AAS 00-076

THE RECOVERY OF TOMS-EP*

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On December 13th 1998, the Total Ozone Mapping Spectrometer – Earth Probe (TOMS-EP) spacecraft experienced a Single Event Upset which caused the system to reconfigure and enter a Safe Mode. This incident occurred two and a half years after the launch of the spacecraft which was designed for a two year life. A combination of factors, including changes in component behavior due to age and extended use, very unfortunate initial conditions and the safe mode processing logic prevented the spacecraft from entering its nominal long term storage mode. The spacecraft remained in a high fuel consumption mode designed for temporary use. By the time the onboard fuel was exhausted, the spacecraft was Sun pointing in a high rate flat spin.

Although the uncontrolled spacecraft was initially in a power and thermal safe orientation, it would not stay in this state indefinitely due to a slow precession of its momentum vector. A recovery team was immediately assembled to determine if there was time to develop a method of de-spinning the vehicle and return it to normal science data collection. A three stage plan was developed that used the onboard magnetic torque rods as actuators. The first stage was designed to reduce the high spin rate to within the linear range of the gyros. The second stage transitioned the spacecraft from sun pointing to orbit reference pointing. The final stage returned the spacecraft to normal science operation. The entire recovery scenario was simulated with a wide range of initial conditions to establish the expected behavior. The recovery sequence was started on December 28th 1998 and completed by December 31st. TOMS-EP was successfully returned to science operations by the beginning of 1999.

This paper describes the TOMS-EP Safe Mode design and the factors which led to the spacecraft anomaly and loss of fuel. The recovery and simulation efforts are described. Flight data are presented which show the performance of the spacecraft during its return to science. Finally, lessons learned are presented.

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* Prepared for technical papers that may later be published in the proceedings of the American Astronautical Society.

INTRODUCTION

The Total Ozone Mapping Spectrometer - Earth Probe (TOMS-EP) is a National Aeronautics and Space Administration (NASA) mission to continue the long-term daily mapping of the global distribution of Earth's atmospheric ozone layer. The satellite was built by TRW for NASA's Goddard Space Flight Center. TOMS-EP collects high resolution measurements of the total column of ozone. The NASA-developed instrument measures ozone directly by mapping ultraviolet light emitted by the Sun to that scattered from the Earth's atmosphere back to the satellite. The TOMS instrument has mapped in detail the global ozone distributions as well as the Antarctic "ozone hole," which forms September through November of each year. In addition, TOMS measures sulfur-dioxide released in volcanic eruptions which may be used to detect volcanic ash clouds that are hazardous to commercial aviation.

TOMS-EP was inserted into orbit by the Pegasus XL booster on July 2, 1996. In the nine days following launch, the spacecraft executed a series of Delta V burns to reach a 500 km circular Sun-synchronous mission orbit with an ascending node mean local time crossing of 11:18 AM. Originally, the data obtained from TOMS-EP were intended to complement data obtained from ADEOS TOMS, which gave complete equatorial coverage due to its higher orbit. However, with the failure of ADEOS in June 1997, the orbit of TOMS-EP was boosted to 740 km and circularized to provide coverage that is almost daily. TOMS-EP is currently the only satellite providing scientific data with an operating TOMS instrument. A QuickTOMS mission is planned for launch in August, 2000 with another TOMS instrument. Figure 1 illustrates the TOMS-EP satellite.

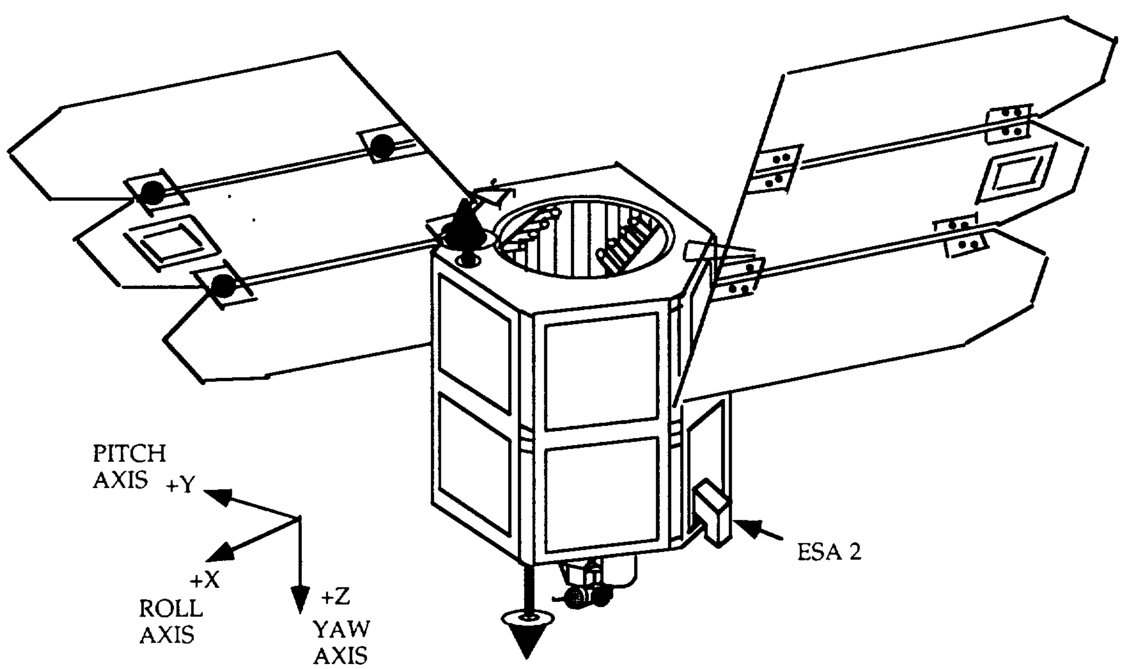

Figure 1 TOMS-EP Satellite

SYSTEM SAFE MODES

To understand the anomaly, it is necessary to understand the system implementation of the active safe modes. The Safe Power Mode uses all standby redundant equipment. It has two submodes, Sun Point Recovery and Long Term Hold, whose functions are defined in Table 1. Both submodes point the +X (roll) spacecraft axis to the Sun. The coarse sun sensor assembly (CSSA) is used for pitch and yaw attitude error and a single two-axis gyro provides rate information about pitch and yaw. The spacecraft undergoes an open loop roll spin-up by two 1 pound hydrazine thrusters prior to entering Long Term Hold.

Table 1
Safe Power Submodes

Mode	Submode	Description	Automatic Transitions
Safe Power	Sun Point Recovery	Two axis inertial sun pointing mode. CSSA and gyro are used as sensors. Thrusters used as actuators	Entry from any other mode due to fault condition. Entry from Long Term Hold due to excessive Sun pointing error.
	Long Term Hold	Spin stabilized Sun pointing precession control mode with two axis rate control. CSSA and gyro are used as sensors. Thrusters used as actuators	Entry from Sun Point Recovery only after successful Sun acquisition. Exit from mode if there is excessive Sun pointing error.

ANOMALY OUTLINE

The anomaly began when an event caused the spacecraft to transition from the prime processor to the redundant processor in response to a critical parameter that exceeded an established limit. The spacecraft successfully aligned the +X axis with the sun line using a two axis inertial controller based on processed coarse sun sensor measurements and a single two axis rate gyro. At this point, the flight software should automatically spin up the spacecraft about the roll axis and transition to a very low fuel consumption momentum based controller. At some point in the transition, the flight software failed to complete the transfer to the momentum based controller. Table 2 provides a concise timeline of events starting just before the processor reboot.

Within approximately 6 hours from entering Safe Power Mode, TOMS-EP had used virtually all of the 25 lb of Hydrazine fuel that remained before the anomaly. The spacecraft was pointed at the Sun, but was uncontrolled and spinning at approximately 18 deg/sec about the +X (roll) axis.

The large amount of thruster activity had a small effect on the TOMS-EP orbit. TOMS-EP is required to stay within an ascending node crossing time of between 11:03 and 11:30. Before the anomaly, ascending node crossing time evolution was not a science life-limiting factor. After the anomaly, the rate of change of the ascending node crossing time was increased by about 3.6 min per year. This rate of change still allows more than 4 years of operation before the ascending node crossing time begins to degrade science collection.

Table 2
Anomaly Timeline

Event	Time	Notes
Corrupted Ephemeris Position (ECI) Data In Telemetry.	347/15:11:26	Previous value of position was 2139.6, 4193.99, -5330.01. Position reading at this time was 1042.67, 5484.46, 4412.70.
Large Pitch Error.	347/15:11:30	Error is calculated by subtracting the onboard propagated position quaternion from the commanded quaternion. Since error did not appear in either roll or yaw, suspect variables for SEU are those related to time (onboard clock, software time or epoch time).
First Thruster Firing To Counter Wheel Spin Down.	347/15:13:06	The first thruster activity occurs more than 1.5 minutes after the Redundant Processor boot is finished. This is the required time to configure the ADCS hardware and initialize Sun Point Recovery. Pitch thruster firings seem to be very clean. The system started virtually sun pointed. Correct thruster pair participate in the removal of wheel momentum as it bleeds into the spacecraft.
End of minimum ten minute window required in Sun Point Recover.	347/15:23:07	The safing logic waits a minimum of 10 minutes in Sun Point Recovery to allow the wheels to run down. This should prevent momentum coupling while the spacecraft spins up.
System begins to monitor the five required conditions necessary to begin the transition from two single axis inertial controllers to a spin stabilized momentum controller.	347/15:23:07	The five conditions required to start the transition are: 1. No presence in Fine Sun Sensor #2 2. Pitch rate within specified threshold 3. Yaw rate within specified threshold 4. Pitch angle within specified threshold 5. Yaw angle within specified threshold At this time, the processed telemetry showed that all five of the conditions above were satisfied. The flight software changes the flag "runup" from 0 (as initialized) to 1 to denote that the system is ready to be spun up.
Start of Roll Spin-up	347/15:23:07	Immediately after the minimum time window, the roll thrusters begin to spin up the spacecraft. Telemetry from the thruster commands shows the total roll on time to be approximately 19.15 seconds. The expected roll rate with this duration pulse should be 3.9 to 4.5 deg/sec. This matches with the algorithm in the flight software and the tank reading in telemetry of 36 counts (8 bit reading) which represents 85 psi. At the start of the roll spin up, the flight software sets the flag "runup" to 2 to let the system know that the roll spin-up has started.
Completion of roll spin up / transition to spin stabilized controller.	347/15:24:04	The telemetry shows that the roll spin-up completed on time and yet the system failed transition to the spin stabilized controller.
Continuous Firing of Pitch Thrusters.	347/15:24:04	Once there was angular velocity in the roll axis, imperfections in the alignment of the inertial and control axes caused a constant pitch rate to appear on the pitch gyro. The inertial control law continuously fired the pitch thrusters to compensate for this rate. The thrusters were ineffective due to the spinning dynamics. A small torque coupling between pitch and roll resulted in a continuous increase in roll rate as the pitch thrusters were fired.
1st Contact after anomaly. Ground acquires downlink with only 3 min to Horizon LOS.	347/16:01:00	Ground observes spacecraft in Sun Point Recovery. Tank pressure 84 psi.
First expiration of sun acquisition timeout.	347/17:08:11	The failure to reach the spin stabilized mode caused the Redundant Processor to reset after 7000 seconds and attempt to acquire the Sun again in Sun Point Recovery. This was the first of three or four resets due to this trigger. The subsequent attempts to acquire the Sun failed due to the system dynamics.
2nd Contact.	347/17:44:00	Ground observes Sun Point Recovery failure to acquire. Tank pressure 78 psi.
3rd Contact.	347/19:19	Ground evaluating problem.
4th Contact.	347/20:57	Ground turns on GRA 1 & 2. Spacecraft processor reset occurs during pass. Tank pressure 77 psi.
5th Contact.	347/22:38	Tank pressure 9 psi. Spacecraft spinning at 18 deg/sec.

ANOMALY CONTRIBUTING FACTORS

There were several factors that combined to produce the state of the spacecraft at the time all of the fuel was spent. This condition is referred to as the "end condition". These factors were distinguished as belonging to one of two classes: factors that were necessary for the end condition and factors that contributed to the end condition. Those that were necessary are:

1. Initial fail over,
2. Wheel bearing friction,
3. Safe mode transition logic,
4. Safe mode design philosophy,
5. Ground controller response.

Those that were contributors are:

6. Location of the failure in the orbit,
7. Thruster force level.

Each of these factors will be examined in the following section.

Factor #1 Initial Fail Over

The anomaly was started by what appears to be a Single Event Upset (SEU) in the on-board Primary Processor. The telemetry stream recorded a jump in the estimated position of the spacecraft at the UTC time 347/15:11:26. This position is calculated onboard to facilitate the nadir pointing function of the attitude control system. The change in position was calculated to be greater than 9888 km in 32.768 seconds. The nominal change in position should be around 245 km.

After identifying and analyzing all reasonable candidates for this anomaly, it is believed the erroneous change in position was due to an SEU in the calculation of the spacecraft state (contained in the ephemeris routine). This conclusion is supported by the fact that:

1. The magnitude of the orbit position vector is consistent between the two vectors. This significantly narrows the possible locations in code for the SEU to occur; and
2. The angle between the position vectors was about 88 degrees. This error appeared in the pitch angle error telemetry as a value of 81.05 degrees (quaternion "small angle" approximation accounts for the difference). Virtually no error appeared in the roll or yaw angle telemetry. This suggests that the spurious position was in the correct orbit plane. Again, this points to a very limited number of points in the processing.

Factor #2 Wheel Bearing Friction

The initial behavior in Safe Power Mode was very nominal. This event represented the seventh entry into Safe Power Mode since the start of the mission and all other entries successfully safed the spacecraft. What made this occurrence different? The key can be found in the timing of the transition from the two axis controlled sun pointing inertial mode (Sun Point Recovery) to the spin stabilized sun pointing momentum based control (Long Term Hold). Initial examination of the playback data showed that there was an anomaly in the dynamics of the spacecraft during the transition between Sun Point Recovery and Long Term Hold. Although there is no direct evidence of the cause because both the attitude decoder electronics (ADE) and the motor driver electronics (MDE) are turned off during Safe Power Mode, the circumstantial evidence presented below points to residual momentum in the wheels.

There is a minimum delay period of ten minutes that the system must spend in Sun Point Recovery before it is allowed to transition to Long Term Hold. This delay was designed to allow for wheel rundown. Thruster activity, gyro readings and CSSA data during this ten minute time period give us important clues about the dynamic condition of the spacecraft upon attempted entry into Long Term Hold. Figure 2 shows the thruster usage within the ten minute delay interval. Note that only thrusters number 2 and 3 are firing and that they are firing in perfect unison. Thrusters 2 and 3 provide positive pitch torque which would be expected as the negative pitch momentum bias is transferred from the wheels to the spacecraft body. Figure 3 shows the spacecraft body rates in the pitch and yaw axes (no roll information is available in the backup mode). The shape of the pitch rate curve shows classic saw-tooth behavior associated with a thruster based controller with a fixed minimum pulse width subject to a near constant disturbance torque (due to the wheel run-down).

The total angular impulse provided to the system in this ten minutes adds up to between 2.0 and 2.25 N-m-sec. This is based on the expected force level of about 0.35 lbf per thruster and the telemetry data which showed 586 counts (2.93 sec) of pitch thruster firing. Since the wheels started with 3.0 N-m-sec of momentum at their nominal 2000 rpm, there was 0.75 to 1.0 N-m-sec of residual momentum in the system when the spacecraft attempted to spin up about the roll axis. This residual momentum would certainly cause the "wobble" observed as the spacecraft began to spin up in roll. This is an unusual case where *lower* than expected wheel bearing drag caused the problem.

Figure 4 is generated from on orbit data and shows a plot of the average voltage needed to keep the TOMS-EP wheels at 2000 rpm over the life of the spacecraft. Based on a linear estimate of the voltage to torque ratio, the drag seems to have leveled off at around 2 mN-m. Figure 5 show the results of a type A scan wheel life test performed at Ithaco over the course of three years. This test was performed under flight like conditions (in vacuum). The data shows that the drag varied from 4.25 mN-m at near beginning of life to around 3.25 mN-m at the end of three years. The lower limit was actually established 16 months into the test.

The shapes in Figures 4 and 5 are very similar. The data suggest that the wheels have reached a steady state and there is no reason for concern over the health of the wheels. The difference is the magnitude of the drop in drag torque. The test wheel showed less then a 25% drop in drag over a 3 year interval. The on orbit wheels show greater than a 60% drop in torque in less than 2 years. The analysis below will show how the unexpectedly low drag torque caused the system to fail.

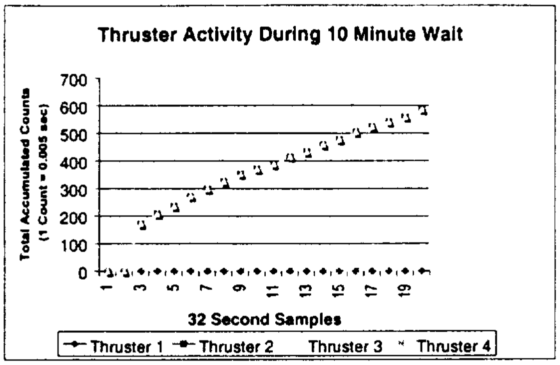

Figure 2 Thruster Firing

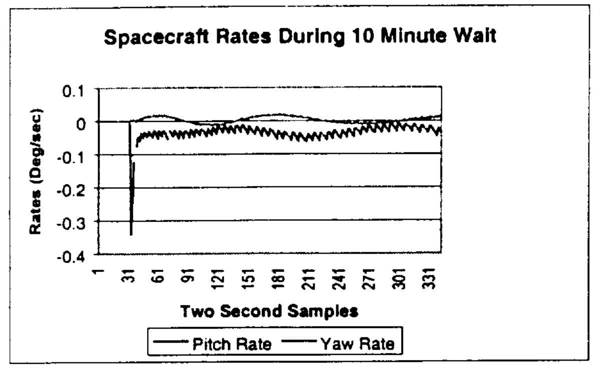

Figure 3 Spacecraft Body Rates

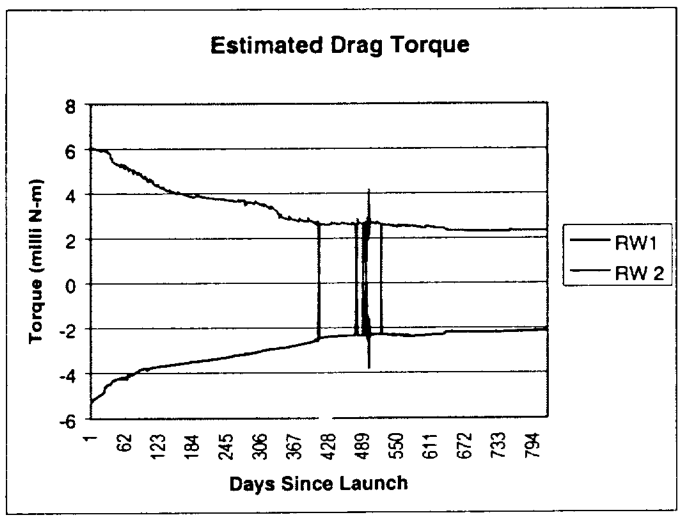

Figure 4 Lifetime Drag Torque (Estimated From Voltage)

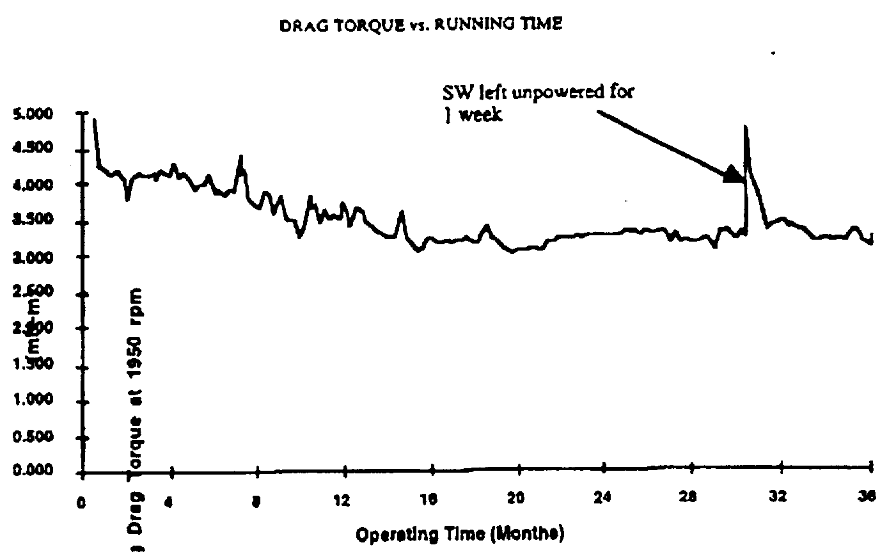

Figure 5 Ithaco SCANWHEEL Drag Torque Life Test Data

Anomaly Simulations

Simulations were run in an attempt to match the behavior of the anomaly. The attitude control and determination subsystem verification simulation (TOMSIM) was used to try and duplicate the behavior of the spacecraft at the time of the failure. Using initial conditions similar to the state of the spacecraft at the time of the failure, the transition from Normal Science Mode to Safe Power Mode was repeated for different levels of wheel bearing drag. The drag value was decreased until the system failed the transition from Sun Point Recovery to Long Term Hold. For reference, the top line in Figure 6 shows the drag torque requirement imposed on Ithaco during the procurement of the wheels.

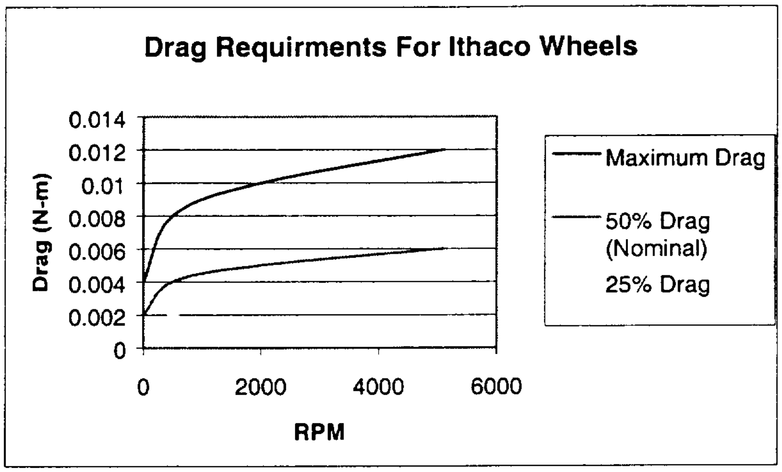

Figure 6 Drag Data Used in Simulations is Derived From Max Drag Requirement

Wheel Drag at 50% of the Maximum Allowed

This simulation shows the expected end of life performance of the Normal Science Mode to Safe Power Mode transition. In this case, the wheel model used the 50% line from Figure 6. Figures 7-10 show the behavior of a system that has the same initial conditions as the anomaly. Figure 7 shows the wheel speeds. The wheel that starts near –2000 rpm is the +Y wheel and the wheel that starts near +2000 rpm is the –Y wheel. At 50% of the maximum specified friction, the wheels are run down before the 10 minute waiting period is finished. Figure 8 shows the spacecraft body rates. A saw-tooth pattern that is similar to the actual anomaly data can be seen. There is a small rate transient when the spacecraft is spun up in roll at around 775 seconds. Figure 9 shows the processed CSSA data which gives sun angles for pitch and yaw. At the time of spin-up, the pitch and yaw error do not exceed 5 degrees. Figure 10 shows the thruster command "on" flags. There is near continuous thruster activity during the spin-up but after the spin-up is completed, thruster usage drops to zero.

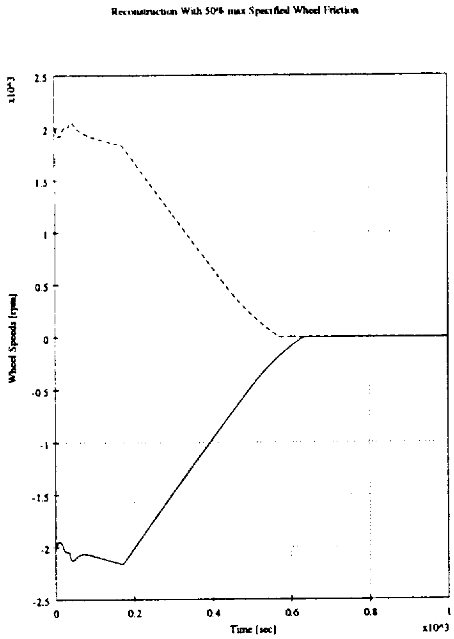

Figure 7 Wheel Spin Down

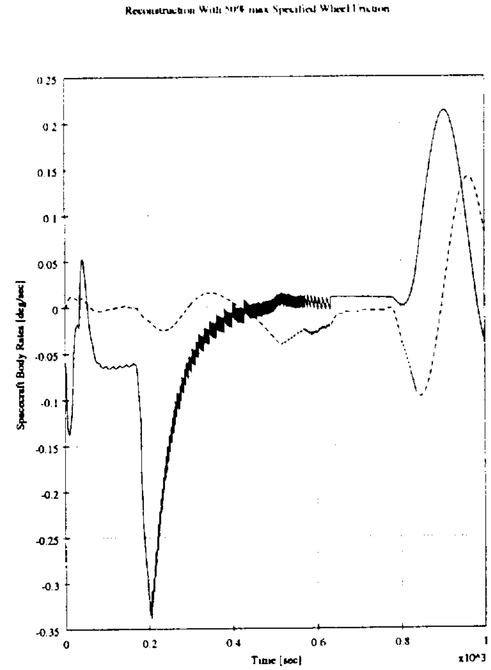

Figure 8 Spacecraft Body Rates

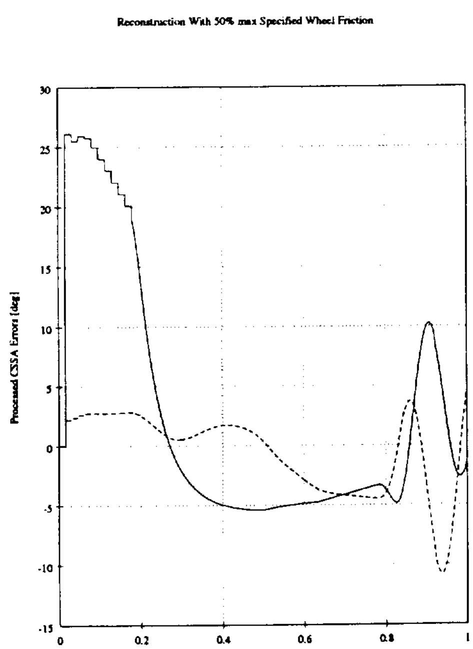

Figure 9 Processed Sun Sensor Angle

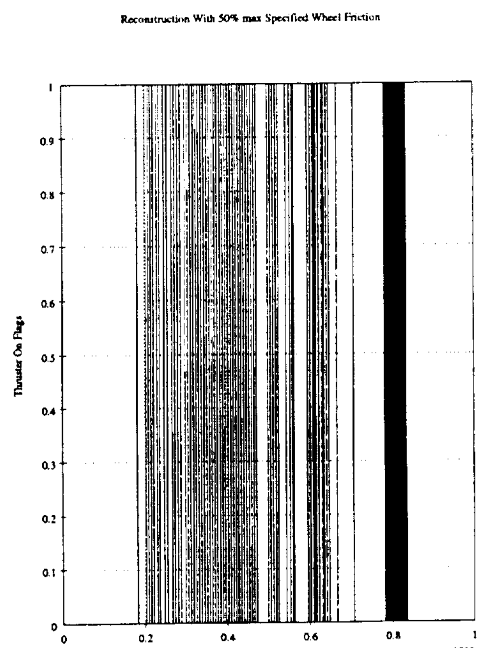

Figure 10 Thruster Commands

Wheel Drag at 20% of Maximum Allowed

The second simulation case presented here shows what happens when there is too much momentum in the system at the time of roll spin-up. Figures 11-14 show the behavior of a system that has the same initial conditions as the anomaly but wheel drag is scaled to 20% of maximum. Figure 11 shows the wheel speeds. At 20% drag, there is still 800 rpm (1.2 N-m-sec) remaining in the wheels when the spacecraft begins to spin up about roll. Figure 12 shows the spacecraft body rates. Coning and nutation are now apparent in the motion of the spacecraft. Figure 13 shows the processed CSSA data which gives sun angles for pitch and yaw. The system is unable to complete the transition from Sun Point Recovery to Long Term Hold because the processed sun angle error is too large. Figure 14 shows the thruster command "on" flags. Since the spacecraft was unable to complete the transition to the momentum based controller, the system is now using a two axis inertial sun pointing control law (Sun Point Recovery) with a high roll rate. This controller is unsuited for systems with a large momentum bias and the pitch thrusters begin to fire continuously in a futile attempt to reduce the observed pitch rate (caused by misalignment of control and inertial axes and the presence of a significant roll rate). The combination of very small misalignments in the thrusters and CG migration over the life of the spacecraft caused a slight pitch/roll torque coupling. As the pitch thrusters continued to fire, the roll rate slowly increased to 18 deg/sec at which point the 25 lb of hydrazine was exhausted.

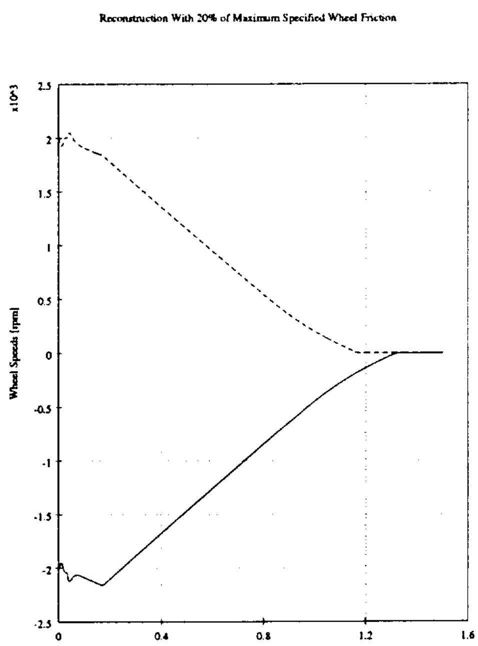

Figure 11 Wheel Spin Down

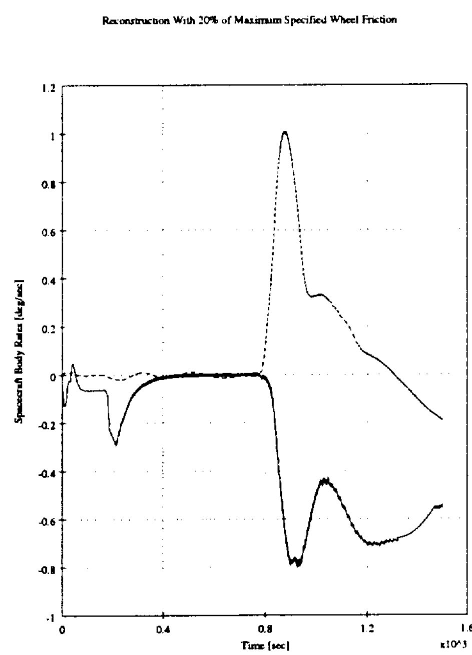

Figure 12 Spacecraft Body Rates

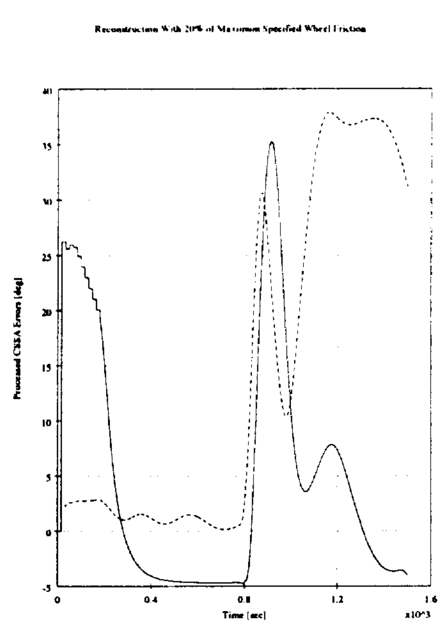

Figure 13 Processed Sun Sensor Angle

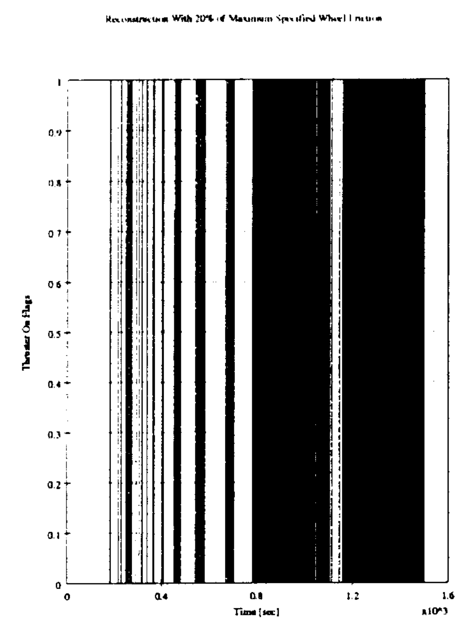

Figure 14 Thruster Commands

Factor #3 Safe Mode Transition Logic

The National Transportation Safety Board (NTSB) approaches investigations with the motto "If any link in the chain of events were broken, the accident would not occur." Of all the contributors, the safe mode transition logic would have been the easiest link to break. The crux of the problem is this: a control flag was used for two purposes, both to turn on and off the momentum controller and to signal the end of the roll spin-up maneuver. The Sun Point Recovery logic interfered with the function of the roll spin-up logic, thus preventing the transition to the momentum control mode.

There are two flags of interest that control the transition to Long Term Hold. These two flags are named "runup" and "isunon" (integer sun control on/off flag). In a nominal scenario, the flight software should go through the procedure outlined below:

1. Sun Point Recovery is entered, "runup" is initialized to 0 and "isunon" is initialized to 1.
2. The spacecraft tries to satisfy the five conditions listed in Table 2 by acquiring the sun and becoming quiescent.
3. When the five conditions are satisfied, "runup" is set to 1.
4. If ten or more minutes have passed, the necessary roll thrust is calculated and the roll thrusters begin to spin up the spacecraft. "runup" is set to 2.
5. When the spin-up is complete, "isunon" is set to 0 to tell the mode transition logic to transition to Long Term Hold.
6. Long Term Hold is initialized with the controller off ("isunon" = 0) and it is usually a day or two before the controller needs to be turned on.

The potential flaw in this logic comes from the fact that the mode transition logic runs at 1.024 second intervals and the Sun Point Recovery controller runs at 0.256 second intervals. There is potential delay between when the "isunon" flag is set to 0 and the mode transition logic reads it. That delay may be anywhere from 30 msec to 798 msec. In that time, the Sun Point Recovery controller may be run either 0,1,2 or 3 times. If the sun angle is outside the 12 degree outer deadzone, the control logic will set the "isunon" flag back to 1 before the mode transition logic can read it.

This logic was tested extensively in both simulation and fixed based test without discovering this potential flaw. That is because the spin-up process does not start unless the angle error is within the 5 degree deadzone and the spacecraft is under active position and rate control during the spin-up. The transition was simulated using worst case thruster misalignments, force mismatch, CG offsets, force vector rotations and nozzle exit location errors. In all cases, the transition to Long Term Hold was achieved.

Factor #4 Safe Mode Design Philosophy

In order to insure that the cause of an anomaly is removed from the system and to eliminate software health checks for equipment, it was decided to use all standby redundant components for the thruster based safe modes. Because of budget constraints, it was impossible to meet the criteria for using all standby redundant equipment in Safe Power and have rate information for all three axes. The choice was made to use a single gyro in the backup mode and maximize the control stability in other ways.

As many "smart" decisions as possible were made to mitigate the lack of roll rate information:

1. Point major moment of inertia at Sun.
2. Use a two-stage safing procedure. The first mode (Sun Point Recovery) is temporary and once the Sun is acquired, the system transitions to a spin stabilized mode that used a momentum controller.
3. The momentum controller was designed to be stable over a wide range of roll rates. Simulation has shown this controller to be stable from 0.75 to > 20 deg/sec.
4. Bias rejection filters were added to remove DC signals associated with roll rate.
5. Control only executed at orbit location where "Earth shine" is minimum.
6. A failsafe check will return to two axis inertial control if momentum controller failed to hold Sun.

It was known that the two axis inertial mode would use a large amount of fuel if significant roll rate accumulated. This was an acceptable risk to the program since the system was designed to pass through this mode in a short period of time.

Factor #5 Ground Controller Response

Once the anomaly occurred, TOMS-EP consumed fuel at a very high rate for a period of approximately 6 hours before the fuel was depleted. Ground controllers had only 4 contacts during this time, the first of which was only 3 minutes long by the time the ground acquired a signal. The other 3 passes were on the order of 10 minutes each.

During the 4 passes, the ground had the ability to disable the thrusters, which in hind sight would have saved fuel and prevented such a high spin-up.

Although ground controllers could have prevented or minimized the effects of the anomaly, it is understandable why they were unable to do so. Although tank pressure was dropping and the spacecraft was spinning up about roll, TOMS-EP remained in the proper Sun-pointing attitude at each contact. Furthermore, while in Safe Power Mode, there is not a direct measurement of roll rate available in telemetry. Further complicating understanding at the time was the fact that when ground controllers turned on other gyros to look at the roll rate, the spacecraft processor reset during the same pass. It was later determined that this was just a coincidental 7000 second timer reset which had nothing to do with turning on gyros.

Factor #6 Location of Failure in the Orbit

The location of the failure had a significant role in the behavior of the system for two reasons. First, after the Redundant Processor had booted up, the spacecraft was almost exactly sun pointed. Figure 15 shows the processed CSSA angles over the entire 10 minute wait period. The maximum angle observed was just above 3 degrees. At the end of the wait period, the spacecraft immediately began to spin up. Second, the presence of Earth shine fooled the spacecraft into thinking it was still sun pointed even after rotating more than 20 degrees. If the spacecraft pitch rate shown in Figure 8 is integrated, it should produce a change in pitch angle as shown in Figure 16. Figure 17 shows the processed CSSA angles with the effects of Earth shine removed. These data match the integrated gyro data much more closely. It is quite possible that in the absence of Earth shine, the system would not have satisfied the five conditions for spin-up immediately and the reaction wheels would have had more time to spin down.

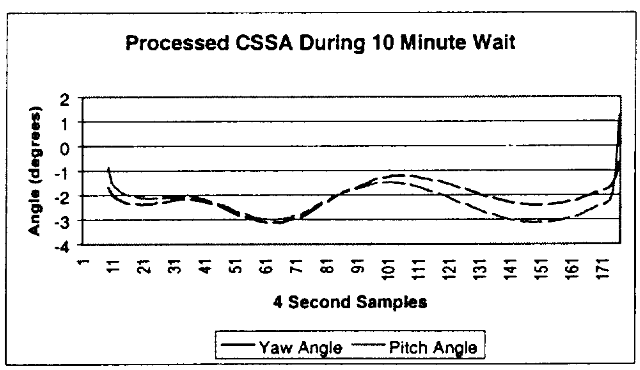

Figure 15 Processed CSSA Sun Angles

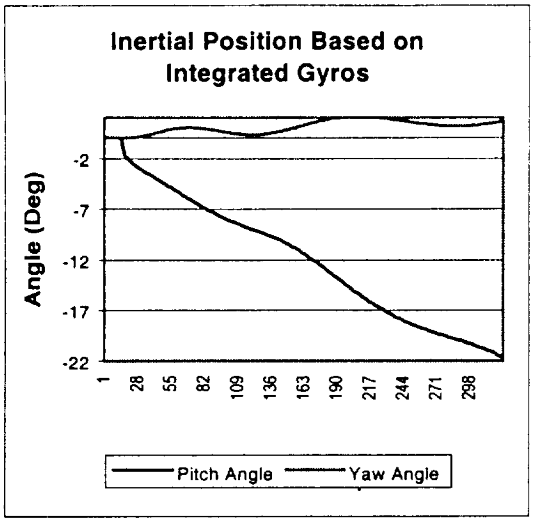

Figure 16 Integrated Rates

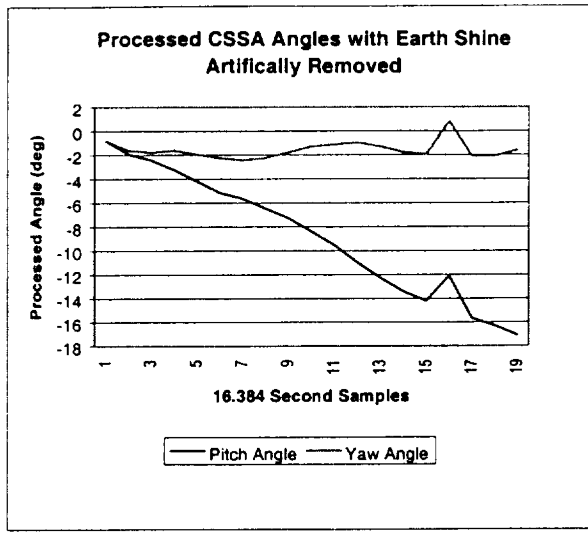

Figure 17 Improved Estimate of CSSA Angle

Factor #7 Thruster Force Level

The last contributor identified was the level of force available at the time of the anomaly. Due to of the initial orbit insertion burns and subsequent orbit change, over 90% of the hydrazine fuel had been exhausted from the blow down propulsion system and the force level was around 35% of the force level available at the beginning of life. The lower force level made the system less capable of countering the ωxH torques generated by the residual wheel momentum.

RECOVERY EFFORT

The focus of the recovery effort was to generate a scenario that was easily implemented and that maximized the probability of spacecraft recovery. To be successful, the recovery must first and foremost maintain the health of the power subsystem.

Power Subsystem Considerations

The original design of the power subsystem makes it very robust to attitude anomalies. The fixed arrays are arranged in a cruciform orientation off of the –Z body axis as shown in Figure 1. Figure 18 shows the power output of the arrays as a function of the solar normal vector (neglecting shadowing effects) scaled to the output of a single sun pointed array. The power system produces enough power to run the spacecraft in all orientations except when the sun is within about 45 degrees of the plus or minus Y spacecraft axis. Since the Y axis is the intermediate axis of inertia, the Sun should not dwell near the axis if there is any significant angular rate in the spacecraft body.

The TOMS-EP battery has 9 amp-hours of capacity. With normal loads, the battery can sustain the spacecraft for about 3 hours without solar array power. It should be noted that there is no provision for "jump starting" the power system after the battery is discharged since the solar array regulators (SARs) are powered from the battery.

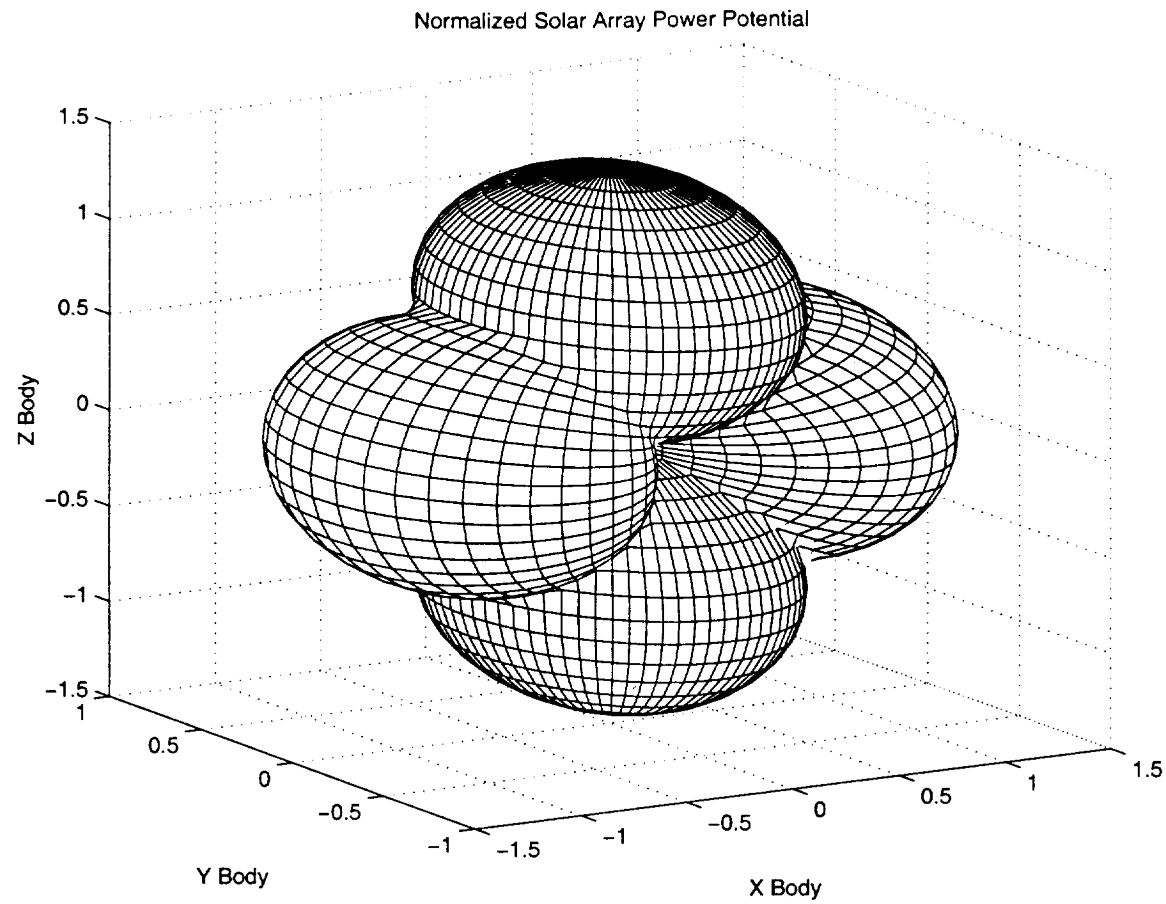

Figure 18 Normalized Power Output For Different Sun Vector Orientations

Attitude Control and Determination System Considerations

The Attitude Control and Determination System (ACDS) was essentially disabled after the fuel was exhausted. A task was immediately undertaken to ascertain the current state of the spacecraft and predict the future orientation with respect to the sun.

The gyros could not be used for on-board roll rate determination because the spacecraft was spinning at 18 deg/sec and the gyros lose polarity at 7 deg/sec. At the beginning of the anomaly, the Sun was within the field of view of one of the FSS; the other FSS was pointed anti-Sun. From the CSSA, it was known that the spacecraft +X axis was pointed within about 5 degrees of the Sun and moving away at a rate of approximately 2-3 degrees per day. Although data from the magnetometer was available, the absolute inertial attitude was very difficult to determine due to the interaction of the high roll rate and processing and telemetry delays. It was certain that if the precession rate observed on the CSSAs continued, the power system would see significant reductions in available power within 2 weeks.

With so little time available, the recovery procedure had to be designed and tested within 10 days. This requirement drove the team toward trying to use the onboard algorithms with minimum modifications. Looking at a block diagram of the TOMS-EP ACDS hardware shown in Figure 19, it can be seen that wheels and torque rods are the only actuators available for maneuvering the spacecraft once fuel is depleted. It was not possible to use the wheels for a large angle maneuver since each was only capable of containing 4 N-m-sec of momentum and the spacecraft body contained about 36 N-m-sec of momentum. The torque rods could be used to slowly maneuver the spacecraft if there was sufficient time. Fortunately, there are two onboard magnetic control algorithms onboard to choose from. The first is a cross product law used for momentum unloading and the second is a B-dot law used for a magnetic Safe Hold Mode. This was the extent of the tools to be used in the recovery.

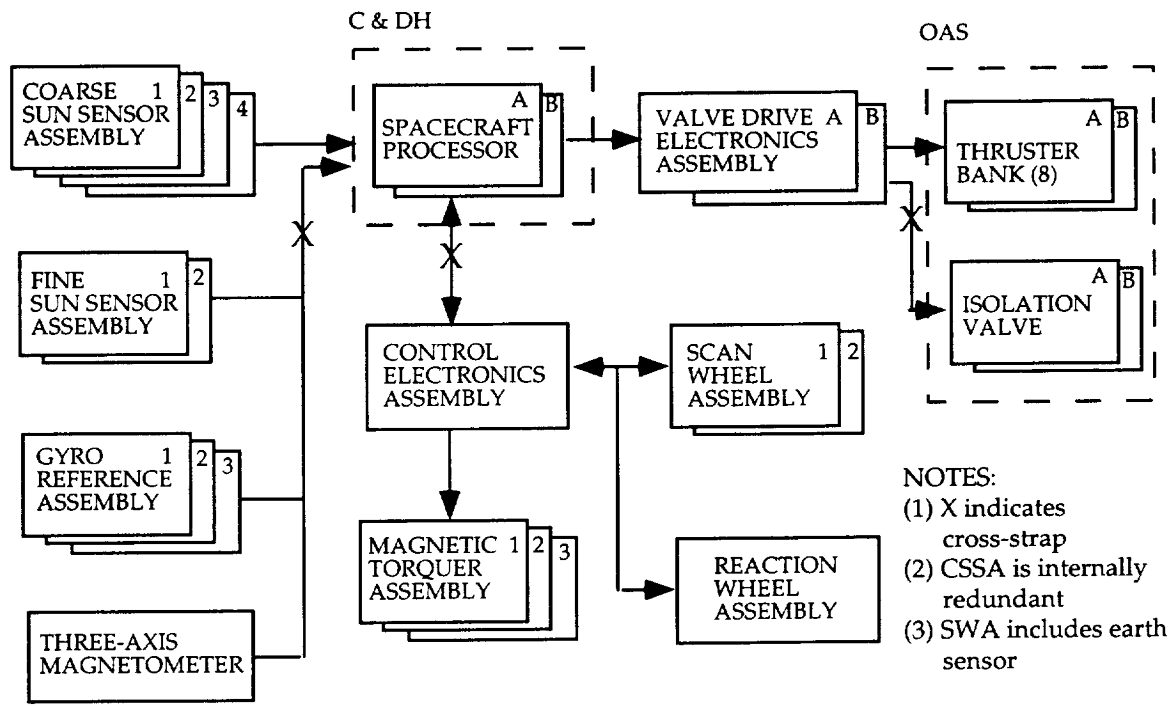

Figure 19 ACDS Equipment Block Diagram

Thermal Subsystem Considerations

The thermal subsystem was designed to radiate most of the heat generated in the spacecraft out the panels on the –Y spacecraft axis. The constraint this placed on the recovery scenario was essentially enveloped by the power system requirements.

Propulsion Subsystem Considerations

Although it was thought that all the fuel in the tank was exhausted, there was known to be fuel in the prime side thruster lines. In addition, there may have been some fuel trapped by the tank bladder against the side of the tank. Immediately after the thruster valves on the redundant side were closed, the pressure in the tank began to rise slowly. Although the recovery could not count on using the impulse from the trapped fuel, an attempt could be made to use it in the most constructive manner possible. When the spacecraft was returned to the prime processor, the Long Term Hold momentum control was selected and the residual propellant precessed the spin vector nearly 10 degrees towards the Sun.

This "last gasp" contribution from the propulsion subsystem gave the recovery team several more days to plan the recovery procedure. It also allowed the team to turn off all propulsion heaters and save power at critical points in the recovery.

RECOVERY PLAN

Of the two onboard magnetic control laws, the B-dot law would need the least amount of modification because it is primarily a minimum energy based design. In the absence of internal momentum, a spacecraft in a polar orbit using a B-dot control law will eventually end up with the maximum moment of inertia (roll) perpendicular to the orbit normal and the roll body rate would approach 2 revolutions per orbit (RPO). If the wheels were spun up to produce momentum in the –Y spacecraft axis, this momentum would end up perpendicular to the orbit plane with 2 RPO rate about the pitch axis.

In this instance, knowing the start and end conditions did not answer the question of whether the spacecraft would pass through an unfavorable power condition somewhere in between. If the wheels were running, the Y axis could act as a pseudo maximum moment of inertia and it is possible for the Sun vector to remain near the Y axis long enough to discharge the battery. The best way to minimize this possibility was to break the recovery into three stages:

1. B-Dot magnetic despin without internal momentum,
2. Wheel capture into the nominal Safe Hold Mode,
3. Science Return into nadir pointing.

Simulations

The entire recovery scenario was extensively simulated using TOMSIM prior to the start of the spacecraft recovery attempt. These simulations calculated both attitude and power potential. Direct measurements from the CSSA provided data on the angle between the spacecraft X axis and the Sun vector. That narrowed the uncertainty in the spin axis attitude to the surface of a cone about the Sun. After the recovery scenario was established, the robustness of the recovery approach was examined by simulating the process using four different sets of initial conditions that resided on the surface of the uncertainty cone. The simulated recovery was successful in all four cases.

Figure 20 shows the spacecraft body rates for a simulated recovery from a position 30 degrees east of the Sun. The simulation predicted a stage 1 duration of approximately 2 days. Although the roll rate could not be directly measure during the actual recovery, Figure 21 shows the rate estimated from the DC bias on the pitch gyro which should be proportional to the roll rate if the system is in a flat spin and has roll-pitch cross products of inertia.

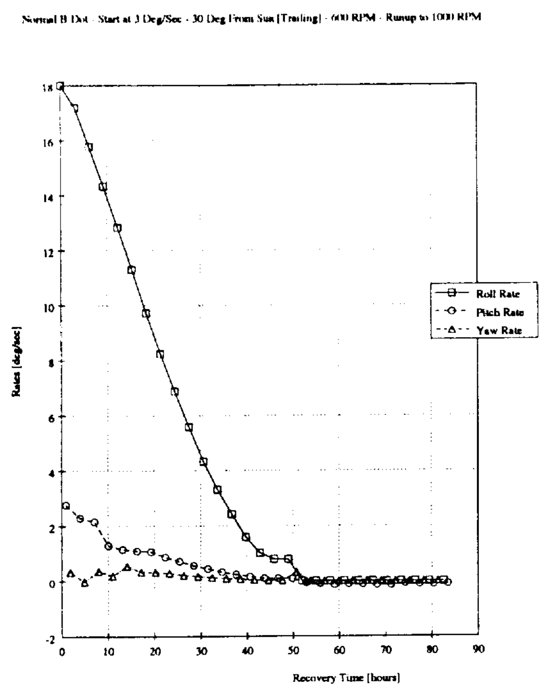

Figure 20 Simulated Recovery Rates

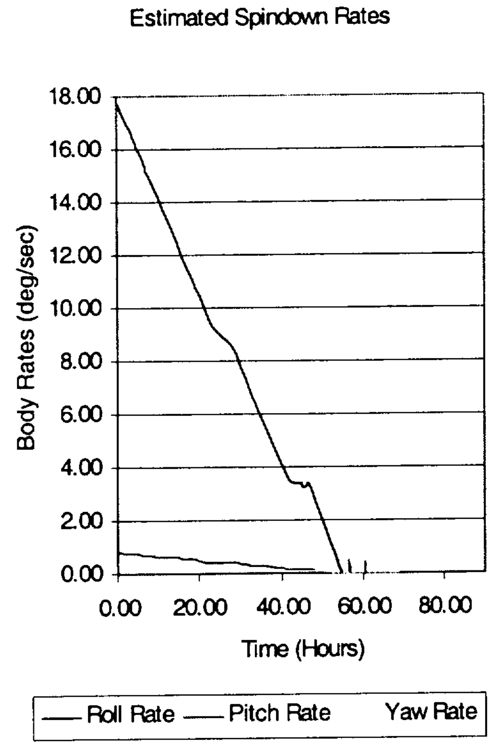

Figure 21 Estimated Recovery Rates

Stage 1

The goal of the first stage was to reduce the spin rate from 18 deg/sec to within the capture range of the Safe Hold Mode (between 2-3 deg/sec). The normal B-dot algorithm processing is executed every 16.384 seconds which allows multiple magnetometer samples to be averaged for noise reduction. In addition, the magnetic field rate is calculated with a differential filter that has a time constant longer than 16.384 seconds. Clearly, the algorithm could not be used successfully with the spacecraft spinning at 18 deg/sec. Changes in the database allowed us to successfully reduce the number of magnetometer samples to 1 and change the characteristics of the differential filter. The next step involved changing the flight software executive to call the B-dot algorithm every 2.048 seconds. Fortunately, this was accomplished by replacing a single byte in an inequality statement. The final "high speed" B-dot algorithm had 1.024 seconds allocated to magnetometer sampling and 1.024 seconds allocated to torque rod firing.

The question still remained whether the magnetic despin would put the spacecraft in an unfavorable power attitude. If you base your guess on the known end condition, you might assume that the maximum moment of inertia would be pushed perpendicular to the orbit plane. In fact, just the opposite is true. Since the body rates are much higher than orbit rate, the B-dot algorithm simply sheds energy wherever it can. The key to understanding its behavior is in the available torque. In a near polar orbit, the magnetic field remains close to the orbit plane (rotating twice per orbit). If you break the total system momentum into the portion projected into the orbit plane and the portion perpendicular to the orbit plane, there is *always* magnetic torque available to reduce the momentum perpendicular to the orbit plane and four points in the orbit where it is very difficult to affect the momentum in the orbit plane. Thus as long as the body rates remain high relative to orbit rate, the maximum moment of inertia will remain close to the orbit plane.

Since TOMS-EP resided in a sun synchronous orbit with an 11:00 to 11:30 ascending node, pulling the maximum moment of inertia toward the orbit plane should not degrade the power potential.

Stage 2
Stage 2 was the riskiest portion of the recovery scenario. At some point, the spacecraft had to transition from spinning about the maximum moment of inertia to spinning about the Y axis. The transition had the potential of pointing the spacecraft in a low power attitude for an extended period of time.

Once TOMS-EP was despun to 3 deg/sec, the nominal B-Dot Mode software parameters were restored. The wheels were set to their minimum rotation rate to minimize the time in transition between sun pointing and normal B-dot pointing. The spacecraft was prepared for the low power attitude by shedding *all* loads not directly involved in the recovery. These loads included prime and redundant platform heaters, prime and redundant propulsion heaters, gyros, and all transmitters. In this configuration, the average load was reduced to 45 watts (1/3 of orbit average). Essentially, it was up to the physics of the B-dot controller to complete the transition. Interference from the ground would only reduce the chance of a successful recovery.

Two orbits after starting stage 2, contact was re-established with the spacecraft. The system had settled with the pitch momentum bias perpendicular to the orbit plane. The wheel speeds were slowly increased until they matched the normal Science Mode speeds.

Stage 3
In the final Science return stage, TOMS-EP was commanded into its originally designed Science Return Mode. Although never used prior on-orbit, the mode was well tested and simulated prior to delivery of TOMS-EP. This mode allowed automatic transition into Science Mode within one orbit.

POST RECOVERY FAULT MANAGEMENT IMPLICATIONS

The loss of the propulsion subsystem left the Safe Power Mode incapable of active control. If the spacecraft switches to an uncontrolled mode, it is known that the momentum stored in the pitch momentum bias will eventually end up as a roll rate of +/-1.9 deg/sec. This should be sufficient to prevent complete battery discharge.

It is still preferred to keep under active control if possible. For this reason, the onboard fault detection software was modified to minimize the number of faults that send the system to Safe Power Mode. Only those faults that require reconfiguration of power system or processor faults send the system to Safe Power Mode. All other faults (pointing anomalies, wheel speed delta, etc.) cause a transition to the B-dot Safe Hold without switching processors.

LESSONS LEARNED

1. SEUs happen. Be ready.

2. Where it is possible, directly measure states that are driving decisions. In our case, it would have been preferable to measure the wheel speeds directly and start sun acquisition after they had completed their run down. Unfortunately, that was not possible because tachometer data is unavailable when the motor drivers or the attitude decoder electronics are off. Additional fault management risk would have been assumed if the wheel electronics were left on in Safe Power Mode.

3. Designers often concentrate on accommodating behavior associated with "worst case" conditions (CG locations, misalignments, friction, structural flexibility etc.). Sometimes, an ideal CG location, perfect alignments, better than expected friction or higher than expected stiffness can cause problems. These should be considered also.

4. Do not use flags for multiple purposes no matter how closely related they are. Carefully check the logic of flags that are set and read asynchronously.

5. The emphasis in fault management at TRW has shifted since the design of TOMS-EP. In subsequent programs, the inherent robustness of the safe mode was considered to be more important than using standby redundant components. Even in light of the TOMS-EP on orbit experience, this is not a clear cut decision. To address the issue of using a single gyro for safe modes on the new programs, flight software chooses which pair of gyros to turn on based on a number of comparison tests. It has been demonstrated in fixed based test that these tests can be fooled under certain circumstances. If the software chooses a failed gyro, the consequences will be worse than the TOMS-EP anomaly.

6. Murphy works smarter than you do. Rely more on general principles to prove system robustness rather than attempting to find degenerative cases.

7. A number of spacecraft have been lost or nearly lost due to anomalous autonomous thruster operation. If a spacecraft has the capability to use wheels rather than thrusters to acquire its safe mode orientation, then it is usually prudent to use wheels over thrusters. Although thruster hardware and the associated electronics are very reliable, a system using expendables always introduces a risk of imparting unwanted high momentum to the spacecraft. This high momentum input can be caused by spacecraft hardware, logic or software anomalies. A wheel-based safe mode limits the amount of spin-up while in the mode, should an anomaly occur.

8. The canted, double-sided solar array orientation on TOMS-EP is very forgiving of spacecraft attitude anomalies. Adequate power can be generated from most spinning or

tumbling conditions. This was of great relief while TOMS-EP was spinning uncontrolled for 18 days. Multiple solar array viewing angles increase the robustness of a spacecraft to anomalies.

CONCLUSIONS

The TOMS-EP spacecraft was successfully recovered in less than 3 weeks from a severe anomaly that depleted all fuel and left the spacecraft uncontrolled with a high spin rate. A team of engineers and spacecraft operators quickly determined the cause of the anomaly and implemented a recovery effort. The TOMS-EP satellite continues to successfully perform its mission well beyond its design life of mapping the global distribution of Earth's atmospheric ozone layer.