ATTITUDE CONTROL AND ORBITAL DYNAMICS CHALLENGES OF REMOVING THE FIRST 3-AXIS STABILIZED TRACKING AND DATA RELAY SATELLITE FROM THE GEOSYNCHRONOUS ARC

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Launched on April 4, 1983 onboard STS–6 (Space Shuttle *Challenger*), the First Tracking and Data Relay Satellite (TDRS–1) was retired above the Geosynchronous Orbit (GEO) on June 27, 2010 after having provided real-time communications with a variety of low-orbiting spacecraft over a 26-year period. To meet NASA requirements limiting orbital debris[1], a team of experts was assembled to conduct an End-Of-Mission (EOM) procedure to raise the satellite 350 km above the GEO orbit. Following the orbit raising via conventional station change maneuvers, the team was confronted with having to deplete the remaining propellant and "passivate" all energy storage or generation sources. To accomplish these tasks within the time window, communications (telemetry and control links), electrical power, propulsion, and thermal constraints, a spacecraft originally designed as a three-axis stabilized satellite was turned into a spinner. This paper (a companion paper to "Innovative Approach Enabled the Retirement of TDRS–1," paper # 1699, IEEE 2011 Aerospace Conference, March 5–12, 2011[2]) focuses on the challenges of maintaining an acceptable spinning dynamics, while repetitively firing thrusters. Also addressed are the effects of thruster firings on the orbit characteristics and how they were mitigated by a careful scheduling of the fuel depletion operations. Periodic thruster firings for spin rate adjustment, nutation damping, and precession of the momentum vector were also required in order to maintain effective communications with the satellite. All operations were thoroughly rehearsed and supported by simulations thus lending a high level of confidence in meeting the NASA EOM goals.

INTRODUCTION

Brief Attitude System Description

TDRS–1 Attitude Control System (ACS) was designed as a bias-momentum type using dual redundant Reaction Wheel Assemblies (RWAs), each pair configured as a V providing 44 Nms of momentum about the satellite's pitch axis. When driven in the same direction the wheels provide

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pitch control, driven differentially they control roll. Yaw is passively controlled through gyroscopic and orbital kinematics coupling. Twenty-four hydrazine monopropellant thrusters (12 in each of two redundant branches) were used for orbit maintenance (station-keeping) and unloading of the secular momentum buildup. Pitch and roll angular attitude deviations about the orbit normal axis and along the velocity vector respectively were measured by a dual redundant scanning Earth Sensor Assembly (ESA). Yaw attitude information was not required for attitude control of the satellite because of the orbital coupling inherent to the bias momentum, but it was needed for open-loop steering of the antennas. Since yaw attitude was only periodically observable (near 0600 and 1800 local satellite time) by means of the Fine Sun Sensor Assembly (FSSA) mounted on the solar arrays, ground software, using a state-space estimator, provided the yaw angle during the rest of the time. Two Gyro Reference Assemblies (GRAs), each containing two miniature rate-integrating gyros (used for station-keeping and attitude reacquisition), had their input axes arranged normal to the surfaces of a pyramid so that any three gyros could provide three-axis information, thus assuring redundancy. Figure 1 shows TDRS–1 on-station.

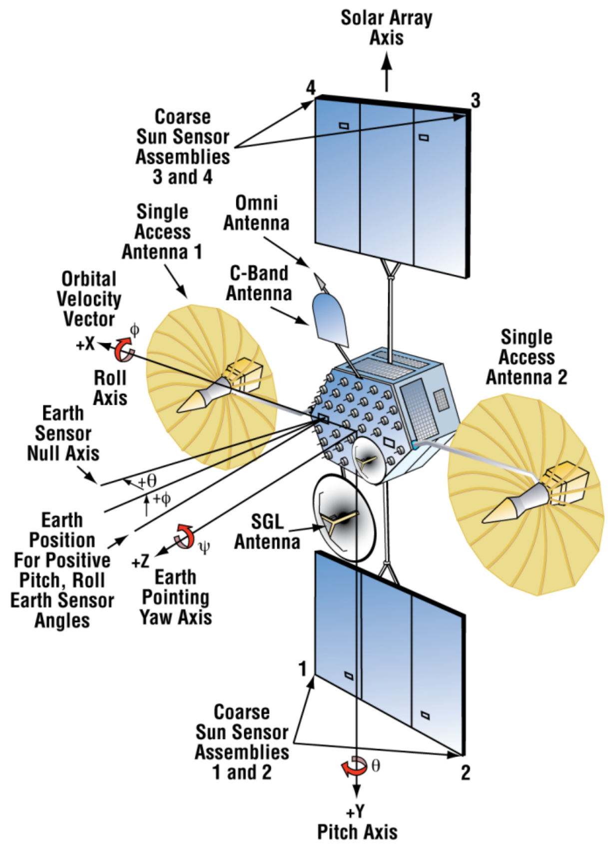

Figure 1. TDRS–1 on-station configuration.

On-Station History

TDRS–1 was launched on April 4, 1983, from the Space Shuttle Challenger (STS–6). During the final orbit insertion, the second stage of the Inertial Upper Stage (IUS) malfunctioned leaving the satellite stranded in transfer orbit. Operational orbit at GEO altitude was successfully reached, however, using two 4-newton station-keeping thrusters. Later, while attempting Earth Pointing Acquisition, a thruster overheated causing a suspected hydrazine detonation that ruptured the A-side propulsion line near the negative roll thruster. This event disabled the entire A-side propulsion system and for precaution the B-side negative roll thruster (because of its proximity to the A-side thruster) was also taken out of service. Orbit control maneuvers were subsequently developed to compensate for the total loss of negative roll thrusters.

ACS thruster overheating remained a problem throughout the life of the spacecraft. Whenever maneuvers required prolonged, low duty-cycle use of the ACS thrusters (i.e., less than 10 percent), elevated thruster temperatures could occur resulting in performance degradations and possibly triggering another hydrazine detonation. Thus, thruster thermal issues eliminated the original north-south station-keeping plan, because it required extensive use of ACS thrusters at low duty-cycles. Not needing north-south inclination control at the beginning of life resulted in an abundance of unspent fuel at the end of life, which ultimately had to be disposed of as part of the TDRS–1 EOM activities.

TDRS–1 experienced other anomalies during its long life, but continued to provide a variety of communication services. Finally, when the last of its six Traveling Wave Tubes (TWT) wore out in October 2009, all ground communications to and from the satellite were transferred to the low-bandwidth omnidirectional antenna. Decommissioned from users' services, TDRS–1 was moved to a temporary orbital slot at 56.5°W longitude awaiting final disposal.

SATELLITE DISPOSAL REQUIREMENTS

The technical standard, NASA–STD–8719.14, *Process for Limiting Orbital Debris,*[3] specifies two main criteria: raising the orbit such that it would remain at or above GEO +200 km (35,986 km) altitude for a period of at least 100 years, and passivation of all stored energy sources. Although TDRS–1 was not designed to meet these current disposal requirements, a TDRS–1 EOM team of experts was assembled to meet this challenge. Analytical studies, simulations, and procedures were developed to perform the tasks summarized below:

- Raise the orbit 350 km above GEO.
- Deplete fuel while maintaining the orbit approximately 350 km above GEO.
- Power down the momentum wheel while maintaining attitude.
- Permanently discharge the battery and remove it from the bus.
- Permanently turn off any RF radiation sources and disable all active RF receivers.

Pressurant (gaseous nitrogen) passivation was the only orbital debris requirement that was not met because the propellant tank was designed with a diaphragm which prevented venting. A waiver was granted for this requirement.

CHALLENGE MEETING THE DISPOSAL REQUIREMENTS

The most severe challenge faced by the EOM team was the depletion of a substantial amount of fuel within the constraints and limitations of a spacecraft that was designed prior to any orbital debris considerations. Furthermore, the partial failure of the propulsion system and the seriously degraded conditions of the thrusters compounded the operational complexity. The following were among the key constraints:

Maneuver Burn Time Limitation

TDRS–1 thrusters were subject to a unique phenomenon referred to as "thruster choking," which reduced the flow of propellant through a thruster, both diminishing its control authority and causing a rise in temperature since less heat was transferred away by the propellant flow. This condition limited TDRS–1 thruster operations to 100 sec or less for most of its operational life, but a later re-evaluation determined that 600 sec was a safe limit for TDRS–1 propulsion system conditions in maneuver mode. With an estimated 22 hours of thruster burn time required to deplete the remaining propellant, the 10-min burn constraint would dictate that 132 such burns be performed. A combination of thruster thermal diurnal variation and the optimal orbital position for burn execution (i.e., apogee or perigee) would further dictate that the burns be separated by 12 or 24 h.

Eclipse Season Implications

All TDRS–1 EOM activities needed to be completed before the start of the fall eclipse season (July 11, 2010) to lower the risks in case an attitude recovery became necessary. TDRS–1 attitude recovery procedure involved the use of a Sun-pointing mode, which would of course have been impacted by an eclipse. In addition, as various fuel-depletion modes were considered, positive electrical power balance also became a concern.

Over-The-Horizon (OTH) Operations

Since ground stations other than the WSC (White Sands Complex, New Mexico) usually experienced significant command loss and telemetry dropouts when communicating with TDRS–1, the EOM activities were constrained to a direct link with WSC. Letting the satellite drift around the Earth and waiting for its reappearance in the WSC LOS (Line-Of-Sight) was considered too risky due to the onboard computer RAM (Random Access Memory) susceptibility to SEU (Single Event Upset). Based on the nominal orbit-raising plan which would induce almost a 5°/day westward longitudinal drift, it was estimated that TDRS–1 would be out of view of WSC (165° W sub-satellite longitude point) on July 4, 2010. EOM had to be completed before this time.

ATTITUDE CONFIGURATION OPTIONS FOR FUEL DEPLETION OPERATIONS

The first priority was to raise TDRS–1 to an orbit above GEO (compliant with the requirements stated in the above section) via a series of 600-sec burn duration maneuvers. Additionally, maneuvers had to begin at the end of the spring eclipse season and once thruster thermal conditions became more favorable. Managing the power system, getting on and off the batteries during the eclipse season would have unduly added complexity to the operations. As a result of these constraints, orbit raising could not start prior to June 5, 2010 and would not complete until June 13, 2010. The remaining EOM activities, fuel depletion and passivation, could then start and be complete before TDRS–1 drifted out of view from WSC. This constraint heavily impacted the viability of the possible fuel depletion options. Prior to orbit raising, TDRS–1 was estimated to have 139.47 kg of propellant. The orbit raising was to use 13.06 kg of propellant, leaving an estimated 126.41 kg that needed to be depleted. Following are the fuel depletion methods considered and their respective advantages and drawbacks.

ΔVelocity Maneuvers

Performing a series of ΔVelocity Maneuvers (east-west station-keepings in particular) was the first approach to be considered since it consisted of routine operations performed during TDRS–1 life. However, the 600-sec maneuver time limitation (discussed previously in "Maneuver Burn Time Limitation") would have required two burns per day for 60 days to deplete the fuel. The time window available, constrained by the eclipse season and the OTH requirements, eliminated east/west or north/south maneuvers as a viable option.

Ground-Commanded Thruster Firings

Another routine operation already performed on TDRS–1 was the ground commanded thruster firings used to dump momentum. But again, considering the depletion time window, this method would have required 1,098 command sequence transmissions per day, or 46 per hour. This approach was deemed operationally impractical, as it required near-continuous command activity, around the clock, for approximately 17 days in order to expend the remaining propellant.

Propellant Venting from the A-Side Leak

Consideration was given to utilizing the ruptured A-side propulsion system to deplete the fuel. This approach sought to vent hydrazine from each propellant tank into space by opening the isolation valve connecting that tank with the A-bank of thrusters. The approach was attractive because it would have depleted fuel with little or no impact to the orbit (due to the low exit velocity of the propellant through the leak), and it was an efficient method of removing propellant from the spacecraft (0.54 kg could be expended per minute). Given this rate, the total residual propellant mass could be removed with approximately 233 min of isolation-valve open time. However, several unknowns about the leak and the valves were considered serious risks. These unknowns included: the ability to control spacecraft attitude while hydrazine vented out the ruptured propellant line; the potential for hydrazine to freeze in the propellant line(s) and at the thruster valve; the ability of the isolation valves to support the required on-off cycling; and the potential for fuel detonation in the propellant line due to the "water hammer" effect. Ultimately, the risk was considered too great given the unknowns.

Z-spin:

Another approach was to fire the station-keeping thrusters (Z_1 and Z_2) while spinning the spacecraft about its Z-axis. The thruster pair torques would approximately cancel each other (see Table 1) while the spin would serve to stabilize attitude and average out most of the ΔVelocity imparted to the orbit. The depletion burns could be performed without any active thruster attitude control, thus eliminating the thruster choking issue (i.e., no thrusters operate in problematic duty cycle regions) that limited normalΔv maneuver durations. Two orientations of the Z -axis were considered.

Z-spin, Sun-pointed: From an electrical power perspective, having the Z-axis Sun-pointed was an attractive orientation. The solar arrays could be positioned to receive full Sun continuously as in normal operations. The communications geometry would not always be favorable since the hemispherical omni antenna would spend half the orbit pointing away from Earth. Rather than lose communications for half an orbit, an attitude maneuver could be performed twice per orbit around spacecraft 6 AM and 6 PM to keep the omni antenna in view of WSC. The maneuver would serve to pitch the spacecraft 180°, maintaining good communication for the next half-orbit and also slewing the solar arrays to maintain Sun pointing. These half-orbit maneuvers added operational complexity and also posed some risk of thruster choking while being executed.

Z-spin Orbit Normal: This Z-axis spin approach would orient the spacecraft +Z-axis in the direction of the orbit normal before introducing a negative spin rate. While solar array power input would be reduced and the communications link would operate near the edges of the omni antenna pattern, this approach was selected. The satellite maximum moment of inertia axis spinning normal to the orbit plane presented low risks, a minimal procedural complexity, and a greater spacecraft stability. Additionally, it allowed long thruster burns, thus gave sufficient time to complete the fuel depletion. However, this approach was not without concerns for the other spacecraft systems: power, communications, thermal, etc. These concerns and how there were dealt with are addressed in Reference (1).

ORBIT RAISING AND SATELLITE MANEUVERING TO THE SELECTED Z-ORBIT NORMAL

Orbit Raising Maneuvers

As already mentioned, the first priority was to vacate the GEO arc. NASA–STD–8719.14[4] stipulates that the minimum safe perigee for a spacecraft with TDRS–1's physical properties (i.e., spacecraft area/mass ratio, and reflectivity) is 290 km above GEO (to meet the 100-year requirement). Considering that approximately 126 kg of hydrazine needed to be depleted once the disposal orbit was reached, a perigee altitude of GEO+350 km was targeted as a precaution against possible orbit lowering during the depletion operations.

This increase in orbital altitude would require a change in velocity of 12.6 m/s. Based on previous Station-keeping and Relocation maneuvers, it was projected that twelve 10-minute burns would be necessary to achieve the ΔVelocity. The orbit-raising performance was within the expected uncertainties caused by the off-pulsing of the thrusters for attitude maintenance and the progressive decrease in tank pressure. Actually, 12 orbit raising burns were executed, each 630 sec in duration.

The final perigee altitude (above GEO) was 351.72 km, about 4 percent less than targeted.

Roll Maneuver to Orbit Normal

The satellite Z-axis was then maneuvered to orbit normal via a positive 90° roll maneuver. The satellite was commanded into an inertial hold mode using integrated gyroscope information for attitude maintenance and then rolled 90° by firing a combination of the positive roll thruster and the yaw thrusters. A combination of yaw thrusters with no resulting yaw torque was fired as a substitute for the unavailable negative roll thrusters. Based on simulations and gyro accuracy, it was predicted that the Z-axis would settle within 1° of orbit normal, pointing northward. Prior to performing the roll maneuver, the solar arrays were pointed along the +X axis (90° from their appearance on Figure 1).

Z-Axis Spin-Up

After pausing to allow the spacecraft to settle into its new orientation, all onboard attitude control modes were disabled and the satellite was spun up to approximately −1.0°/s. (in two −0.5°/s steps) via simultaneously firing the Z_1 and Z_3 yaw thrusters. This two-step approach mitigated the nutation build-up and resulted in a minimum spin axis offset (0.4° at most) from the orbit normal. Nutation and precession resulting from disturbance torques would have been lessened at a higher spin rate. However, if the spacecraft Z-axis were to tilt away from orbit normal, the omni antenna view of the ground station would have been blocked periodically by spacecraft structure, requiring reacquisition of the command link, a difficult operation at a higher spin rate. Simulations indicated that a spin rate of −1.0°/s was an acceptable compromise between spin stiffness and communication integrity. The thruster orientation with respect to the control axes is shown in Figure 3. Table 1 shows the thruster (limited to those used during the EOM operations) torques corresponding to a nominal thrust of 1.0 lbf at the beginning of life. A scale factor was applied to the thrust used in the estimation of the thruster firing effects to account for some degradation over life and a lower pressure in the tanks.

Using the following roll, pitch, and yaw torque components for the Z_1 and Z_3 thrusters (converted to Nm):

$$T_x = -0.023 \text{ Nm} \qquad T_y = -0.015 \text{ Nm} \qquad T_z = 5.1 \text{ Nm}$$

The time required to spin up to −1 deg/sec was about 39 seconds, resulting in a momentum about the Z axis of 200 Nms (I_{zz} = 11,496 kgm^2). The momenta built up about the transverse axes were M_y = 0.595 Nms and M_x = 0.905 Nms, or a momentum in the roll/pitch plane of $[M_y^2 + M_x^2]^{-2}$ = 1.08 Nms. This corresponds to an angular Z axis tilt from the orbit normal of: γ = Arctan (1.08÷200) = 0.31 °.

Although the nutation angle could not be measured directly, it could be calculated by observing the pitch and roll rates (given by the gyros) cycling in quadrature.

The nutation angle θ is given by the relation:

$$\sin\theta = I_T\,\omega_T \div H \qquad \text{Equation (1)}$$

Where I_T is the transverse inertia ($I_T = I_{xx} = I_{yy}$ for the case of an axis-symmetric spacecraft), ω_T the resultant transverse angular rate, and H the magnitude of the angular momentum. For TDRS–1, an acceptable approximation for I_T could be:

$$I_T = (I_{xx} + I_{yy}) \div 2 = (8258 + 4806) \div 2 = 6532 \text{ kgm}^2$$

From Fig. 2a, the roll (ω_x) and pitch (ω_y) angular rates are approximately 0.012 deg/sec. The resulting transverse angular rate is then: $\omega_T = (\omega_x^2 + \omega_y^2)^{-2}$ = 0.017 deg/sec or 0.3 10^{-3} rad/sec.

Equation (1) gives the nutation angle: θ = Arc sin (6532 x 0.3 10^{-3} ÷ 200) = 0.6°

At completion of the Z axis spin-up, this 0.6° nutation with a 8-minute period was observed using the gyro information (Fig. 2a).

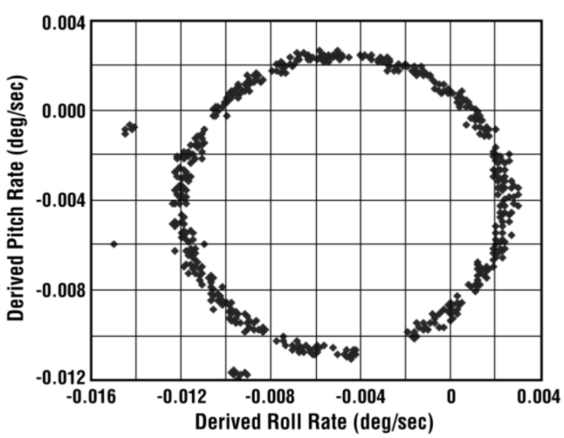

Figure 2a. Nutation cycle post spin-up.

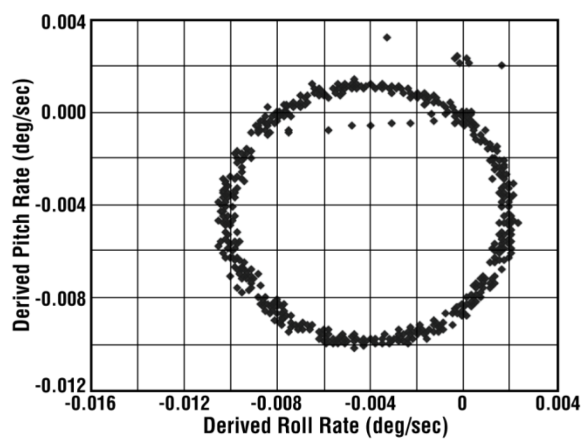

Figure 2b. Nutation cycle pre fuel depletion burn.

Fig. 2b is a plot of the nutation cycle, 20-hours after the end of spin-up. It indicated that the nutation angle had decreased by approximately 6 percent due to the energy dissipation caused by the satellite flexible appendages and the fuel slosh in the tanks (a reflection of torque-free rotational dynamics).

ROTATIONAL DYNAMICS OF Z-SPIN ORBIT NORMAL

In the Z-Spin-Orbit Normal option, the satellite is spinning normal to the orbital plane about its axis of maximum moment of inertia. Although inherently stable, one of the main dynamics concern was the spin axis stability while firing thrusters to deplete fuel. Additionally, would an adequate knowledge of the inertial orientation of the angular momentum vector be available to control it if required?

The orientation of the angular momentum vector was not directly measurable. However, the CSSs (Coarse Sun Sensors) mounted on the outmost corners of the solar panels could be used to provide some clock-angle information with respect to the Sun line as the satellite spun. On station, with the satellite Y axis normal to the orbital plane, the CSSs provide signals (Pitch and Yaw) for the Sun Mode, a recovery mode used in case Earth pointing is lost. Once maneuvered with its Z axis normal to the orbit, the CSS yaw channel crossed through zero as the array normal pointed in the Sun direction. By defining a phase angle from the yaw zero crossing, the time to start thruster firing was determined. The orientation of the momentum vector tilt angle was obtained as a product of the orbit determination process provided by the GSFC Flight Dynamics Facility (FDF). For the record, it should also be mentioned that monitoring of the AGC (Automatic Gain Control) of the T&C link was also available as backup.

Analytical Discussion

The principal axes of a body are determined by the mass properties, and TDRS–1 was designed so that the principal axes are nearly aligned with the body axes, the largest moment of inertia being nearly aligned with the spacecraft Z-axis. A well-known property of spinning bodies is that rotation about the axis with the maximum moment of inertia is stable and will continue without change in the absence of external torques.

In spinning dynamics or uncontrolled rotational motion a distinction needs to be made between rotation about a principal axis (an axis about which the products of inertia are zero) and any other body axes. In a so-called “pure rotation,” the body is spinning about the principal axis, and the principal axis is aligned with both the angular spin vector ω_s and the angular momentum vector H. However, when the angular rate ω_s is not parallel to the principal axis, nutation occurs. The principal axis then rotates about the angular momentum vector on a cone of which the half-angle is referred to as the nutation angle. Precession, on the other hand, is the change in the direction of the angular momentum vector H in inertial space and is always due to external torques.

For a maximum axis spinner, the body spins faster than the spin axis nutates—the exact ratio being determined by the body mass properties. For TDRS–1’s mass properties, this ratio is approximately 4:3, meaning that after four spin periods and three nutation periods, the spacecraft body will return to almost the same position it was in at the beginning of the periods.

The relations between spin ω_s and nutation ω_n is $\omega_n = (\lambda\text{-}1)\ \omega_s,$ where λ is the inertia ratio between the spin axis and the transverse axes, an acceptable approximation of which is:

$$\lambda = Ispin \div [(Ixx + Iyy)\div 2] \qquad \text{Equation (2)}$$

where $I_{zz} = I_{spin}$, I_{xx} and I_{yy} are the inertias about the spin axis and of the transverse axes respectively.

For TDRS–1, the momenta of inertia (neglecting the cross-products of inertia) are:

$$I_{xx} = 6{,}080 \text{ slug-ft}^2 = 8{,}258 \text{ kg-m}^2$$

$$I_{yy} = 3{,}545 \text{ slug-ft}^2 = 4{,}806 \text{ kg-m}^2$$

$$I_{zz} = 8{,}479 \text{ slug-ft}^2 = 11{,}496 \text{ kg-m}^2$$

When Equation (2) is applied using the above momenta of inertia for a spin rate, ω_s = 1 deg/sec, nutation ω_n = 0.76 deg/sec (a nutation period of 8 min was observed on orbit).

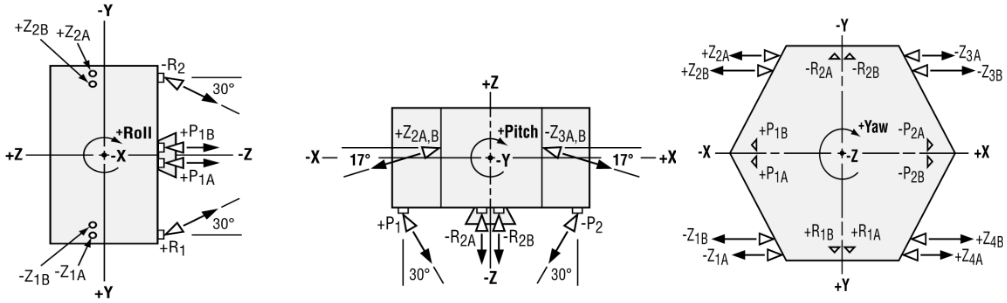

Figure 3. Thruster Orientation with respect to the Control Axes

TDRS–1 thruster orientation with respect to the control axes is shown in Figure 3. Table 1 gives the torque components of the thrusters.

The Z-axis thrusters are used either in coupled pairs or individually for yaw attitude control, or in thrust pairs for Δv. Used in thrust pairs, they ideally would balance the torques generated about the spacecraft pitch, yaw and roll control axes resulting in a pure force through the spacecraft center of mass. Actually, the torque cancellation is not complete, leaving residual torques about two of the axes resulting in perturbations to the spin during the fuel depletion burns.

Table 1. TDRS Thruster parameters (1.0 lb Nominal Thrust Beginning of Life)

Thruster	Torque (ft-lbs)		
	X (Roll)	**Y (Pitch)**	**Z (Yaw)**
$-Z_1$	1.000	0.0905	−3.2835
$+Z_2$	−1.017	0.0805	3.3166
$-Z_3$	−1.031	−0.111	−3.3731
$+Z_4$	1.005	−0.111	3.2867
−P	0.1213	5.0648	0.0701
+P	−0.0584	−5.0481	0.0338
+R	4.5725	−0.0682	0.0394

A residual torque about the Z-axis will cause the spin speed to either decrease or increase. Therefore, adjustments to the spin speed would need to be made during the fuel depletion burn in order to keep the spin speed close to the desired rate.

Spin speed adjustments could be made by off-modulating one of the two Z-axis thrusters resulting in a temporary imbalance of the thruster torques to correct the spin speed. Although the residual torque about roll is relatively small, the residual torque about the pitch is significant and will perturb the orientation of the spin vector. There are two effects of this external torque: nutation and precession.

If an applied external disturbance torque is constant throughout the many rotations, then the spin vector would experience a periodic combination of nutation and precession (the bounds determined by the size of the disturbance torque, the spin rate, and the ratio of the transverse products of inertia with the maximum inertia). Termination of the disturbance torque at the end of the burn will cease the periodic precession motion but will cause the spin vector to start a different (i.e., torque-free) nutation path about the precessed mean orientation.

FUEL DEPLETION BURNS

Pre-Operational Dynamic Simulations

Fuel depletion thruster firings applies external torques causing the spin vector to experience a periodic combination of nutation and precession. At the end of a burn the disturbance torques cease and the spin vector starts a torque-free nutation path about the precessed mean orientation.

To predict the attitude dynamic behavior, a computer model was developed to simulate and display the motion of the rotating rigid body (appendage flexibility and fuel slosh were considered negligible) subject to thruster torques. By integrating the Euler's equations of motion, the angular momentum components in the body frame were determined. It was also deemed useful to transform body components of these and other vector quantities into inertial coordinates.

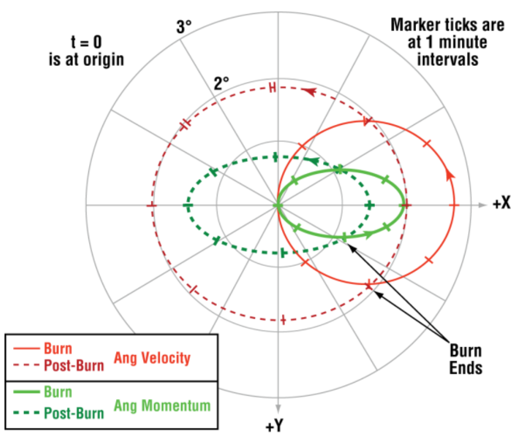

Figure 4a. Angular velocity and Momentum in the Body Frame.

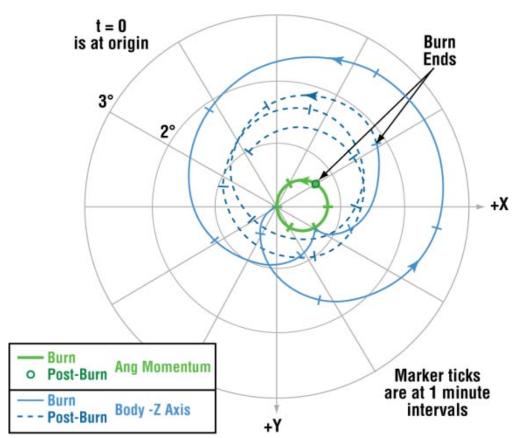

Figure 4b. Angular Momentum and Body Spin Axis in Inertial Coordinates.

Figure 4a shows the typical trajectory of the angular velocity and angular momentum in the body frame over a 600-sec thruster firing. The center of the coordinate system corresponds to the –Z principal axis. Markers indicate 1-min time intervals starting from the origin at t=0. At this time, a constant transverse torque along +Y (simulating the thruster firings) was applied, causing both the angular momentum and angular velocity to leave the origin and orbit counterclockwise from the center of the figure along offset elliptical trajectories that cycle in 8-min intervals (the nutation period). When the torque was removed at 600 sec, both quantities switch to more pronounced elliptical trajectories centered about the origin; this is torque-free body nutation. Had the thrust continued six more minutes in this simulation case, the trajectory would have returned to the origin and the final nutation would have been near zero.

Figure 4b shows the same case as Figure 4a, with angular momentum and the body –Z axis plotted in the inertial coordinate system coinciding with the body coordinates at t=0. Both quantities start at the origin. Note that the circular motion of the angular momentum orbits over a 6-min interval (the spin period). The –Z-axis direction coincides with the angular momentum direction every 8-min (nutation cycle). This is indicated at the points where the cusps of the body Z-axis curve (burn portion) touch the angular momentum path in Figure 4b. Also note that the post-burn angular momentum ceases to precess as shown by the marker on the angular momentum path at burn end.

Fuel depletion Burn Impact on the Disposal Orbit Characteristics

The Flight Dynamics Facility (FDF) at NASA's Goddard Space Flight Center (GSFC) was tasked to evaluate the effect of fuel depletion burns on the orbit apogee/perigee altitude. FDF evaluated the effects of thrust deviation from orbit normal for two burn orbital locations: the ascending and descending nodes. The thrust deviations from orbit normal were applied in each of four principal in-plane directions: in the thrust and opposite thrust directions along the velocity vector and toward nadir and anti-nadir directions. Only the in-plane thrust component was modeled, with its size defined in magnitude and direction deviation from orbit normal. Thrust in the orbit normal direction was neglected as it would only affect the orbit inclination. Off-pointing angles of 5° and 10° and burn durations of one and eight hours were evaluated. Each of the tested directions produced a change in apogee/perigee altitudes. Figure 5 shows the results for a one-hour burn with 10° of angular off-pointing. This worst-case off-pointing direction lowered perigee by almost 20 km. The maximum change to either apogee or perigee was 2 km for one hour of thrusting per degree off orbit normal. The actual in-plane thrust was not sufficient to separate these effects from other assumption-related perturbations.

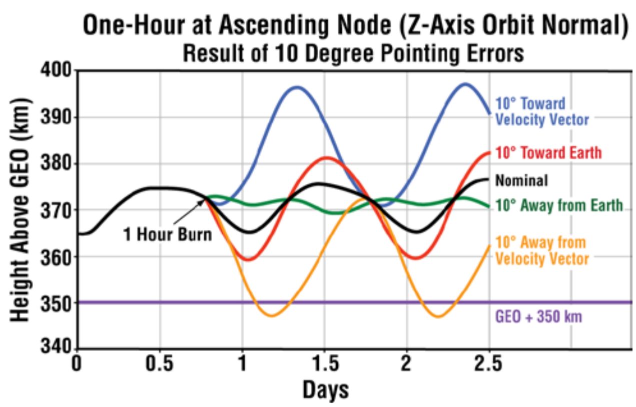

Figure 5. One-hour burn with 10° off-pointing from orbit normal.

Fuel Depletion On-Orbit Results

Ten fuel depletion burns were performed during the campaign, all but one burn being executed near perigee to prevent lowering the orbit. Due to the initial spin axis orientation uncertainty, the first burn was performed at perigee, thus minimizing the risks of lowering the orbit before the spacecraft orientation could be determined.

Table 2 shows the magnitude and orientation of the inertial Δv resulting from the burns. These parameters were obtained as a product of the orbit determination process. Some inertial Δv is observed because the thrusters had a component of their thrust aligned with the Z-axis which was not cancelled by the spin. In addition, spacecraft nutation can result in variation of thruster inertial pointing. As designed, the inertial Δv r ealized during the Z-axis normal spin-mode was well below the approximately 155 m/sec available, given the amount of propellant expended and the thruster's specific impulse. The Δv angle from orbit normal was controlled throughout the burns to less than 9^o offset. This was accomplished by controlling the burn duration (an integer number of nutation and spin periods), the nutation angle, and by precession maneuvers to realign the angular momentum closer to orbit normal. A more complete discussion of these techniques follows.

Table 2. Summary of fuel depletion burns.

Burn no.	DOY	Burn Start Orbit Position (hours/min before perigee)	Duration	Inertial Δv magnitude (m/sec)	Inertial Δv angle from orbit normal (deg)	Inertial Δv angle from sun vector* (deg)	Δapogee (km)	Δperigee (km)
1	167	32m	1h 9m 0s	1.6	1.1	157	1.9	1.7
2	168	Burn centered on descending node	1h 10m 46s	1.7	2.3	143.7	3.6	0.4
3	169	1h 4m	1h 53m 17s	3.4	1.5	122.8	5.1	−0.9
4	170	58.5m	3h 6m 7s	3.7	4.3	136.6	16.8	−2.1
5	172	18m	3h 3m 0s	4.6	8.1	82.4	16	−3.5
6	173	1h 22m	2h 2m 47s	2.9	7.2	81.2	18	−1
7	174	1h 25m	1h 38m 54s	2.2	7.7	90.6	18	−1
8	175	2h 11m	3h 20m 58s	4.1	7.8	92	32	−1.5
9	176	2h 21m	4h 24m	5.2	8.6	104.2	23	−0.8
10	177	3h 41m	1h 46m 58s	2.1	6.5	93	10.5	0.4

**This angle is measured in the orbit plane.*

Post-Burn Nutation Damping: In actual operations, precession and nutation were minimized by timing the fuel depletion maneuvers to end at an integer number of nutation and spin periods. For the TDRS–1 mass properties as previously calculated, the ratio between the spin rate and the nutation rate was approximately 4:3, i.e., a nutation period of 8 minutes to a spin period of 6 minutes. To accomplish this, the fuel depletion burns were planned in increments of 4 spin periods or in multiple increments of 24 minutes. Nevertheless, each burn did end with some small nutation, which was damped before the next burn. The classical nutation damping technique which consists of firing a thruster that provides a torque normal to the momentum vector was used. A pitch thruster (0.8 sec on-time) was selected for use because both polarities were available (versus constraint on using only the positive roll thruster). A + pitch torque (+Y) was applied when the rate about the roll axis (X) was zero.

Simulations found that firing the positive pitch thruster in three equal sets of burn-times—each spaced by one nutation period—were successful in reducing nutation without precessing the angular momentum vector. This technique was utilized at the end of each fuel depletion burn (Figure 6).

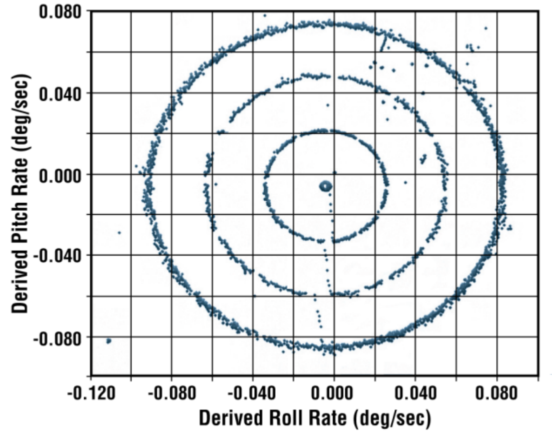

Figure 6. Nutation damping via 3 Pitch thruster firings.

Z-Axis Precession Maneuvers: During the long fuel depletion burns, variations in individual thruster performances and transverse-spin axis torque components caused the angular momentum to precess away from its position at burn start along an irregular path that did not return to the starting position. As the tilt became larger, precessing the Z-axis back toward orbit normal was necessary.

Just as a technique was found that adjusts nutation without affecting precession, what was needed here was the converse. Simulations showed that spacing a pair of pitch thruster firings two spin periods apart—as opposed to one nutation period for nutation control—would result in a desired change in precession while leaving minimal residual nutation. The procedure was to issue thruster firing commands at approximately the time when the positive pitch thruster was opposite the Z-axis tilt. This timing was determined as follows. The effective ΔV direction, a product of the orbit determination activity, gave by inference the Z-axis tilt in inertial space referenced to the Sun. Then, knowing the spin rate, the spatial orientation of the body axes (thus the thruster location) could be determined relative to the Sun crossings as well. While there were no direct observation of the angular momentum precession, Figure 7 shows a cross plot of actual spacecraft derived roll and pitch rates during a precession adjustment and the effectiveness of the technique in leaving the system with very low nutation.

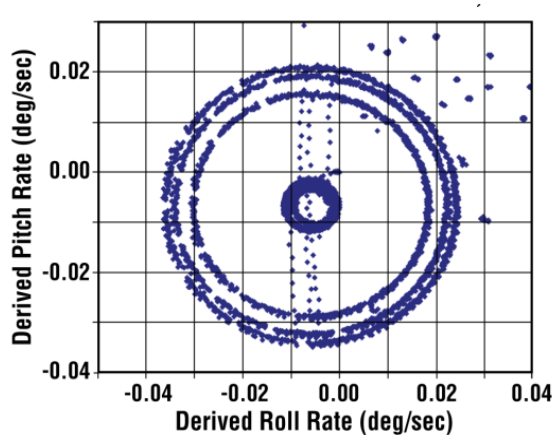

Figure 7. Body rates during precession adjustment.

CONCLUSIONS

Once the fuel depletion was deemed compliant with the Orbital Debris requirements, final equipment passivation commenced. Passivation" or deactivation was driven by the EOM requirement to deactivate all spacecraft equipment that contained or produced mechanical (kinetic), chemical/explosive, RF, or electrical energy. Given the systems interdependency, equipment deactivation had to proceed systematically to prevent irreversibly turning off a component upon which other systems relied such as power converters, heaters, command and telemetry units, etc. Details of the passivation process and the lessons learned applicable to future satellite decommissioning could be found in Reference (2).

Although the satellite ended its life in a configuration which departed substantially from its original design, the TDRS–1 de-orbit was a success. The Z-spin configuration was well suited to support the fuel depletion process. Careful system modeling permitted meeting the de-orbit requirements even with a spacecraft which had lost some of its capabilities. In the final phase, a slower spin rate of 0.5°/sec was commanded to increase the power cycling period from solar arrays to batteries from 3 to 6 minutes and thus support an adequate TT&C (Tracking, Telemetry and Command) link to the very end (simulations had shown that a minimum of 3 minutes were required). Ultimately, after multiple attempts to re-establish communications, TDRS–1's failure to respond meant that a complete decommissioning was achieved.

Requiescat in pace, TDRS–1!

ACKNOWLEDGMENTS

TDRS–1 successful retirement from GEO orbit was conducted by technical experts and managers belonging to many organizations within NASA and its contracting companies. Weeks of analytical studies, simulations, procedure preparation, and operational rehearsals preceded the actual removal from the GEO arc campaign. The contributors and their affiliations are mentioned by names in the companion paper in Reference (2). WSC (White Sands Complex) satellite controllers performed the actual decommissioning with the support of members of the TDRS Project at GSFC (Goddard Space Flight Center) and NGST (Northrop Grumman Space Technology, formerly TRW), the TDRS–1 designer and manufacturer. The FDF (Flight Dynamics Facility) at GSFC evaluated the orbit-raising maneuvers and the effects of the fuel depletion burns on the spin axis orientation and disposal orbit.

REFERENCES

[1] NPR 8715.6: *NASA Procedural Requirements for Limiting Orbital Debris,* May 14, 2009.

[2] "Innovative Approach Enabled the Retirement of TDRS–1," IEEE 2011 Aerospace Conference, pp 1699-0.

[3] NASA Technical Standard, *Process for Limiting Orbital Debris* (NASA–STD–8719.14), September 6, 2007.

[4] Ibid.