

**FLIGHT RESULTS OF THE CHANDRA X-RAY OBSERVATORY
INERTIAL UPPER STAGE SPACE MISSION**

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ABSTRACT

Under contract to NASA, a specially configured version of the Boeing developed Inertial Upper Stage (IUS) booster was provided by Boeing to deliver NASA's 1.5 billion dollar Chandra X-Ray Observatory satellite into a highly elliptical transfer orbit from a Shuttle provided circular park orbit. Subsequently, the final orbit of the Chandra satellite was to be achieved using the Chandra Intergral Propulsion System (IPS) through a series of IPS burns. On 23 July 1999 the Shuttle Columbia (STS-93) was launched with the IUS/Chandra stack in the Shuttle payload bay. Unfortunately, the Shuttle Orbiter was unexpectedly inserted into an off-nominal park orbit due to a Shuttle propulsion anomaly occurring during ascent. Following the IUS/Chandra on-orbit deployment from the Shuttle, at seven hours from liftoff, the flight proven IUS GN&C system successfully injected Chandra into the targeted transfer orbit, in spite of the off-nominal park orbit. This paper describes the IUS GN&C system, discusses the specific IUS GN&C mission data load development, analyses and testing for the Chandra mission, and concludes with a summary of flight results for the IUS part of the Chandra mission.

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INTRODUCTION

The IUS vehicle was developed under contract to the Air Force Space and Missile Systems Center. The vehicle is designed to take payloads to geosynchronous or other orbits after being boosted to low earth orbit. The Space Shuttle or the Titan IV launch vehicle are used to place the IUS and its payload into low earth orbit. The IUS is a two-stage solid rocket motor (SRM) vehicle which uses a hydrazine gas reaction control system (RCS) for attitude control and velocity trim burns. The vehicle can place about 5000 pounds into geosynchronous orbit. IUS payloads placed in orbit include the Air Force's Defense Support Program satellites and Defense Satellite Communication Systems satellites and NASA's Tracking and Data Relay Satellites. In addition, the Galileo and Magellan planetary probes and the Ulysses solar probe were successfully placed into targeted orbits using the IUS. Several flights are manifested for the future but will use the new Flight Controller (FC) to perform the GN&C and mission sequencing computational functions. Characteristics of the FC have been previously described.^{1,2} The Chandra mission was to be the last planned IUS flight using the flight proven Redundant Inertial Measurement Unit (RIMU), built by the Hamilton Standard division of United Technologies, and a pair of redundant Delco built flight computers. The FC, in a single package, replaces all three of these older avionics components thus avoiding avionics obsolescence issues and reducing system costs for future missions.

Boeing development of the IUS flight design, flight software and mission data load for the Chandra mission began on 1 July 1994 under special contract to the George C. Marshal Space Flight Center (NASA). This work included development of the IUS GN&C computer data load necessary to handle the special requirements of the Chandra mission. These requirements included providing IUS GN&C functions to support an IUS payload much heavier than ever handled by IUS. Whereas the typical IUS payload has been about 5000 pounds, the Chandra spacecraft with IUS adapter weighed 12495 pounds. This and other mission factors were accommodated through a GN&C mission data load design, analysis and testing process described below. Since the IUS GN&C system used for the Chandra mission, its development and associated analyses have been described before,³⁻¹⁵ only a brief description of this system is provided here. The focus instead is providing a summary of the IUS GN&C data load design process, associated analysis tasks and testing for the Chandra mission and then providing actual flight results.

The Chandra X-Ray Observatory (Figure 1), formerly known as the Advanced X-Ray Astrophysics Facility (AXAF), is the third of NASA's grand space based telescopes. The Hubble Space Telescope (HST), launched by Shuttle Discovery in April of 1990, and the Compton Gamma Ray Observatory (CGRO), launched by the Shuttle Atlantis in April of 1991 preceded it. The IUS/Chandra combined payload was launched to low earth orbit by the Shuttle Columbia on 23 July 1999. The combined payload in the Shuttle payload bay is shown in Figure 2.

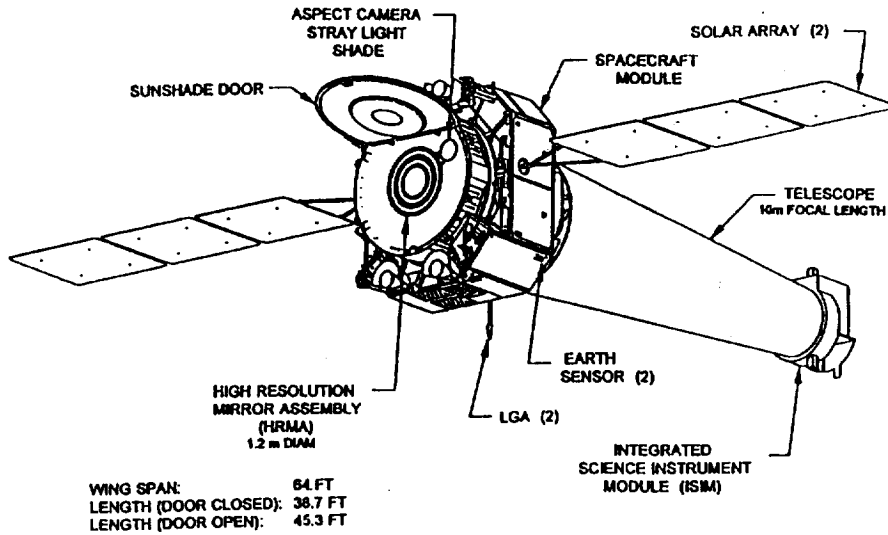


Figure 1 Chandara X-Ray Observatory



Figure 2 Inertial Upper Stage/Chandra

MISSION DESCRIPTION AND CHRONOLOGY

The purpose of the IUS/Chandra mission was to deliver the Chandra spacecraft from low earth orbit into a highly elliptical earth orbit through the use of both IUS SRMs and the Chandra integral propulsion system (IPS). Figure 3 shows a pictorial representation of the mission. The IUS mission began, however, with prelaunch IUS power up at about 20 hours before the opening of the launch window at 4 hours 35 minutes Universal Time Coordinated (UTC) on 20 July 1999. The health and the sequencing of the IUS system were generally monitored via IUS telemetry, from power up to IUS battery depletion, at a combination of the Onizuka Air Station (OAS) facility at Sunnyvale, California; the Eastern Launch Site (ELS) in Florida; and the Boeing Kent Space Center in Kent, Washington. During the prelaunch period, the IUS navigation subsystem went through a normal two-hour self-calibration process to refine compensation of certain flight critical RIMU errors. This calibration period was followed by a continuous and normal gyrocompass process to maintain navigation alignment knowledge up to IUS go-inertial, nominally occurring at 11 minutes prior to liftoff. However, at about 7 seconds from liftoff, a Shuttle sensor indicated dangerous levels of hydrogen in the Orbiter aft compartment and the launch was aborted. Later analysis determined that the reading was erroneous. The second launch attempt occurred on 22 July 1999, but was also scrubbed due to weather concerns at KSC. The third launch attempt, on 23 July 1999, culminated with liftoff occurring at 4 hours 30 minutes 59.9 seconds UTC.

During the ascent a premature Shuttle main engine shutdown occurred resulting in a park orbit of 154 by 145 nautical miles instead of the targeted park orbit of 153 nautical miles circular. However, this off-nominal park orbit still met all constraints for payload operations and no additional orbit adjustments were requested or required. A real-time run of the IUS guidance Avionics Transfer Orbit Mission Simulator (ATOMS), developed by Boeing, confirmed the IUS on-board guidance algorithms would converge with minimal impact to meeting transfer orbit objectives. Use of a previously prepared and tested guidance data load overlay - commanded by the Orbiter crew, if needed, to minimize guidance error in the advent of an off nominal park orbit - was also deemed unnecessary.

At 4 hours 8 minutes mission elapsed time from liftoff (MET) a planned state vector (position and velocity) update of the IUS navigation subsystem was performed by the Orbiter crew. This was accomplished by transferring the Orbiter state vector, itself updated from uplinked ground track data, to the IUS using the normal hardware interface

between the Orbiter and IUS. A second planned state vector transfer to the IUS occurred at 6 hours 26 minutes MET, in the same manner as the first transfer.

After the IUS/Chandra stack was elevated in the supporting tilt table, the IUS/Chandra was deployed from the Orbiter at 7 hours 16 minutes MET, as planned. Ten minutes after deployment the IUS Reaction Control System (RCS) was automatically enabled by command of the IUS primary computer. Immediately after this normal event, the IUS began the RCS attitude maneuver to the programmed thermal control orientation. Normal guidance initialization was verified to occur at 8 hours 2 minutes MET. Subsequent events could not be monitored due to an unexpected loss of telemetry coverage across the two IUS SRM burns (SRM-1 and SRM-2). It had been planned to obtain IUS telemetry during this period using a Deployable System (DS) stationed near Sao Paulo, Brazil and a P3 aircraft positioned to receive the portion of the burns not covered by the DS. However, both of these systems experienced difficulty receiving the IUS signal. This resulted in the majority of the IUS burns being performed without ground monitoring. When data did finally return via the Hawaii tracking station, it was verified that the burns had been normal as were the subsequent events through to the automatically sequenced IUS/Chandra separation.

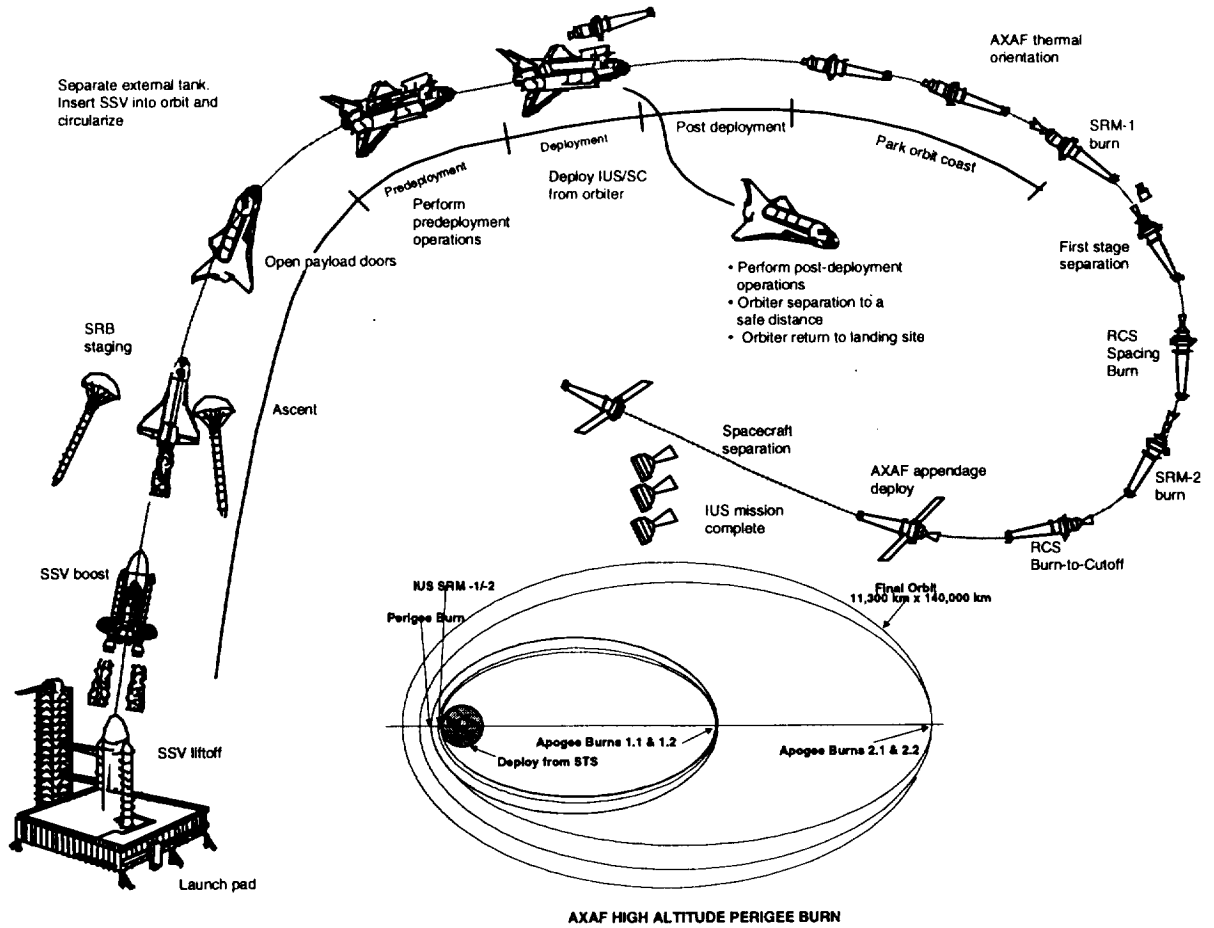


Figure 3 IUS/Chandra Mission Overview

IUS GN&C SYSTEM DESCRIPTION

The IUS GN&C architecture includes a pair of redundant avionics strings, A & B as shown in Figure 4, and a redundancy management technique which provides at least single fault tolerance to improve the probability of mission success. The measurement of vehicle dynamics (sensed acceleration and angular rate) to support the GN&C functions are provided by the RIMU. Required GN&C computations are performed in parallel in each of the two Delco flight computers each using the same measured vehicle dynamics from the RIMU. Only one of these computers is in control at any time but each is continually performing a self-check and communicating with the other. If the side in control detects a serious failure, then it declares itself not OK. If the other side is still OK, then it will take control. Each computer performs, in addition to GN&C, all the other necessary functions to perform the mission. These include mission sequencing, redundancy management and telemetry and command processing.

Each avionics string includes a signal conditioning unit (SCU), as shown in Figure 4, which translates computer generated commands to operate RCS thrusters and command SRM ignition as the mission proceeds. Also, the airborne support equipment (ASE) provides the hardware to pass signals to and from the Orbiter. State vector data to update the IUS navigation system on-orbit, for example, ordinarily comes from the Orbiter computers through the ASE to the IUS.

Guidance System Description

The IUS guidance system provides the ignition times and burn directions for the combination of SRM burns to place the IUS payload in the desired mission orbit. This information is relayed to the mission sequencing, communications, and attitude control functions for implementation of the desired burn sequence. The guidance function is performed by a technique called Gamma Guidance which solves a two-point boundary value problem in ordinary differential equations.^{7,15} One boundary is the current IUS state vector and the second is the desired mission orbit. Gamma Guidance is an explicit, adaptive guidance scheme mechanized as an onboard, autonomous, real-time algorithm. The algorithm iteratively converges to a solution of the transfer orbit problem based on the park orbit, mission orbit, and an initial guess specified in the MDL. Once initiated in flight, the algorithm requires

no input from the ground. The algorithm performs all guidance functions including onboard targeting, midcourse updates, and closed loop guidance for all SRM and RCS burns.

Gamma Guidance is a general algorithm capable of guiding the IUS to many different mission orbits dependent only upon the energy available. This adaptability is achieved via menus.⁷ Each menu provides a generic list of available choices in a particular category (e.g., order and number of propulsive elements, constraints, control variables and optimization variables). This unique implementation allows the user flexibility to adapt the software to the particular mission. The selected strategy is entered into the software via the mission data load (MDL). This eliminates the need to redesign and completely retest the entire onboard software for new missions.

Navigation System Description

The navigation system hardware consists of the RIMU, the two Delco flight computers and the interface between these components. All navigation computations are performed redundantly in the two flight computers during the mission. The RIMU is itself inherently redundant in that it provides outputs from five single degree-of-freedom floated rate integrating gyros and five single axis pendulous accelerometers, to both flight computers. The input axes in each of these sensor sets are uniformly arranged in space. These strapdown sensors are independently powered through 3 separate channels, as shown in Figure 5, so that only two channels are required to maintain the 3-axis navigation function in either flight computer. This ensures that at least three gyro and three accelerometer outputs are available to one or both computers if a single RIMU power supply fails. In addition, a failure detection and isolation (FDI) algorithm and built in test equipment (BITE) code, operating in each flight computer, ensure that at least the first gyro failure and first accelerometer failure will be detected and computationally isolated before adversely affecting the mission. The FDI algorithm uses sensor parity vectors to gauge disagreement among the gyro or accelerometer sensor sets. By comparing these parity vectors to parity thresholds in the flight computers, the means is provided to reliably detect and isolate single sensor failures which are deemed significant.^{3,4,6,8-14}

Control System Description

The flight control function includes two elements; the coast attitude control system (CACCS) and the thrust vector control (TVC) system as illustrated in Figure 6.^{5,13} The CACCS maintains required vehicle orientation and required angular rates through all coast phases of a mission, provides roll control through powered flight and provides vernier translational burns, as required. The control authority for the CACCS is provided by a 12 thruster dual redundant reaction control system (RCS). During the coast flight phase, the phase plane control logic^{5,13} used by the CACCS maintains the attitude and rate in each axis. Control is accomplished by firing the appropriate RCS thrusters to maintain attitude and attitude rates within predetermined deadbands. The phase plane switching curves for the control logic are determined directly from the IUS/payload vehicle mass properties. Mission sequencing driven changes to commanded attitude, attitude and rate tolerances, and maneuver rates are inputs to the phase plane logic. Attitude and rate errors result in thruster firings until the desired attitude and rates are attained.

The TVC system is utilized for pitch and yaw control when IUS guidance has commanded an SRM burn. It uses two electromechanical actuators to deflect the SRM nozzle to obtain control authority on the two axes.

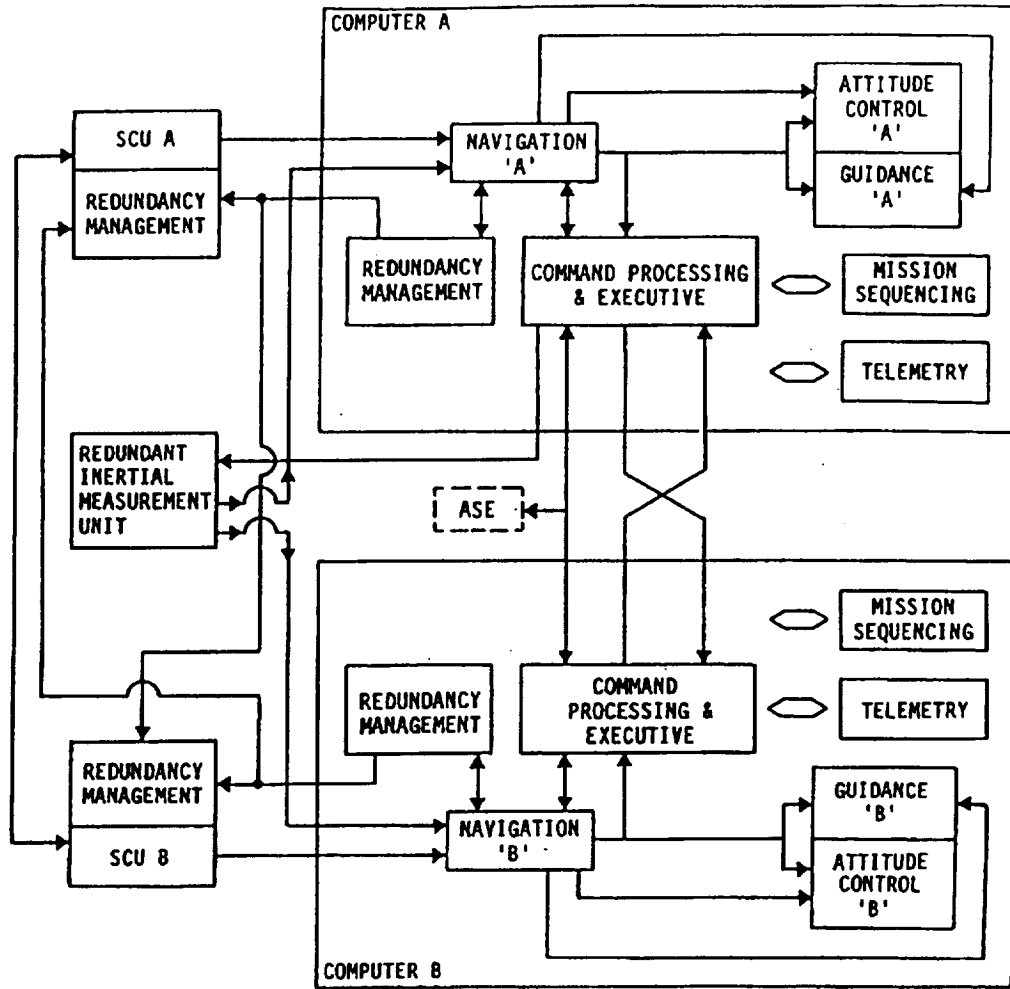


Figure 4 GN&C Processing Flow

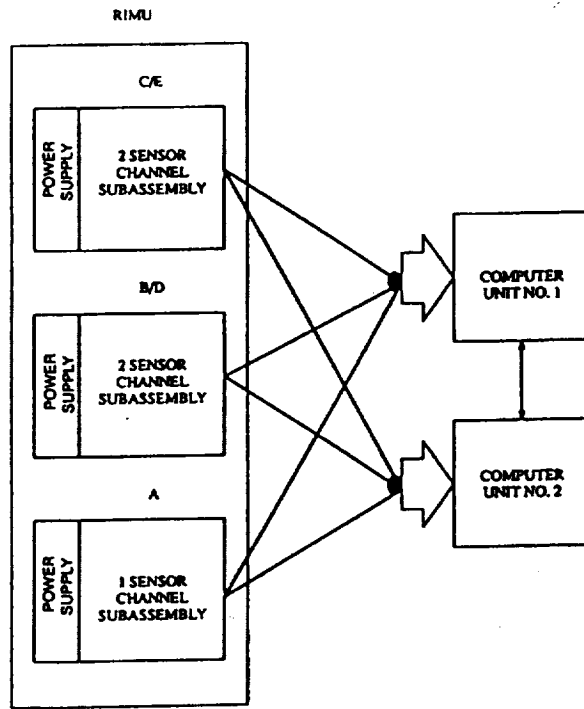


Figure 5 Redundant Inertial Navigation System

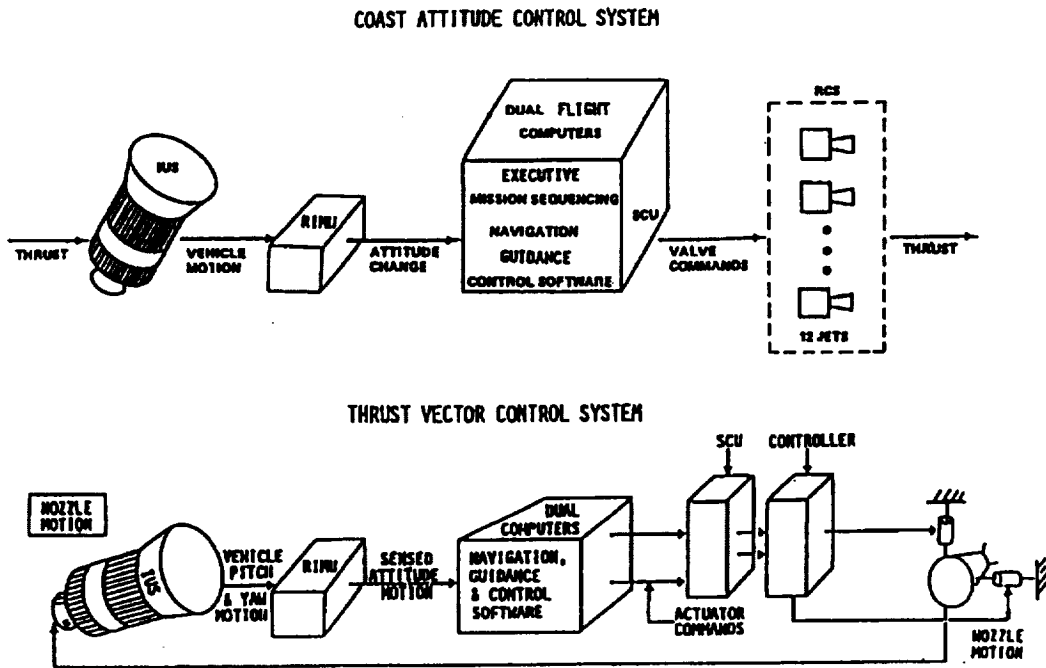


Figure 6 Flight Control System

IUS FLIGHT COMPUTER GN&C DATA LOAD DESIGN

Development of the IUS flight design and software for the Chandra mission began in July 1994. The ultimate goal of the process was to produce a validated On-board Digital Data Load (ODDL) ready to support the IUS/Chandra flight. The ODDL consists of two components; the operational flight software (OFS) and the mission data load (MDL). This ODDL when loaded into the IUS flight computers, along with the mission specific RIMU overlay, supports prelaunch and predeployment operations and autonomously controls all aspects of the IUS flight after deployment from the Orbiter. The software development was conducted in 3 phases. The initial flight feasibility phase produced a preliminary flight design as a proof of concept showing that Chandra mission requirements could be met. This included development of preliminary software design requirements and supporting GN&C analyses. Completed in early 1995, these data in turn supported development of the IUS/Chandra Interface Control Document (ICD) and follow on subsystem design analyses.

The second software development phase, which began in December 1996, was the mission/MDL design phase. During this phase, the flight design was completed, software design and validation test requirements were developed and preliminary flight software and MDL were produced. Each ICD requirement was carefully translated to software design requirements. Each software design requirement then had a corresponding test requirement. Each test requirement was validated either by standalone subsystem analysis or during fully integrated system level flight software testing. The entire range of perceived uncertainties and vehicle characteristics were modeled to assure compliance for any off-nominal condition. Various options were pre-programmed and validated, to cover anticipated variations that could not be determined prior to flight; such as launch date, launch pad, and actual park orbit altitude.

During the third and final phase, which began in January 1998, the flight ODDL was tuned to reflect final Chandra requirements. This final ODDL was then validated through testing and delivered sufficiently early to support the launch in July 1999. Great expense and delay was avoided by having validated options which were easily selected by simple memory shift configuration commands. For example, a change in the launch pad following the start of

the final software design was non-impacting because of the availability of the Shift commands. Also, numerous launch delays were handled with a small amount of analysis and selection of the appropriate pre-built options.

The MDL design process for IUS GN&C began with establishing mission design requirements and then translating those requirements into GN&C MDL design requirements. Accordingly, it was recognized in the early phases of the IUS/Chandra mission design that to achieve the Chandra orbital objectives in Table 1 a combination of the two IUS SRM burns and Chandra IPS burns were required. Moreover, to minimize required Chandra energy (i.e. maximize IPS propellant margin) and achieve the highest possible transfer orbit apogee, it was essential to perform the two SRM burns as close together as possible. This was a departure from the more typical IUS geosynchronous mission which performs one SRM burn in park orbit, coasts for about 5 hours in a geosynchronous transfer orbit, and then performs the second SRM burn to circularize in an equatorial or near equatorial orbit.

Table 1 Chandra Operational Mission Orbit

Target Orbital Parameter	Desired Value
Apogee altitude	140,000 km
Perigee altitude	10,000 km minimum
Inclination	28.45 degrees
Right ascension of ascending node (RAAN)	171 degrees to 219 degrees
Argument of perigee	270 degrees

Note: target orbit values are in True of Date, Earth Centered Coordinates.

After the GN&C MDL design was complete and standalone tested, it was integrated with other subsystem MDL (e.g. mission sequencing) and tested together using two six degree-of-freedom simulations. The first of these simulations is the Simulation of Trajectories and Related Subsystems (STARS) which checkouts the combined MDL over the simulated mission using a scientific FORTRAN version of the OFS. The

other simulation, the Verification and Validation Simulator (V&V), tests the actual flight computer load (ODDL) running in a computer comparable to the flight computer. For these simulations, nominal missions and special mission scenarios were run to verify that the OFS and MDL complied with mission requirements and agreed with subsystem standalone simulation results.

Guidance Design

The guidance MDL was designed to satisfy the mission orbit objectives within the mission/configuration space while maximizing the Chandra IPS propellant margin. The mission/configuration space for which the nominal MDL was designed included the specification of the launch date dependent Right Ascension of Ascending Node (RAAN) and argument of perigee. Other variables included were the SRM propellant loads, RCS propellant loading, and IUS and Chandra weight variations. The guidance MDL was designed and verified with the standalone guidance six degree-of-freedom simulation tools; Avionics Transfer Orbit Mission Simulator (ATOMS) and the Monte-Carlo driver for ATOMS (MCATOMS).

The baseline configuration for the mission consisted of an off-loaded first stage SRM and an on-loaded second stage SRM and a spacecraft weight of 12466.1 pounds. On-load refers to an additional propellant load exceeding that of a standard SRM. The baseline configuration and the performance range about this baseline were within the capability of the nominal MDL design.

To organize the guidance progression, the IUS mission is divided into arcs. Each arc represents either a coast (non-propulsive) or a burn (SRM or RCS) portion of the trajectory. A brief description of the arc definition for the Chandra mission is given in Table 2. The guidance strategy is determined by selecting appropriate propulsive elements, constraints, control variables, and optimization variables from available menus for each arc to satisfy the mission objectives.

Table 2 Mission Arc Descriptions

Arc Number	Description
1	Park orbit coast to SRM1 ignition
2	SRM1 burn
3	Coast prior to SRM2 ignition, including spacing burn
4	SRM2 burn
5	Coast prior to RCS burn to cutoff
6	RCS burn to cutoff
7	S/C separation, CCAM burn/ burn to depletion

Arc numbers 1 and 2 in Table 2 are the coast phase from IUS deployment to SRM 1 ignition and the SRM1 burn portion of the trajectory, respectively. Arc number 3 is the coast between the IUS SRM burns and includes the IUS staging event. As previously mentioned, this coast period needed to be minimized in order to meet the mission goal of maximizing Chandra IPS propellant margin. A limiting factor in reducing the coast period is the SRM1 outgassing occurring after SRM burnout which tends to move the two IUS stages back together after separation. To mitigate this effect an RCS spacing burn was added to the arc to ensure adequate physical separation between the stages at the time of SRM2 ignition. This added RCS burn was designed to occur shortly after the IUS staging event and provide a stage 2 velocity change of 4 feet/second. Analysis indicated this would provide the minimum desired separation of 25 feet between the two stages at SRM2 ignition for the minimum selected coast interval. With the added RCS spacing burn the coast between SRM1 and SRM2 was shortened to only 150 seconds.

Arc numbers 4 and 5 are the SRM2 propulsive phase and a short coast following this burn, respectively.

A second RCS burn was designed for arc 6. This guidance directed burn was scheduled to occur 35 seconds after SRM2 burnout and was designed to expend unused RCS propellant not needed to complete the mission. This RCS burn, in addition, was designed to increase the transfer orbit apogee as much as possible using the real-time estimated available RCS propellant.

Arc number 7 is the final IUS coast phase and included the IUS/Chandra separation event and a special final RCS burn. The special RCS burn was included to provide the combined functions of a collision/contamination avoidance

maneuver (CCAM) and RCS burn-to-depletion (BTD). The CCAM/BTD was designed to preclude physical recontact and to minimize contamination of Chandra by IUS SRM outgassing or RCS thrusting. The CCAM/BTD burn was scheduled by the mission sequencing function using a fixed attitude specified in the MDL and was not part of the guidance MDL design.

Navigation Design

No special changes to the IUS data load were required for the IUS/Chandra mission to support the IUS navigation function. However, the routine navigation data load changes associated with every IUS mission were required. For example, to fly the specific RIMU assigned to the Chandra mission, a RIMU overlay file is created. This file is loaded into the flight computers after the ODDL is loaded during the launch countdown. The overlay file contains RIMU factory calibration information for error compensation during prelaunch and flight operations. Over 700 RIMU parameter values are represented in the overlay. To ensure reasonable performance the multiple day testing of the RIMU necessary to create the overlay is performed within 6 months of launch. Other minor but important changes include changing date sensitive parameters and adjusting the flight gravity model coefficients to optimize its performance for the anticipated flight trajectory. The RIMU overlay is validated through special testing in the Boeing Inertial Guidance Laboratory (IGL) located in Kent, Washington. In this testing the RIMU is mounted on a three-axis test fixture and operated through programmed flight type maneuvers to verify expected navigation performance.

Flight Control Design

Attitude control analyses and MDL development were performed for both coast and powered flight phases of the IUS/Chandra mission. The attitude control MDL was designed to accommodate the range of SRM performance and IUS/Chandra mass properties, all possible variations in RCS temperature and pressure, and all contingencies (including any single RCS thruster failed off) without in-flight modification.

Coast Control Design

The MDL design for IUS coast control during the mission was based on nominal IUS and Chandra vehicle characteristics. This design was then tested against variations in those vehicle characteristics, along with certain RCS failures, to determine overall control system effectiveness. Performance and compliance with requirements was verified using a FORTRAN simulation of the coast control system called Three Degree-of-freedom Attitude Control Simulation (TDACS).

Specific IUS maneuvers and attitudes were designed to accommodate IUS and unique Chandra requirements. These included thermal attitudes to protect sensitive Chandra sensors from the sun while orienting other portions of the vehicle to maintain a certain amount of vehicle warming.

Addition rate filtering was designed for activation following Chandra solar array deployment. Very low frequency (0.25 Hz minimum) structural bending modes were filtered with an existing Finite Impulse Response digital rate filter. Coefficients were designed which represented a 30 stage, 0.2 Hz stop band, 0.3 Hz roll-off frequency filter. This allowed for efficient attitude control in response to rigid body rates while isolating the control system from structural oscillations.

Powered Flight Control Design

Powered flight analyses were performed for the IUS/Chandra mission to verify performance of the TVC MDL design. Analyses performed included assessment of TVC stability margins with the effect of the Chandra propellant tank slosh modeled. Nominal and worst case SRM thrust data effects were examined also. Frequency domain TVC stability analysis was performed using the EASY5 system emulation computer program. Results of the stability analysis showed that standard TVC stability requirements were satisfied for the range of propulsion characteristics and mass properties identified for the IUS/Chandra mission. Control loop natural frequency was carefully selected to move the spacecraft propellant tank slosh frequencies away from the gain and phase margin frequencies. This effectively set the controls response so as not to drive the slosh modes.

FLIGHT RESULTS

Mission Performance

The Shuttle Columbia was inserted into a lower than nominal park orbit due to a Shuttle engine hydrogen leak which caused a premature cutoff of the Shuttle main engines during ascent. As a consequence, the IUS transfer orbit apogee altitude was lowered by 536 km because the targeted perigee altitude did not match the actual park orbit altitude.

Based on available IUS telemetry the SRM and RCS burn parameters were well within expected ranges during the mission. The data indicated the performance for both SRM burns were lower than nominally expected however. The lower SRM-1 performance was consistent with either a higher vehicle weight of 72 pounds or a lower motor ISP by 0.23% (-1.4 sigma). The lower SRM-2 performance was consistent with either a higher vehicle weight of 33 pounds or a smaller motor ISP by 0.19% (-1.1 sigma). The RCS burn to cutoff, occurring shortly after the SRM-2 burn, was successful in maximizing the IUS transfer orbit apogee altitude, as planned, while reserving sufficient RCS propellant for the AXAF solar array deployment and Chandra separation events.

Guidance System Performance

Analysis of available flight data indicated the guidance system functioned correctly given the off-nominal park orbit and the subsequent SRM dispersions. This was confirmed by running a simulation of the flight conditions using ATOMS program. The results of the simulation were then compared with flight data to check the orbit-to-orbit transfer selection, guidance update occurrences, convergence behavior, steering misalignment correction (SMC) response, commanded pointing vectors, and RCS vernier burn operations. No problems were found. The guidance system selected the initial set of SRM-1 and SRM-2 ignition times and burn angles properly to achieve the orbit transfer from the actual park orbit to the desired orbit. Further, when the SRM dispersions occurred the guidance system responded correctly to give a final orbit that met mission requirements within the allocated budgets.

Navigation System Performance

The Navigation system performed nominally throughout the mission. Two RIMU self-calibration sequences leading up to launch were performed due to the aborted launch attempt on 20 July and the subsequently observed trends of navigation azimuth error and accelerometer parity following the first self-calibration. These self-calibrations were performed on 19 July and 21 July. Both were nominal in sequencing and results, producing calibration update values well within tolerances. Entry into the navigation flight mode (go-inertial) prior to actual launch on 23 July was also nominal.

During the mission (pre-launch power-on to IUS battery depletion), no RIMU reconfigurations due to FDI or BITE occurred. FDI parity magnitudes appeared reasonable throughout the mission. FDI threshold to parity vector ratios remained above 5.1 for the gyros and above 5.3 for the accelerometers throughout the flight (just prior to liftoff to battery depletion). All of the threshold to parity ratios remained well above the minimum desired ratio of 3.0, suggesting good sensor performance throughout the mission.

Control System Performance

Attitude control during the mission was successful and met all of the mission requirements. The amount of RCS propellant consumed for attitude control was well within that allocated. The RCS propellant consumption during the flight, where IUS telemetry was available, was within 1% of that predicted. The values were compared to nominal values predicted before flight as part of the design analysis work. Stable attitude control was also maintained following solar array deployment. Angular rate filtering isolated the rigid body rates for the control system and no unnecessary thruster firings were commanded in response to Chandra structural oscillations. Each of the attitudes were maintained within required tolerances.

The TVC system provided stable control throughout the first 60 seconds of the SRM-1 burn. Following the ignition transient, normal limit cycling was established and maintained throughout this part of the burn. No further TVC

data was available due to loss of telemetry coverage. The observed flight performance was typical compared to prior TVC performances.

Mission Accuracy

Subsequent to IUS injection of the Chandra spacecraft into its transfer orbit, Chandra was tracked by the Chandra Operations Control Center in Cambridge, Massachusetts. This tracking occurred prior to any significant activity of the Chandra spacecraft. The tracking solution orbital values are shown in Table 4 in comparison to the IUS telemetered navigation solution, the expected nominal transfer orbit result and the requirement. From these whole values IUS subsystem and total orbital errors were determined. These are also shown in the table. In addition, the total errors are compared to estimated 0.9973 fractile bounds ("3 sigma") for the total errors and compared to the error requirement. As can be seen the total system errors were well within the estimated statistical bounds and well within the required error limits. Estimated total injection error bounds were estimated from a combination of analyses for guidance/control/performance effects and navigation effects. Guidance, control and performance effects were analyzed with the MCATOMS simulation program. The MCATOMS analysis modeled random effects arising from SRM dispersions and weight uncertainty. The Navigation effects were analyzed using a covariance analysis tool called Generalized Navigation Instrument Error Analysis (GENIE). The GENIE computer program models all significant RIMU accelerometer and gyro error uncertainties expected to exist from on-pad self calibration and estimates their navigation error effect through the mission. For all errors, distribution free statistical techniques were applied to determine orbital element error bounds. The Total "3-sigma" row of Table 4 is a combination of all the estimated GN&C and performance effects.

Figure 9 shows the IUS accuracy for the Chandra mission relative to sixteen other successful IUS missions. Here the system accuracy for each mission is represented as the ratio of the measured injection error over the injection error requirement for each. The figure shows the IUS performance for the Chandra mission is among the best of all the successful IUS missions relative to mission requirements.

Table 3 IUS/Chandra Transfer Orbit Injection Errors

	Apogee Alt. (km)	Perigee Alt. (km)	Inclination (deg)	RAAN (deg)	Arg. of Perigee (deg)
Values					
Nominal	73164	329.91	28.45	196.263	270
Telemetry	72016	329.97	28.4503	196.258	270.018
Tracking	72065	329.44	28.4516	196.334	269.982
Requirement	>58220	>281	28.45	171-236.5	270
Errors					
Perf/G&C Errors	-1148	0.06	0.0003	-0.005	0.018
Nav Errors	49	-0.53	0.0013	0.076	-0.036
Total Errors	-1099	-0.47	0.0016	0.071	-0.018
Total "3 sigma"	2000	4	0.017	0.21	0.31
Requirement	NA	NA	+/- 1	+/- 1	+/- 1.6

Note: Nominal apogee altitude was reduced by 536 km due to a lower than nominal park orbit

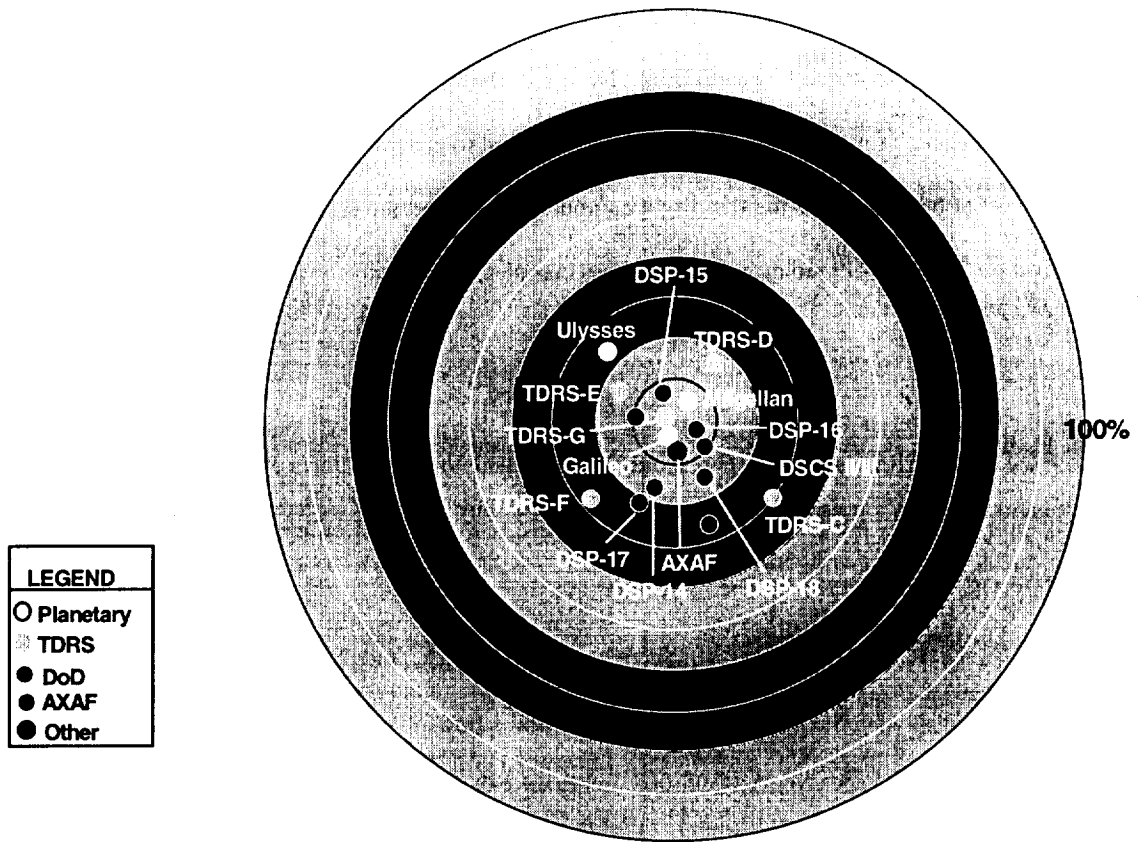


Figure 9 IUS Injection Error Versus Requirements

CONCLUSIONS

A specially configured IUS vehicle was provided to inject the Chandra X-Ray Observatory into a highly elliptical transfer orbit from a Shuttle provided low earth orbit. Design of the IUS GN&C data load was accomplished and tested to meet all associated requirements of the mission. This led to generation of a validated Operational Digital Data Load which could be loaded into the IUS flight computers to support the IUS/Chandra flight. Flight results of the mission flown on 23 July 1999 indicate this effort was completely successful.

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