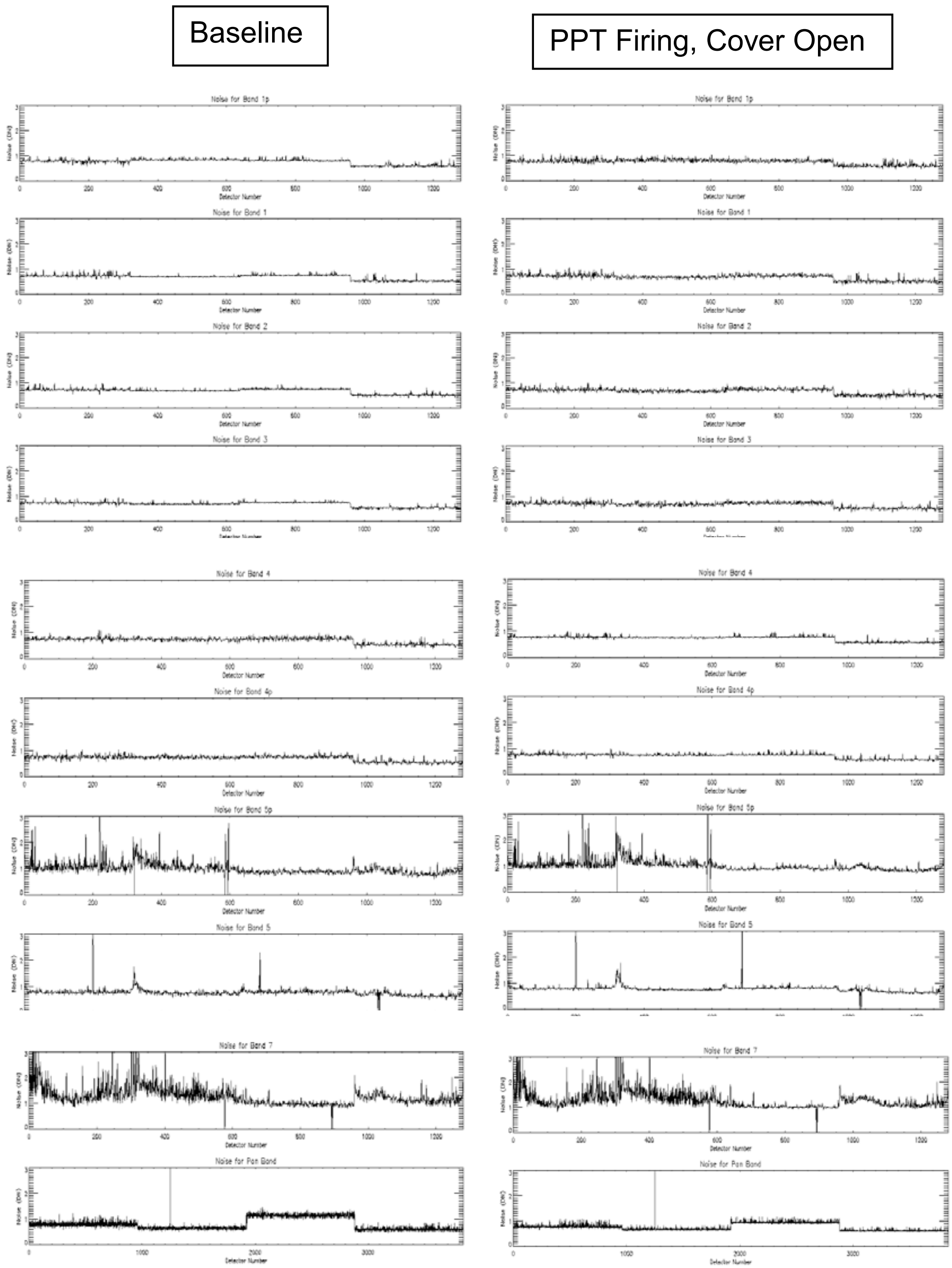

Figure 17. ALI Noise for Each Band

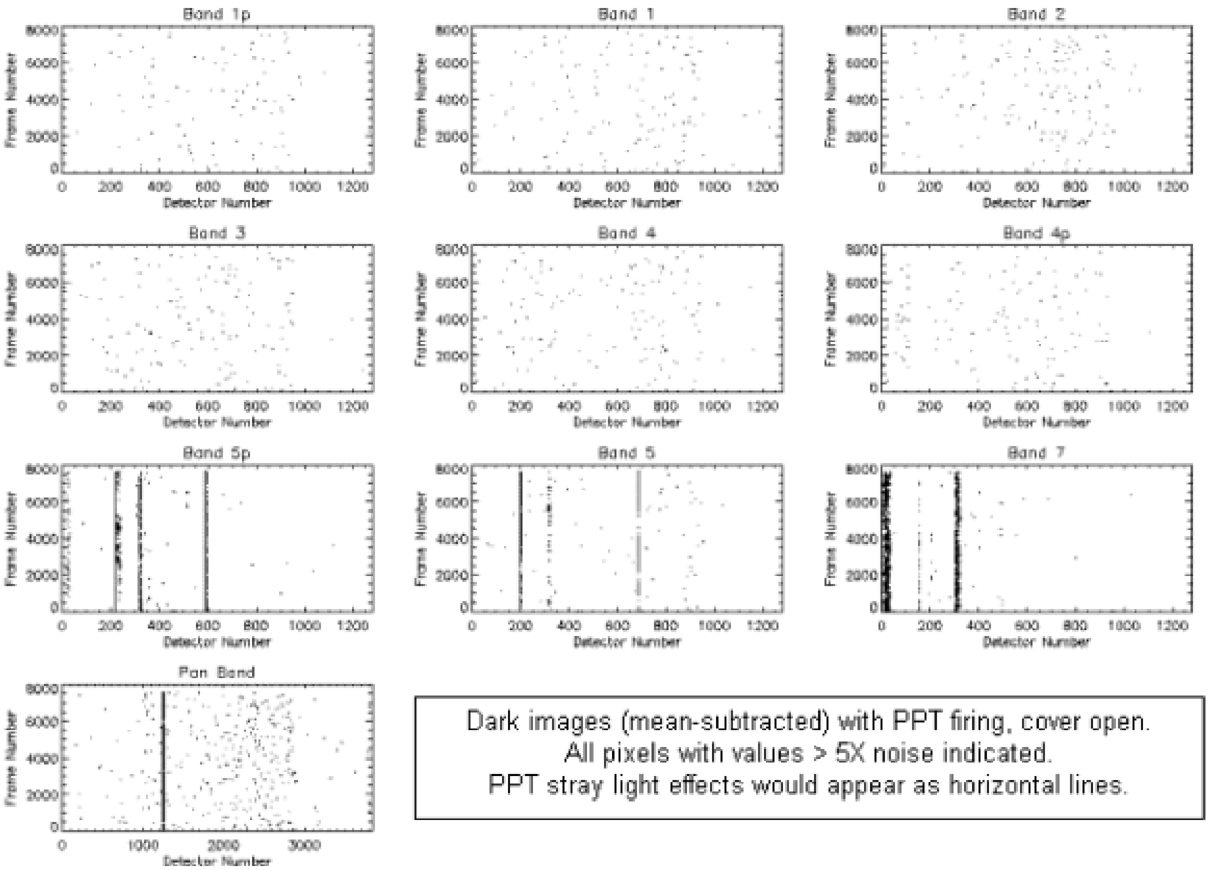

Figure 18. ALI Mean-Subtracted Dark Images

3.2.5.2 Hyperion

As mentioned above, during all PPT operations to date, the Hyperion instrument was powered off and in a "warm" outgas mode. At the end of the first PPT validation sequence, the Hyperion was powered on. All Hyperion data looked nominal and there were no indications of any harm to the instrument from PPT operations. Hyperion operation after all additional PPT testing also confirmed that the instrument operated nominally and was not affected by PPT operation. Plans exist for imaging with the Hyperion during PPT control in a future testing.

3.2.5.3 Atmospheric Corrector

During the third PPT test period five AC images were collected. This data is currently in the process of being analyzed. Currently there have been no indications of any AC anomalies due to PPT operation.

3.2.6 PPT Performance

In addition to the spacecraft's attitude response to PPT actuation, PPT voltage, temperature, and fuel bar readings were used to evaluate PPT performance. These parameters are discussed in detail below.

Overall, during PPT operations, these parameters indicated nominal PPT performance. There were only two exceptions.

First, the initial attempt to fire side 2 of the PPT, at its minimum design energy level of 4 ACE charge cycles (160 msec), failed. This was anticipated. Experience with PPTs on the ground indicated that a layer of oxidation or other form of contamination may build up on sparkplugs if they are not used for an extended period of time, thus preventing discharge at lower energy levels. This layer is quickly “burned” off after firing at higher discharge energies. Side 2 was able to consistently discharge at a minimum charge cycle of 4 after it was fired once at its maximum energy level (23 charge cycles). As mentioned above, experiments during the second PPT test period showed that the PPT was able to be fired reliably at a minimum charge level of 3 ACE cycles (120 msec charge time) but that as expected, the PPT cannot fire reliably at 2 ACE cycles (80 msec charge time).

The second exception to nominal performance was higher than anticipated PPT temperature readings. The maximum temperatures were approximately 10 degrees higher than anticipated. Because of the limited time for continuous operation during the first PPT test period, it was unclear whether the PPT reached a steady-state temperature or if the temperature could gradually rise above the maximum operating temperature. Sufficient operating time, which was obtained during the second and third PPT operating period, demonstrated that the PPT could be operated continuously without concern for exceeding temperature limits.

3.2.6.1 Main Capacitor Voltages

The on-orbit main capacitor charge and discharge voltage profile looked identical to the charge and discharge characteristic measured on the ground. Figure 19 shows the main capacitor voltage (using the 10 sample/sec ACE dwell data) for 10 consecutive pulses on side one at maximum charge level. Because of the points at which the voltage is sampled, the maximum voltage was not captured with the 10 samples/sec data. However, the peak voltage for each cycle can be seen in Figure 20, which shows the capacitor voltage before discharge for the same period. (The difference in absolute times is a function of time tagging of different data telemetry packages.) The peak telemetry voltage of 4.2 V corresponds to capacitor energy of approximately 58 J, which is also show in Figure 20.

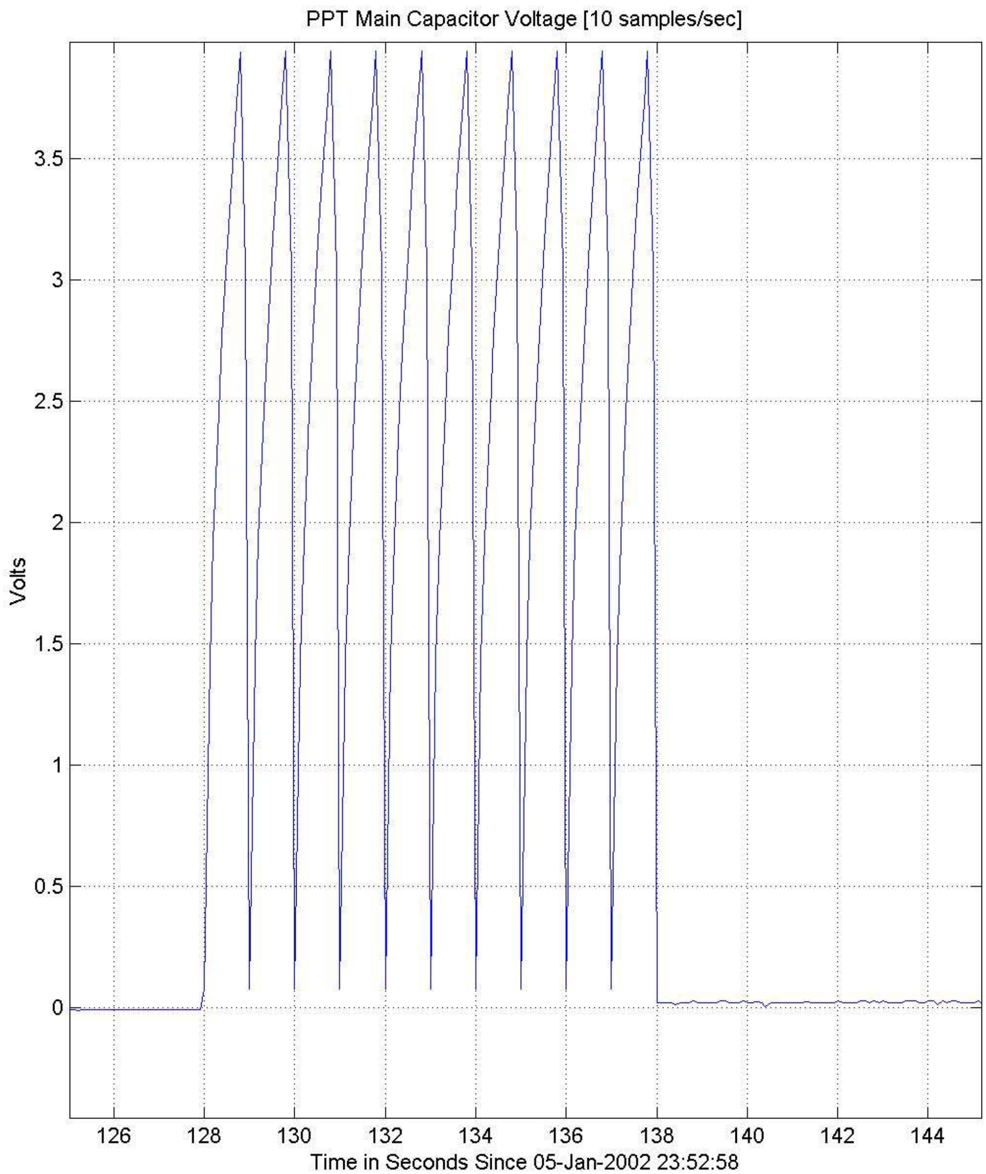

Figure 19. Main Capacitor Voltage Charge Profile

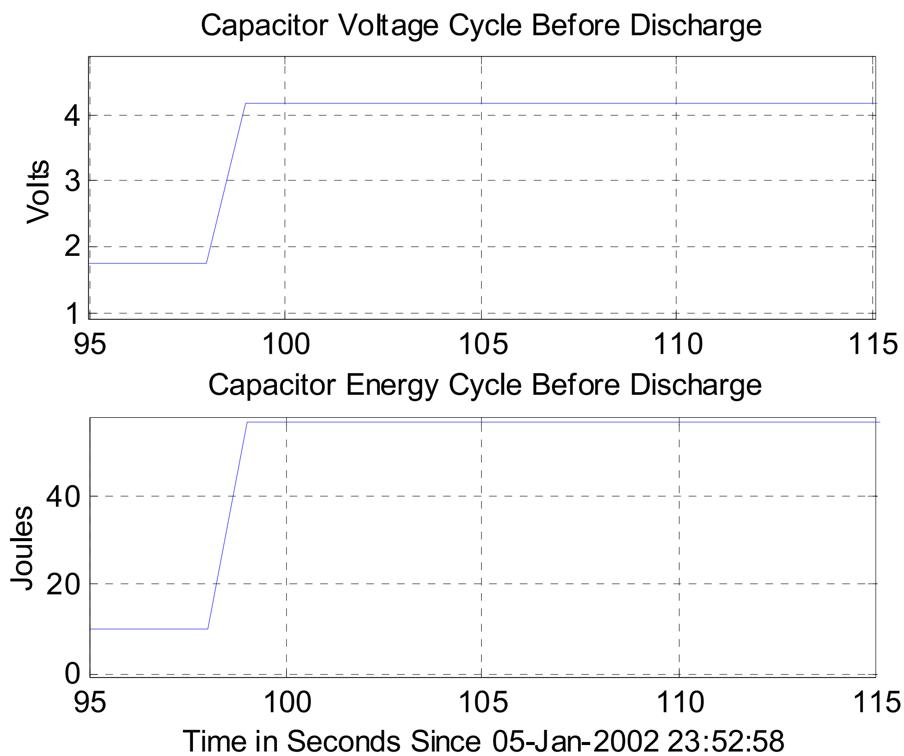

Figure 20. Capacitor Energy and Voltage Before Discharge

3.2.6.2 Spark Plug Capacitor Voltages

The voltage on both spark plugs is captured at two different ACE time cycles, the cycle before discharge and the last cycle, cycle 25. Figures 21 and 22 show these data points plotted for the same 10 pulses on side 1 at maximum energy as shown above for the main capacitor voltage. Figure 21 shows both spark plug capacitors being charged to the same voltage, as expected, prior to PPT firing. In Figure 22 it is seen that the side that was discharged, side 1, has a much lower voltage, indicating a good discharge, while side 2's voltage has decreased only slightly after the firing event, indicating that this side did not discharge but is only slowly bleeding down.

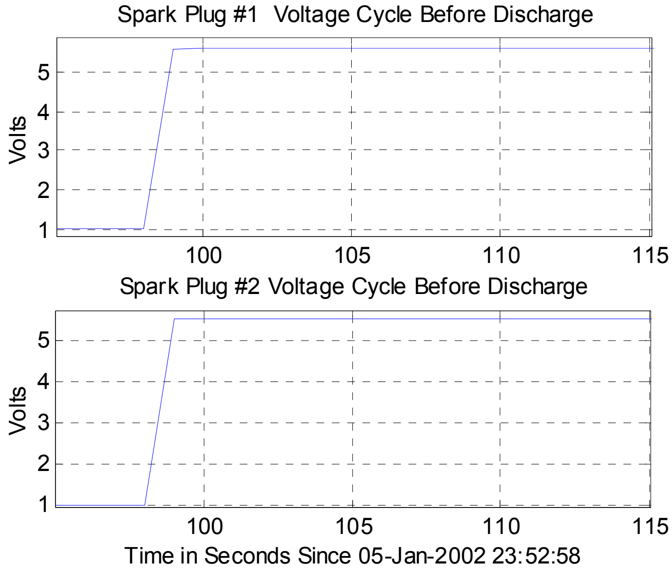

Figure 21. Spark Plug Voltage Before Discharge

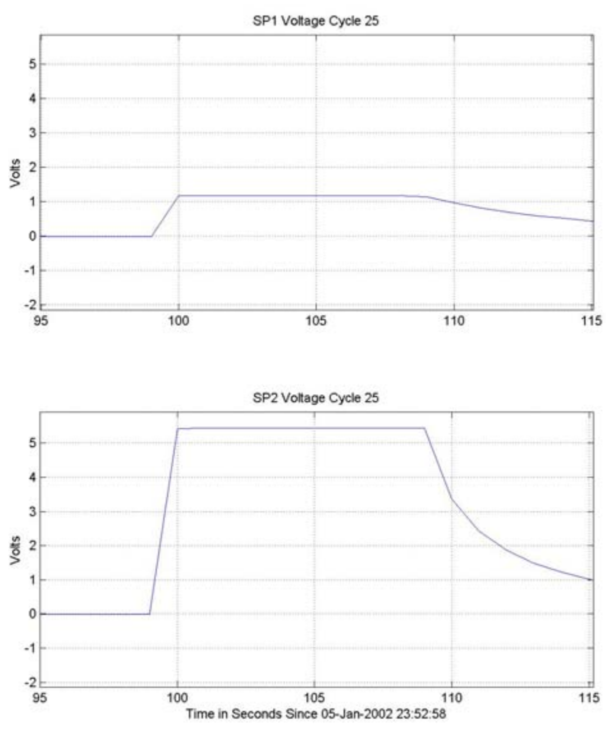

Figure 22. Spark Plug at ACE Cycle 25

3.2.6.3 Temperatures

As mentioned above the PPT temperatures during PPT operation exceeded their expected values by as much 10^{o}C. Based on temperatures reached during protoflight-level testing of the EO-1 PPT, the maximum temperature limit of the PPT was set at 54^{o}C for the main capacitor and at 110^{o}C for the transformer electronics. During the first PPT test period, the PPT was only operated for a maximum of 4 hours continuously and the temperatures did not appear to reach equilibrium. During the second PPT test period, a maximum of 5.5 hours continuous PPT operation time was achieved and it appears that thermal equilibrium was achieved. Plots in Figure 23 show both the transformer and main capacitor temperatures. While the main capacitor temperature does get close to the 54°C limit, the orbit trend indicates that peak temperature would exceed this limit if PPT operation would continue indefinitely. Results from the third PPT test period are very similar to those of the second test period. The PPT temperatures approach 54°C but do not exceed this limit as a steady-state profile is achieved. An investigation is now underway to determine the difference between on-orbit and predicted results.

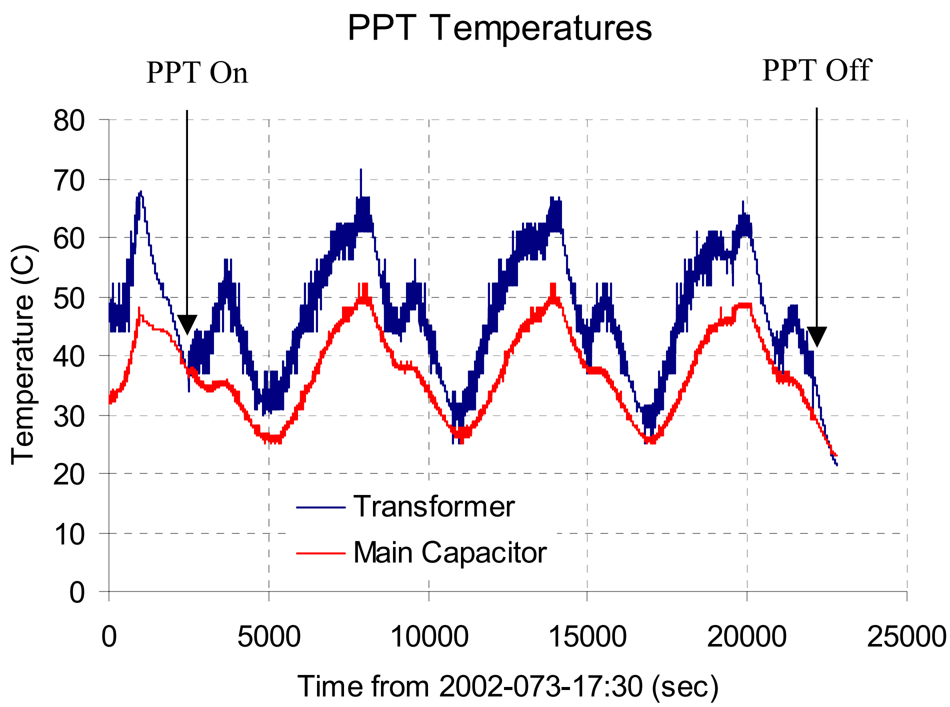

Figure 23. PPT Temperatures for Test Period 2

3.2.6.4 Fuel Bar Readings

PPT fuel consumptions based on PPT commanded charge times is estimated at 1.6 grams for side 1 and 1.1 grams for side 2. This corresponds to over 47,000 pulses on side 1 and 37,000 pulses on side 2. The fuel consumptions estimates are based on extrapolated specific impulse ground test data as a function of capacitor charge level.

Fuel bar readings from the cumulative on-orbit firing time of the PPT were not sufficient to obtain reliable fuel consumption measurements. The fuel bar telemetry points relate the linear distance the fuel bar travels as it is consumed. The amount of Teflon consumed in each pulse is so miniscule (micro grams) and the resolution of the resistive strip lines used to measure fuel is limited. Noise introduced into the fuel bar measurements from what appears to be thermal variations has made it impossible to discern fuel bar movements from the fuel bar voltage readings. Shown in Figure 24 is a plot of the fuel bar 2 telemetry spanning approximately 8 days around the second PPT test period. Fuel consumption estimates based on commanded charge times predict a 10 milli-Volts (mV) change in this reading between the start and stop of PPT operations. The peak-to-peak variations in the voltage readings, which correspond to the orbit cycle, make it impossible to discern such a difference. Extended PPT operation may enable more reliable fuel bar measurements to be made as the expected voltage difference increases with increased fuel consumption.

It should be noted that during PPT protoflight qualification testing it was determined that the side 1 fuel bar reading was anomalous due to a manufacturing defect in the resistive strip. Because the fuel bar readings are not critical for PPT operation, the PPT was launched with the understanding that the side 1 telemetry could not be relied upon but that it may provide useful readings at some portion of the fuel consumption.

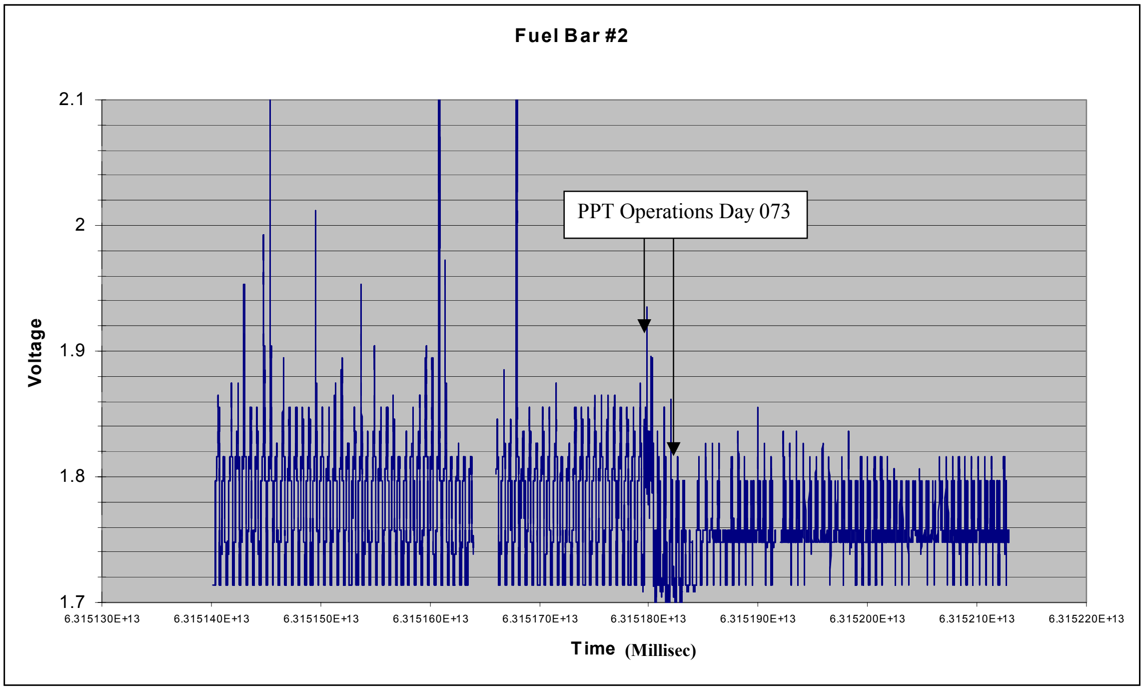

Figure 24. Fuel Bar 2 Readings for PPT Test Period 2

3.3 On-Orbit Usage Experience

The on-orbit use of the PPT has thus far shown that PPT performs as expected and is able to serve as a precision attitude control actuator. The only difficulty with the PPT hardware appears to be the lack of resolution of the PPT fuel bar readings due to thermal transients. Although not a performance-limiting factor, the higher than expected temperatures reveal the need to improve the PPT/spacecraft coupled thermal analysis. On-orbit testing also reveals the capability to decrease the minimum charge time, and therefore minimum impulse bit, from 160 msec to 120 msec. Attempts to calibrate PPT thrust performance on-orbit have been difficult and are still on going. Although the PPT performance has been shown to be adequate for closed-loop control, calibrated thrust levels have not been obtained yet because of the uncertainty associated with other system disturbance torques.

The implementation of the attitude control experiment has gone exceedingly well. PPT control of the pitch axis has been nominal, transitions into and out of PPT mode have been nominal, and there has been no need to adjust control gain parameters. The limitation of the PPT torque authority did not allow the PPT to react within conservative operational constraints to higher than expected torques from solar array tracking step changes and coupled momentum from the wheel-biasing mode. These higher than predicted torques were specific to the EO-1 configuration and could be handled by the PPT through additional changes in the control system design. This experience does indicate that an increase in the range of PPT thrust capabilities may be beneficial to future missions and that there are some nuances with a complex momentum wheel, torquer bar, and PPT control mode that are difficult to model.

4. NEW APPLICATIONS POSSIBILITIES

PPTs can be considered for a wide variety of missions because of their precise impulse bit capability, high specific impulse, and simplicity and ease of operation. General mission categories that may benefit from PPTs include formation flying, precision pointing, disturbance reduction, micro and nano-satallites, and large space structures. Examples of such missions are given in Table 6.

Table 6. Future PPT Applications

Type	Example	Comment
Formation Flying		
Interferometry	Starlight Terrestial Path Finder (TPF) Planet Imager	- Requires 1-cm separation control between spacecraft - PPTs have been leading candidate due to high precision thrust and high Isp
Earth Observing	Techsat 21 Leonardo	- NASA and Air Force are studying ways to deploy constellations of small satellites in co-orbiting formations - Typically requires 1 mN – 100 mN thrust with capability to generate 0.5 mN-s - 2 mN-s impulse bit - PPTs serve well because of small impulse bit, high Isp, and small volume
Precision Pointing		
	Maxim	– Fine attitude control for pointing optical instruments
Continuous disturbance reduction		
Drag free control	GRACE GPS follow-ons	– Repeatable low thrust range of PPT used to cancel atmospheric drag forces – Maintains orbit, improves prediction accuracy
Other	TDRSS-type GEO missions	PPTs can cancel disturbance forces to reduce size of attitude control system
Micro/Nanosats		
	Dawgstar MMS	– Low mass/volume/power ideally suited for microsats – Simple to integrate, No chemical/pressure hazard – Well-suited multiple s/c on a deployer ship and university project
Large Space Structures		
	Antenna platforms	- Used as active control actuators

5. TECHNOLOGY INFUSION OPPORTUNITIES

The EO-1 PPT technology is currently being infused into the Air Force-sponsored University NanoSat Program on the Dawgstar spacecraft being designed and built by the University of Washington. PPTs for Dawgstar were developed by the university with the guidance and assistance of General Dynamics, the EO-1 PPT vendor.[5]

A near-term technology infusion opportunity for the PPT is the Starlight mission. Starlight is the first-ever formation flying optical interferometer consisting of two spacecraft to be launched in 2005/2006. It is part of NASA's Origins theme and is the technology pathfinder for the Terrestrial Planet Finder (TPF) mission. PPTs were selected as a formation flying technology during the initial New Millennium Program phase of this mission.

A far-term technology infusion opportunity is the MAXIM Pathfinder mission. Trade studies performed as recently as May 2002 at GSFC show PPTs to be the leading candidate to provide precision positioning control for the multi-spacecraft MAXIM Pathfinder design. MAXIM Pathfinder is scheduled to be launched in 2015. PPTs would be used on the detector and free flyer spacecraft for precision pointing and reorientation. Their high specific impulse, pulse mode operation, and relatively large range of impulse bit capability make them the leading candidate for this mission.

6. LESSONS LEARNED

The major lessons learned from the EO-1 PPT Flight Validation Experiment included the following:

1) The new generation of Pulse Plasma Thrusters is capable of being used as precision attitude control actuators.
   - PPTs can be implemented as attitude control actuators with minimal impact on existing attitude control subsystem architectures.
2) **EO-1 PPT EMI emissions and plume effects** **DO NOT** affect other spacecraft subsystems or sensitive earth imaging instruments.
   - No impact to ALI image-taking during PPT operation.
   - Hyperion functioned nominally after continued PPT operation.
   - Hyperion image-taking during PPT operation to be tested at end of life.
   - Atmospheric Corrector data during PPT imaging currently being evaluated.
   - All spacecraft subsystems performed nominally during PPT operations.
3) On-orbit PPT electrical performance parameters closely matched ground test data.
4) On-orbit thrust calibration measurements are difficult to obtain due to limited knowledge accuracy of other system torques.
5) Method of gauging propellant use could be improved in succeeding PPT designs.
   - Temperature effects introduce high level of noise in fuel bar readings.
6) Thermal modeling of PPT/spacecraft interface requires refinement.
   - Higher than expected, but acceptable, PPT operating temperatures currently under investigation.
7) Increased PPT thrust levels would expand use of PPT as attitude control actuator in future missions.
8) Addressing EMI concerns earlier in the mission may have resulted in being able to eliminate later concerns over PPT operations that were demonstrated by analysis and test to be unfounded.

7. CONTACT INFORMATION

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8. SUMMARY/CONCLUSIONS

The Pulse Plasma Thruster has been successfully validated as a precision attitude control thruster on the EO-1 spacecraft. The PPT has been demonstrated to be compatible with all spacecraft subsystems and all instruments' modes of operations that have been tested to-date. Hyperion imaging during PPT control is scheduled for the end of EO-1 mission life and will complete PPT compatibility tests. All PPT performance parameters appear nominal and correspond with ground measurements. Additional testing of the PPT is scheduled to complete performance evaluation and give insight into life issues. Minor anomalies with PPT fuel bar readings, PPT temperature predictions, and predictions of non-PPT attitude disturbances have not significantly affected the on-orbit evaluation of the EO-1 PPT and do not reveal any problems that may prevent the PPT technology from being infused on future missions. The success of the EO-1 PPT Flight Validation Experiment enables this new generation of PPT technology to be considered for future missions with negligible risk.

9. TECHNICAL REFERENCES

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