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TRAJECTORY DESIGN AND CONTROL FOR THE COMPTON GAMMA RAY OBSERVATORY RE-ENTRY

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Abstract

The Compton Gamma Ray Observatory (CGRO) controlled re-entry operation was successfully conducted in June of 2000. The surviving parts of the spacecraft landed in the Pacific Ocean within the nominal impact target zone. The design of the maneuvers to control the trajectory to accomplish this re-entry presented several challenges. These challenges included the timing and duration of the maneuvers, propellant management, post-maneuver state determination, collision avoidance with other spacecraft, accounting for the break-up of the spacecraft into several pieces with a wide range of ballistic coefficients, and ensuring that the impact footprint would remain within the desired impact target zone in the event of contingencies. This paper presents the initial re-entry trajectory design and traces the evolution of that design into the maneuver sequence used for the re-entry. The paper also discusses the spacecraft systems and operational constraints imposed on the trajectory design and the required modifications to the initial design based on those constraints. Data from the re-entry operation are also presented.

INTRODUCTION

Mission Overview and Re-boosts

The Compton Gamma Ray Observatory (CGRO) was launched from the Space Shuttle Atlantis in April 1991. Shortly after launch, the 'A' side of the propulsion system was shut down due to damage caused by a water hammer effect that occurred when the isolation valves were opened. This left the spacecraft with only two of the four orbit adjust thrusters (OATs) and four of the eight attitude control thrusters (ACTs) available for maneuvers. The orbit decay rate of the spacecraft allowed for completion of the two-year science lifetime requirement; however, the discoveries made by the CGRO during these two years made a compelling argument for continuing the mission. A decision was made to attempt a re-boost with the 'B' side of the propulsion system to an orbit of approximately 450 km in altitude.

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During the first re-boost attempt, a problem with the B2 ACT resulted in early termination of the re-boost. After extensive analysis, the second attempt to re-boost the spacecraft with the 'B' side of the propulsion system was conducted in December 1993 using modified operational procedures. To ensure a power-positive condition on-board during the maneuvers, the re-boost was conducted in two phases. The first phase consisted of ten maneuvers near orbit noon to raise apogee to 450 km. This was followed by a 52-day waiting period to allow the rotation of the line of apsides to bring apogee into alignment with orbit noon, and concluded with a second phase of nine maneuvers, resulting in a final orbit of 454 km x 450 km. A final re-boost, designed to raise the spacecraft altitude to approximately 512 km, was conducted in April 1997. It was also conducted in two phases. The first phase consisted of five maneuvers to raise apogee to 517 km. Once again, this was followed by a 52-day waiting period to allow apogee to migrate into alignment with orbit noon and concluded with a second phase of six maneuvers to raise perigee to 501 km.

In June 1999, the decision was made to begin preliminary re-entry planning. When a gyro failure in December 1999, left the spacecraft with only two gyros, it was decided to begin full up re-entry planning in earnest, targeting the March/April 2000 timeframe for the re-entry operation. Maneuver and trajectory design of the re-entry began in January 2000, and the re-entry operations were conducted in late May/early June 2000 with the spacecraft's final orbit on June 4, 2000.

Pre-Launch Re-entry Plan

Because of the size and composition of the CGRO, a controlled re-entry or retrieval by the Space Transportation System (STS) were considered to be the only viable options for the disposal of the spacecraft at the mission end-of-life. An observatory re-entry plan was developed by TRW, the spacecraft prime contractor, prior to launch. This plan defined the nominal impact target zone (Figure 1), breakup analysis, maneuver scenarios and contingencies. Goddard Space Flight Center's (GSFC) Flight Dynamics group performed additional analysis to verify the TRW plan. This initial re-entry plan assumed initiation from a 350 km circular orbit and consisted of three maneuvers performed with the four OATs to bring the spacecraft to a terminal perigee between 50 km and 70 km. The nominal impact target zone selected was a large open stretch of the Pacific Ocean, southeast of Hawaii. This pre-launch plan served as the baseline from which the final re-entry plan evolved.

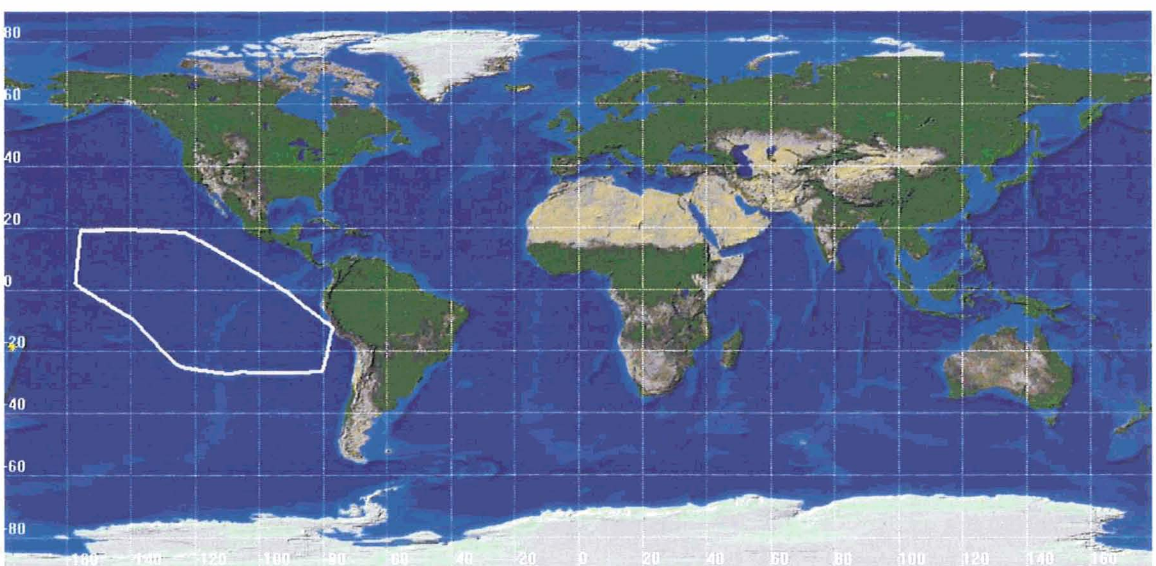

Figure 1 - Original CGRO Nominal Impact Target Zone Defined by TRW

RE-ENTRY TRAJECTORY DESIGN REQUIRMENTS

Impact Zone and Debris Field Requirements

The nominal impact target zone used for the final re-entry planning (Figure 2) was only a slight modification of the original zone (Figure 1) defined in the pre-launch plan. The essential difference was in redefining the zone in terms of the criteria specified in NASA Safety Standard (NSS) 1740.14 for avoidance of debris impact in United States and international territories. The zone is located in the Pacific Ocean, extending south and east from Hawaii to a point just off the South American coast near Lima, Peru. The established criteria for avoidance of debris impact on land was 25 nmi (~46 km) from United States territories and 200 nmi (~370km) from international territories.

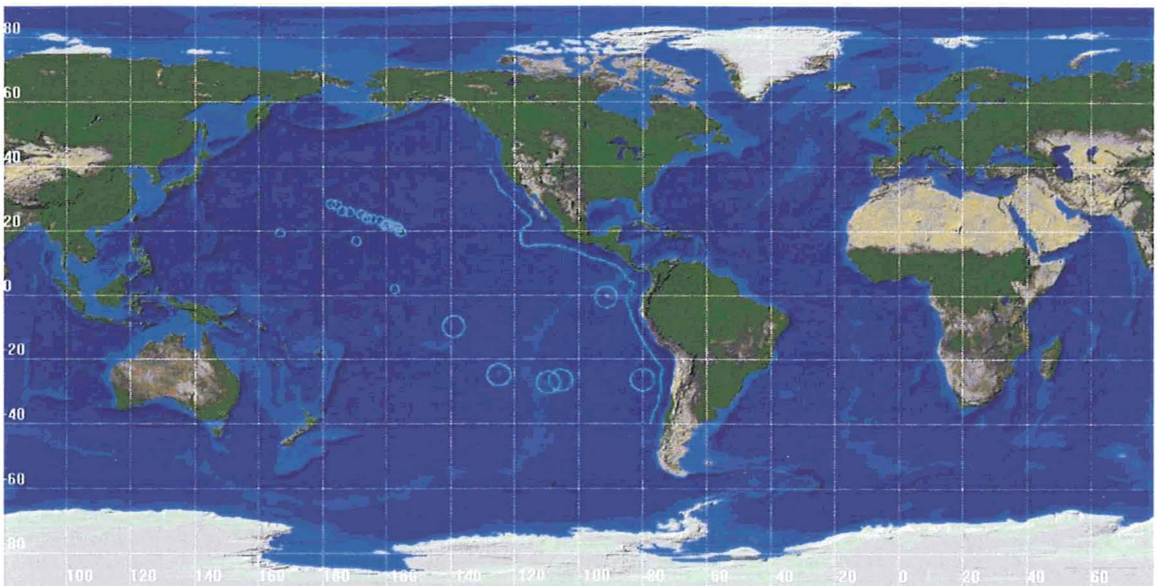

Figure 2 - CGRO Nominal Impact Target Zone – Final Re-entry Plan

In developing the maneuver plan, analysis was performed to characterize the size of the debris footprint. Impact predictions were made based on the debris segments with the largest and smallest ballistic coefficients (β), along with the nominal value for an intact spacecraft (Table 1), and considered deviations in propulsion subsystem performance of up to $\pm$10%.

Table 1 - Ballistic Coefficients (β) Used for CGRO Debris Impact Predictions (m^2/kg)

Low β	0.001517
Nominal β	0.003286
High β	0.409632

The bounding cases for the nominal debris footprint were high β/+10% and low β/-10%. As an added margin of safety against raining debris over South America, a case that consisted of a low β and a minus 10% thrust deviation in combination with a 12-minute delay in the execution of the final maneuver was also considered. The impact prediction analysis was performed using Analytical Graphics' Satellite Tool Kit™ with the High-Precision Orbit Propagator (HPOP) option and a Harris-Priester atmospheric model, along with a predicted Cartesian post-maneuver vector for the final maneuver from the General Maneuver Program (GMAN) as input. The results of that analysis are shown in Figure 3. In addition to this analysis, Johnson Space Center (JSC) engineers performed high-fidelity debris survivability and impact predictions with their Object Re-entry Survival Analysis Tool (ORSAT) software package to verify that the maneuver targeting would satisfy the requirements of NSS 1740.14.

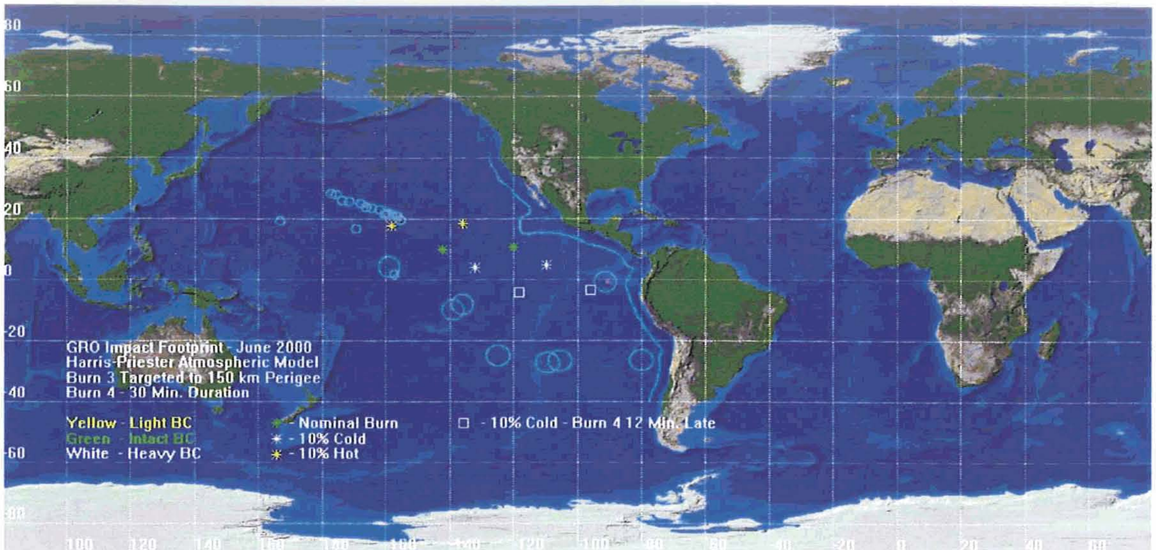

Figure 3 - CGRO Nominal Impact Target Zone with Debris Footprints

Spacecraft Systems and Operational Design Requirements

Several of the spacecraft subsystems imposed requirements on the re-entry trajectory design. The power subsystem required that the Sun vector be within ±30° for the final two maneuvers to ensure a power-positive state during the maneuvers. This requirement combined with the impact zone targeting requirements restricted the timeframe within which the final two maneuvers could be performed to periods when apogee occurred close to orbit noon near the ascending node. This limited the opportunities to perform the final re-entry maneuvers to a 4-day period every 54 days. This requirement also put the maneuver times near local (GSFC) midnight. Figure 4 depicts acceptable conditions for the final two re-entry maneuvers.

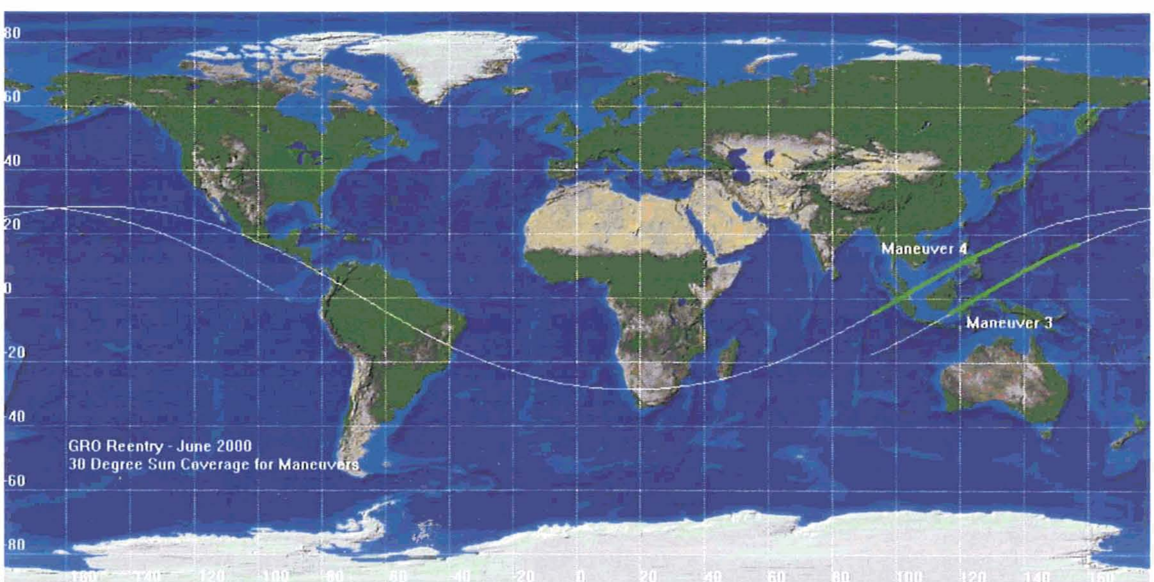

Figure 4 – Sun Coverage for Maneuvers 3 and 4

The thermal and attitude control subsystems required that the minimum perigee altitude prior to the start of the final re-entry maneuver be no lower than 140 km to avoid aerodynamic overheating and ensure that the control system was not overcome by aerodynamic torques. This required the last maneuver to lower the orbit perigee from about 150 km to 50-60 km. The propulsion subsystem restricted the thrusters used for orbit adjust maneuvers and attitude control to those on the 'B' side of the propulsion system. The orbit maneuvers would have to be performed with only two OATs instead of the four OATs called for in the pre-launch re-entry plan. The propulsion subsystem also limited the maximum duration of a maneuver to 30 minutes and it was deemed desirable to avoid having the propellant pressure fall below 689.5 kPa at 17°C (100 psi at 62.6° F).

There were several operational requirements placed on the maneuver design. It was desirable to have all maneuvers of similar duration to promote repeatability of propulsion subsystem performance and more accurate targeting. The length of the final maneuver to reduce perigee altitude from 150km to 50km was the driver for determining burn duration. The post-maneuver perigee altitude for the final maneuver was to be 50 kilometers or less with a corresponding flight path angle of $-1.2°$ or less to prevent skip-out and ensure impact near the first post-maneuver perigee. All maneuvers with the exception of the final two were to be performed at least 24 hours apart to allow for tracking, orbit determination, and post-maneuver reconstruction to evaluate propulsion subsystem performance for targeting of subsequent maneuvers. The final two maneuvers were to be performed on back-to-back orbits to reduce the probability of systems overload, damage, or failure during low perigee passes. This was a major change from the pre-launch re-entry plan, which called for several orbits between the final two maneuvers. Finally, communication with the spacecraft was not an issue in the re-entry trajectory planning since the CGRO had almost continuous view via the Tracking and Data Relay Satellite (TDRS) System.

RE-ENTRY MANEUVER PLANS

Nominal Re-entry Maneuver Plan

A nominal re-entry maneuver plan was developed factoring in the requirements described in the previous section. The plan called for four orbit lowering maneuvers to be performed over a 5 to 6 day period. The first maneuver would lower the orbit perigee from 500 km to 350 km and define the orientation of line of apsides such that when the final two maneuvers were performed, perigee would be positioned over the nominal target zone. The second maneuver would be performed about 24 hours after the first and would lower perigee from 350 km to 250 km. The final two maneuvers would be performed about 48 hours after the second maneuver on consecutive orbits and would take the orbit perigee from 250 km to the terminal perigee of 50 km. All maneuvers would be between 23 to 30 minutes in duration.

Analysis indicated that given nominal propulsion subsystem performance, a maneuver of 24 minutes duration was needed to lower perigee altitude from 150 km to 50 km. The second and third maneuvers could be adjusted based on the observed performance for the previous maneuver; however, the final maneuver would be executed as planned - it would not be possible to properly analyze performance for the third maneuver in the short time period between the third and fourth maneuvers. To compensate for off-nominal performance in the propulsion subsystem, the fourth maneuver was lengthened to 30 minutes duration. Analysis showed that this would serve to compensate for up to a ±20% deviation in effective delta-v performance for the final two maneuvers.

A summary of the nominal re-entry maneuver plan is listed in Table 2. Target dates for execution of the maneuvers were determined based on the power subsystem requirement of having apogee occur near orbit noon. There were two opportunities considered: the first in early April, and the second in early June. In order to allow adequate time for testing and planning, the early June opportunity was selected as the target for the re-entry. Based on the maneuver timing set forth in the nominal plan, achieving the target impact date of June 4 required the maneuver sequence to begin on May 30. Once the maneuver schedule had been defined, a predicted impact footprint for the range of ballistic coefficients was computed for the nominal re-entry maneuver sequence (Figure 5) to verify that the final targeting would satisfy the requirements of NSS 1740.14.

Table 2 - Nominal CGRO Re-entry Maneuver Plan Summary

Maneuver #	Duration	Burn Center	Fuel Used (kg)	Delta-V (m/sec)	Post-burn Perigee (km)
1	23 min.	325° Argument of Latitude	238.74	36.04	350
2	23 min.	Apogee	218.56	33.109	250
3	24 min. 15 sec.	Apogee	201.96	33.313	150
4	30 min.	Apogee	240.77	36.843	30

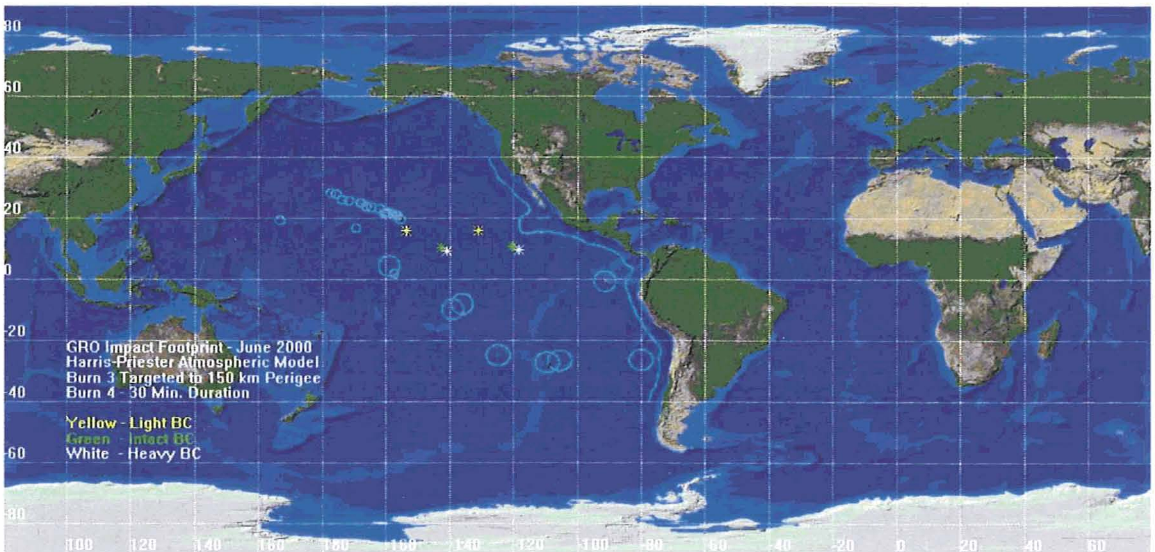

Figure 5 - Nominal Impact Footprints for June 2000 Re-entry

Contingency Plans

As described in the previous section, the nominal maneuver sequence for the re-entry of CGRO consisted of 4 maneuvers. The first maneuver defined the line of apsides and positioned it such that when it was time to perform the final maneuver, perigee was over the north edge of the primary impact target zone in the Pacific for two back-to-back descending pass ground-tracks. To achieve this condition, the first maneuver was centered on an argument of latitude of 325°. This placed perigee high in the Northern Hemisphere such that the rotation of the line of apsides (approximately 12°/day) brought perigee over the target zone in time for the nominal back-to-back maneuver 3-4 sequence.

Based on the nominal targeting, it would take approximately 4 days for perigee to drift through the primary impact target zone as it moved from north to south along the ground-track. Once the first maneuver was executed, the timing of the remainder of the maneuver sequence was set. This required that the fourth maneuver be executed during one of the two opportunities each day in that 4-day period in order to impact in the primary target zone. A delay greater than 4 days would have required consideration of an alternative target zone. Several contingency cases were built into the nominal maneuver plan. Listed below are optional maneuver scenarios based on where in the maneuver sequence the delay may have occurred.

Maneuver 2 Delay

After a nominal first maneuver, the orbit lifetime was estimated to be approximately 260 days. Recall that the apsidal rotation rate was approximately 12°/day, so it took approximately 30 days for a complete rotation. If the remainder of the maneuver sequence could not have been executed within the 4-day window of opportunity over the primary target zone, it would have been delayed for approximately 26 days. At that point, the rotation of the line of apsides would once again have brought perigee to the north edge of the primary target zone. There would have been 8 additional rotation cycles available to complete the remainder of the maneuver sequence over the primary target zone before re-entry by natural orbit decay.

If the nature of the delay was such that waiting for perigee to migrate back over the primary target zone was not a viable option, secondary target areas in the Atlantic, Indian, and Pacific Oceans would have been considered. The final and least attractive option was to rotate the line of apsides to move perigee back over the north side of the primary target zone. This would have involved several maneuvers and would cut substantially into the fuel margin. This option was not explored in detail due to the additional operational complications it would have entailed.

Maneuver 3 Delay

After a nominal second maneuver, the orbit lifetime was estimated to be approximately 80 days. Given the 30-day apsidal rotation cycle described above, if the remainder of the maneuver sequence could not have been executed within the 4-day window of opportunity over the primary target zone, there would have been 2 additional rotation cycles available to do so before re-entry by natural decay.

Once again, if waiting for perigee to migrate back over the primary target zone was not a viable option, the alternatives mentioned in regard to a maneuver 2 delay would have been considered.

Maneuver 4 Delay

After a nominal third maneuver, the orbit lifetime was estimated to be approximately 7 days, with about 5 days of usable lifetime to complete the controlled re-entry. Analysis indicated that at approximately 6 minutes into a nominal maneuver 4, the spacecraft would be committed to re-entry within the next orbit. A maneuver of 17 minutes duration would ensure impact in the primary target zone. Simulations had shown that in the event of a prematurely commanded shutdown, a minimum of 12 minutes was required to reconfigure and restart the maneuver. As a result, maneuver 4 was planned for a nominal duration of 30 minutes in order to accommodate an interruption anywhere within the first 17 minutes of the maneuver in addition to a thruster performance variance of ±20%. Had there actually been an interruption and attempted restart of the maneuver, it is likely that the thrusters would have been commanded to burn to depletion to increase the probability that the debris would impact in the primary target zone.

If maneuver 4 could not have been executed within the 4-day window of opportunity over the primary target zone, the alternatives mentioned in regard to a maneuver 2 delay would have been considered.

Collision Avoidance Analysis

The descent of CGRO during the controlled re-entry maneuver sequence resulted in the spacecraft crossing the orbital paths of numerous critical and non-critical space assets owned by NASA, as well as those of other space-faring entities. Concern regarding the possible conjunction of CGRO with these assets prompted the inclusion of a capability to adjust the maneuver plans as late as 8 hours prior to each of the first three maneuvers in order to avoid potential collisions with a select subset of these assets. The most notable were the Space Shuttle (STS), the International Space Station (ISS), and Mir.

JSC had primary responsibility for performing the conjunction analysis, and the Flight Dynamics Facility at GSFC provided nominal maneuver plans to JSC as input. JSC performed the analysis for STS, ISS, and Mir, and forwarded the maneuver plans to NORAD for additional analysis against other assets. If the analysis predicted a probability greater than 1 in 29,000 of conjunction with any of these assets, JSC minutes, or a wave-off for that day. If the probability of conjunction were lower, the nominal maneuver plan would be used.

No adjustments to the nominal maneuver plans were required during the re-entry operation.

Other Planning Considerations

Several other items were considered during the planning of the re-entry maneuvers. These included notifications to mariners and aviators, verification of the maneuvers plans prior to executing them, and updating the predicted impact footprint after execution of each maneuver.

CGRO RE-ENTRY OPERATIONS

Re-entry Operations Summary

CGRO re-entry operations were begun on May 28, 2000, with the execution of several engineering test burns. These burns were performed to test the operation of the various propulsion subsystem components. All of the engineering burns were executed successfully.

The first maneuver in the re-entry sequence was performed on May 31, 2000 with a start time of 01:51:05 GMT. The maneuver was 23 minutes 6.2 seconds in duration and lowered perigee to 364 km.

The second maneuver was performed on June 1, 2000 with a start time of 02:36:54 GMT. The maneuver was 26 minutes in duration and lowered perigee to 241 km.

Following the second maneuver there was an approximately 3-day wait until the execution of the third maneuver. This time was used to refine plans for the final two maneuvers and to update the predicted impact footprint.

The third maneuver was performed on June 4, 2000 with a start time 03:56:02 GMT. The maneuver was 21 minutes and 39 seconds in duration and lowered perigee to 148 km.

One orbit later, with a start time 05:22:24 GMT, the final maneuver in the re-entry sequence was executed. The maneuver was 30 minutes in duration and lowered perigee to approximately 30 km. The estimated time of impact was approximately 06:18:50 GMT. Spacecraft impact in the target zone was confirmed by United States Air Force (USAF) personnel on-board an observation aircraft that was dispatched to monitor the terminal phase.

Table 3 summarizes the results of the re-entry maneuver sequence.

Table 3 - Re-entry Maneuver Summary

Maneuver #	Start Time yymmdd.hhmmss	Burn Duration	Fuel Used (kg)	Duty Cycle	Thruster Efficiency	Modeled Cal. Factor	Post-Burn Perigee (km)
1	000531.015106	23min. 6sec.	235.81	0.71	1.018	0.723	363.34
2	000601.023654	26min.	246.07	0.735	1.02	0.7488	240.55
3	000604.035602	21min. 39sec.	192.62	0.76	1.014	0.7725	146.14
4	000604.052224	30min.	247.32	0.786	1.014	0.7974	26.69

Orbit Determination Accuracy

Orbit determination was performed after each maneuver in support of post-maneuver reconstruction to verify propulsion subsystem performance. The post-maneuver 1 and 2 solutions were used to adjust the subsequent maneuvers. No adjustments were made to the fourth maneuver after executing the third maneuver since the volume of tracking data that could be collected between the maneuvers was insufficient to accurately evaluate performance for the third maneuver. Table 4 summarizes the accuracy of the post-maneuver orbit determination performed during the re-entry operation. Since there was limited tracking data available between the third and fourth maneuvers, a predicted ephemeris based on nominal performance for the third and fourth maneuvers was used for comparison in place of a definitive ephemeris. Accuracy of the post-maneuver 3 and 4 ephemerides is difficult to determine because of the limited amount of tracking data. Short-arc solutions will match the a-priori vector if the tracking data is reasonably close.

Table 4 -CGRO Post–Maneuver Orbit Determination Accuracy

Maneuver #	Compare	Arc Length	Definitive Overlap	Burn + 24 hrs	Computed TSF
1	Burn + 1.5hr /Burn + 2 day	1.5 hrs.	8 m	637 m	-0.0152
	Burn +3hr/Burn +2 day	3.0 hrs.	9.9 m	1.67 km	
2	Burn +1hr/Burn+1 day	1.0 hrs.	134 m	13.8 km	0.0053
	Burn +2.5hr/burn +1 day	2.5 hrs.	38 m	7.2 km	
3	Burn +30min./Nom. Burn 3	0.5 hrs.	13.6 km	n/a	0.0185
4	Burn + 18 min./Nom. Burn 3/4	0.3 hrs.	2.1 km	n/a	0.0392

Attitude Control for Re-entry Maneuvers

During the orbit maneuvers, the spacecraft's attitude control system maintained approximate alignment of the thrust vector with the anti-velocity vector. This was accomplished by rotating the spacecraft around the pitch (+Y) axis at a rate of 0.64°/sec (1 revolution per orbit) using the ACTs. After the maneuver was completed, the spacecraft was placed in a power positive attitude that kept the +X axis toward the Sun. Once the spacecraft was aligned for the third maneuver, the attitude was controlled with the ACTs through the final perigee and re-entry. Figure 6 depicts the attitude reorientation sequence for a maneuver.

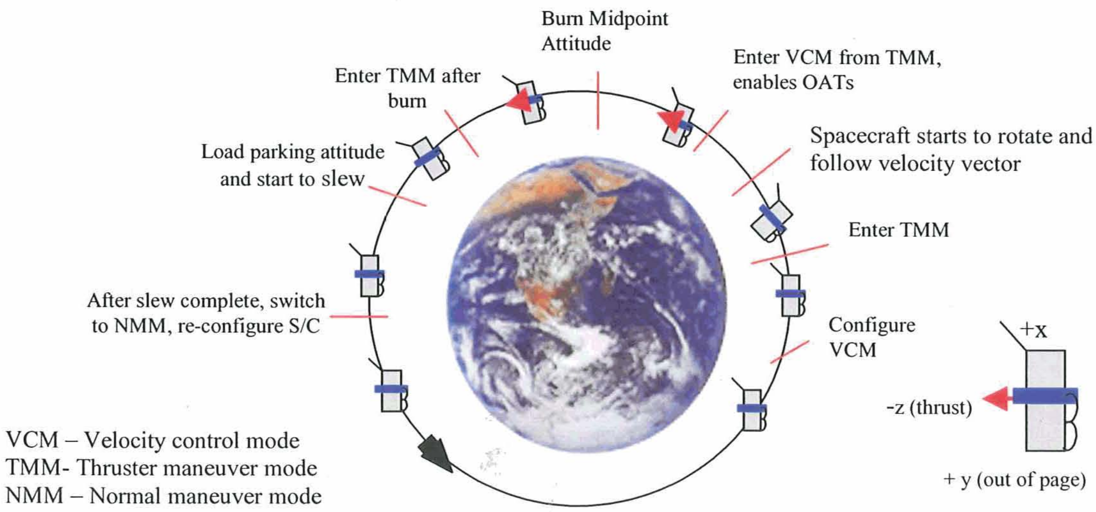

Figure 6 - Attitude Reorientation Sequence for Re-entry Maneuver

After completion of the final maneuver, attitude control was maintained for about 17 minutes, until 0609 GMT. Spacecraft telemetry and tracking were received until approximately 0611 GMT after which communication with the spacecraft was lost. Figures 7 through 11 are plots of the roll, pitch and yaw rates in the spacecraft body frame from the final perigee to the point where telemetry was lost.

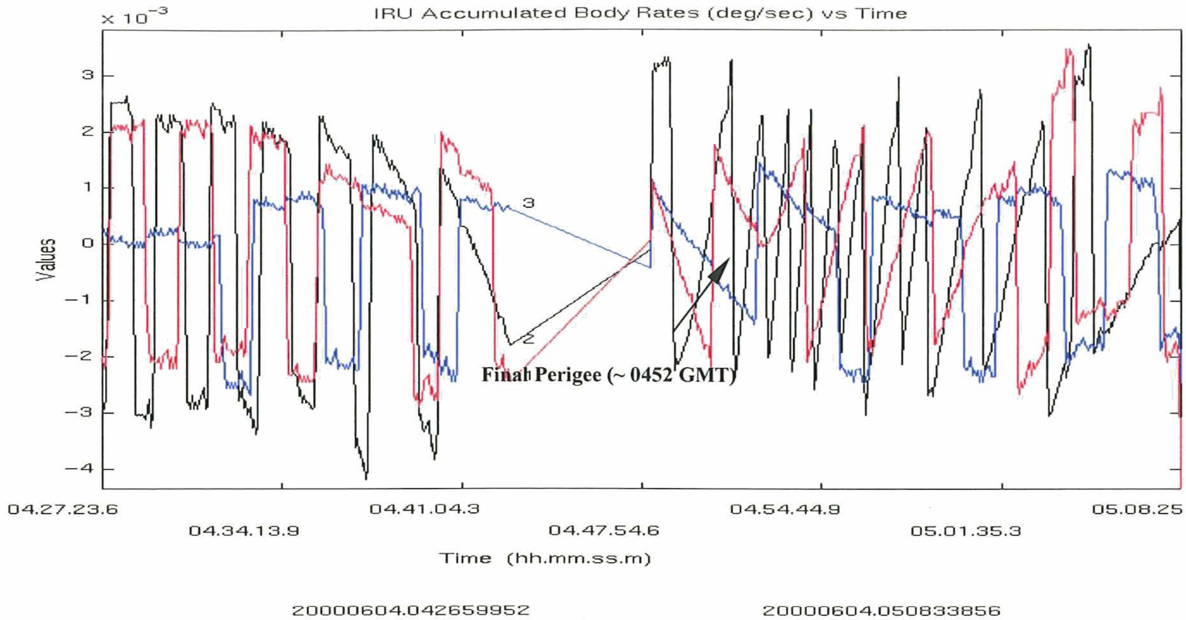

Figure 7 - CGRO Body Rates During Final Perigee Pass

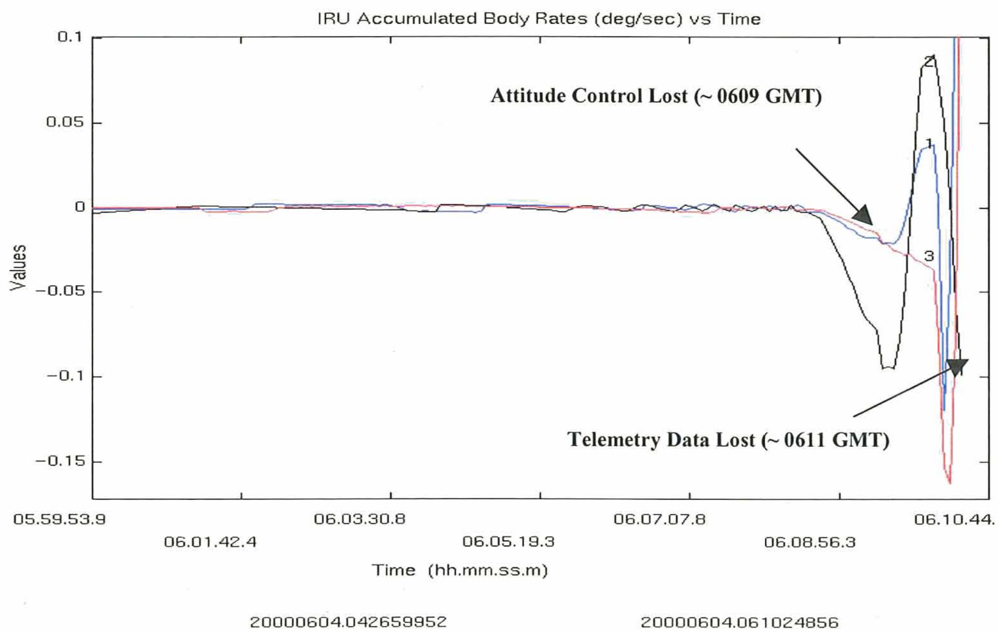

Figure 8 - CGRO Body Rates During Re-entry

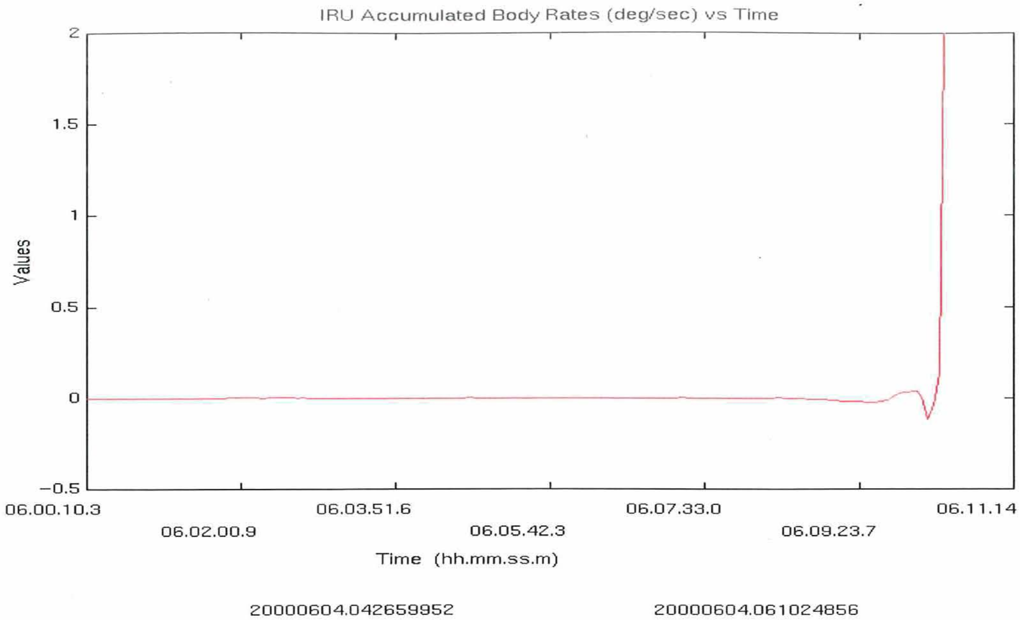

Figure 9 - CGRO Roll Rate During Re-entry

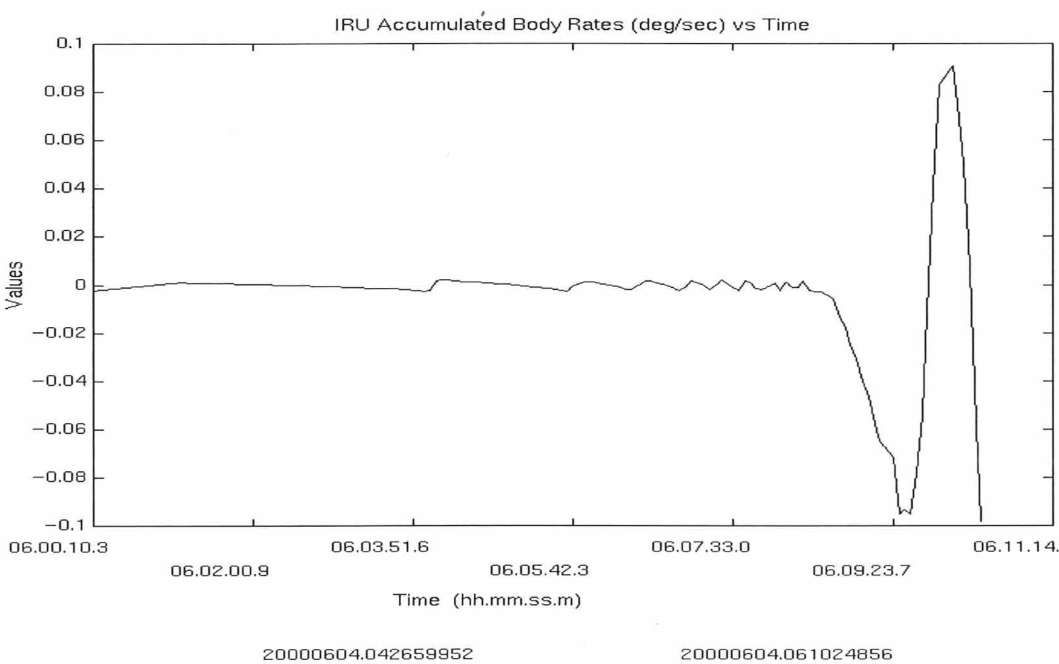

Figure 10 - CGRO Pitch Rate During Re-entry

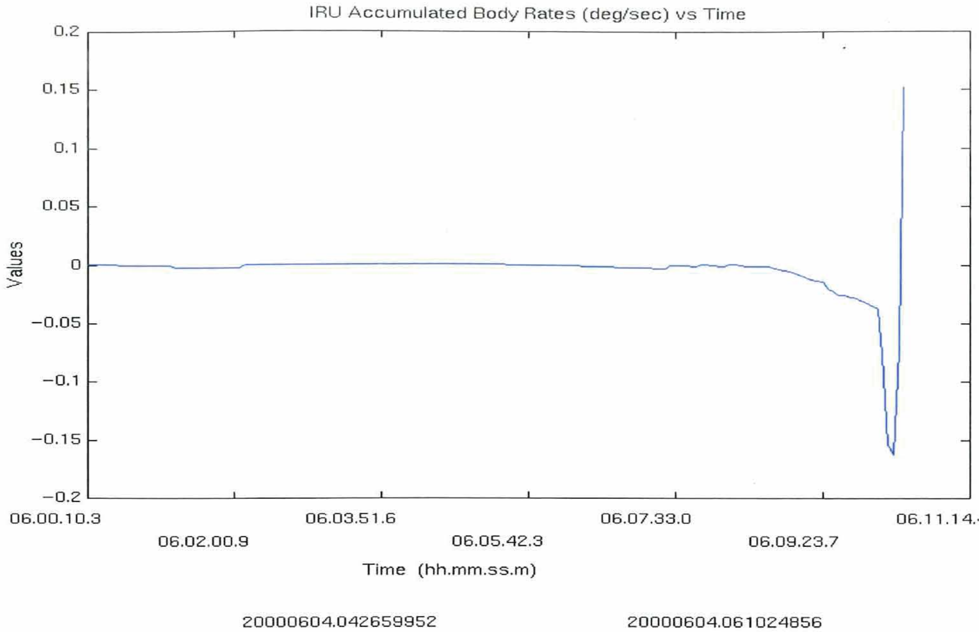

Figure 11 - CGRO Yaw Rate During Re-entry

Conclusion

After a highly successful mission that spanned over 9 years, the CGRO spacecraft was successfully de-orbited in a controlled re-entry on June 4, 2000 – a first for a NASA spacecraft not designed to survive re-entry. The design and control of the final orbit trajectory required incorporating multiple constraints and contingencies into an integrated and operationally feasible maneuver plan. Future operations of this nature will benefit from the experience gained through this initial effort.

Acknowledgements

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