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Mission Planning for the CHANDRA X-ray Observatory

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Abstract

The CHANDRA X-ray observatory started life as the Advanced X-ray Astrophysics Facility (AXAF) but was renamed CHANDRA in December of 1998 at the conclusion of a nationwide contest by NASA to name the new observatory. The name Chandra honors the Nobel Prize winning astrophysicist S. Chandrasekar who taught astrophysics at the University of Chicago for more than 50 years, following graduate studies at Cambridge University in England. The observatory has been under construction for a decade under the management of the Observatory Projects Office at the Marshall Space Flight Center; the same office that oversaw the construction of the Hubble Space Telescope and the Compton Gamma Ray Observatory. This observatory is a member of NASA's great observatory series of missions of which Hubble and Compton are members.

The scientific purpose of the new observatory is to do astronomical research in the x-ray portion of the electromagnetic spectrum (0.1 - 10.0 keV). It does both high resolution spatial imaging (0.5") and moderate to high resolution spectroscopy. It consists of an x-ray telescope, two scientific instruments; the AXAF CCD Imaging Spectrometer (ACIS) and the High Resolution Camera (HRC) and a spacecraft module to provide power, thermal control, pointing and attitude control and data handling functions. Uplinked commands to the observatory for mission control and downlinked science data from the observatory are all through the JPL DSN network. The original launch date of CHANDRA was August 1998 but due to hardware problems encountered during test and checkout, it was delayed. As of this writing, launch of CHANDRA is scheduled for July 1999.

This paper describes the mission planning that was conducted at MSFC to design the orbit and launch window that would permit the new observatory to function properly within its constraints and resources for at least 5 years and maybe 10 years without any reservicing (which is impossible for the orbit that it is in). This mission planning also addressed the orbital transfer sequence required to take the observatory from its initial parking orbit to the final operating orbit. This included performance optimization and tracking coverage analysis.

Since the scientific observing program operates in the x-ray portion of the electromagnetic spectrum and the Van Allan trapped radiation belts around the Earth can produce considerable background noise in this portion of the spectrum, it was required by the science to get the CHANDRA orbit high enough in altitude to get out of this background radiation noise. For mission planning, this minimum altitude, above which science can be done, was 60,000 km (or a radial distance of about 10 earth radii).

The launch vehicle chosen to put the observatory in its operating orbit was the Space Shuttle with an upper stage (an IUS) attached to the CHANDRA to boost it from its Shuttle parking orbit to a highly elliptical orbit. The capability of the IUS is insufficient to obtain an orbit high enough to meet the desires of the science so an additional liquid propulsion system (the IPS) was built into the spacecraft module to provide additional boost capability. The final operating orbit achieved, after a series of transfer orbits, is an elliptical orbit with a perigee of about 10,000 km altitude and an apogee of about 140,000 km altitude. This gives an orbital period of just over 64 hours

(2.7 days) with about 80% of that time at an altitude above the 60,000 km required for the scientific observations.

Orbits of this magnitude are highly perturbed by the Lunar and Solar gravitational fields and so these disturbances must be accurately accounted for in the long-term integrations required for the mission planning activities. The integration methods themselves must be stable and accurate over long duration time intervals; 10 years in this case. These accurate integrations are required to determine the long-term evolution of the orbit, i.e., to ensure that the orbit evolves in an acceptable manner and that auxiliary events such as Earth and Lunar eclipses are accurately predicted. These external gravitational fields can cause large oscillations in the orbital parameters so initial values of the parameters must be chosen such that the orbit evolves in an acceptable manner. Once the observatory is placed in its final orbit there can be no further orbital adjustments over the life of the mission. The observatory from then on has only attitude control, not orbital adjust capability.

The eclipse events are important because of the limited amount of time that the observatory can function on batteries alone. The battery limitation is two hours and the eclipse durations in some orbits can exceed four hours. Those orbits must be avoided. Thus, the initial orbit must be chosen such that eclipse events of this magnitude do not occur during the expected 10 year mission life. It is also required that the perigee altitude of the orbit not dip much below its starting value over the life of the mission. This was desired by the scientists in order to stay out of some of the most intense Van Allen radiation environment.

Two integration methods were evaluated; one was a 7th order Runge-Kutta-Fehlberg method with step size control, integrating an Encke formulation of the equations of motion. The second was an Adams-Bashforth, Adams-Moulton predictor-corrector method integrating a Cowell formulation of the equations of motion. The predictor-corrector method was ultimately chosen as the slightly faster and more accurate of the two methods. The forces integrated included the first three zonal harmonics of the Earth's gravitational field and the solar and lunar gravitational fields with the solar and lunar positions computed from analytical models. After extended efforts to get results to compare with integrations by other codes, which included a careful effort to make all physical constants match exactly, a close comparison with other codes was finally achieved. Some final slight variances were attributed to slightly different ephemerides of the Sun and Moon that are used by the different codes. Atmospheric drag and solar radiation pressure were found to be insignificant contributors to the long-term evolution of the orbit and thus ignored in the integrations.

The eclipses of the Sun by the Earth were calculated by an analytical technique; that of finding the intersection points of an ellipse with the surface of a right circular cone. The eclipses of the Sun by the Moon were found by a more tedious point-by-point determination of the angular separation between the Sun and Moon as seen from the position of the CHANDRA in its orbit about the Earth. Earth eclipses were found to occur in regular semiannual eclipse seasons. Lunar eclipses were random events occurring on average about once per year and were almost always partial eclipses.

Ten year integrations over extended sets of initial orbital conditions revealed a limited range of initial values of argument-of-perigee (ω) and right ascension of the ascending node (Ω) that would produce acceptable long term behavior of the orbit (perigee not dipping too low) and that would also avoid long duration Earth eclipse events (less than two hours). Interestingly enough, these ranges of values of ω and Ω seemed to be more or less independent of launch date. The range of acceptable values of Ω determine the daily launch window for any given day of the year.

The long-duration Lunar eclipses were not controlled (or avoided) by the choice of the initial values of ω and Ω but by slight changes in the initial perigee altitude which, in turn, caused slight period changes and thus phasing changes between the positions of the CHANDRA and of the Sun and Moon.

Another issue addressed was the chance of collision with existing satellites. CHANDRA passes through the equatorial plane twice per orbit and also through the 12-hour GPS orbital shell twice per orbit. This gives the potential of collisions with existing satellites in these regions. Poisson statistics were used to estimate collision probabilities and they were found to be acceptably low; less than one chance in a million for the geosynchronous satellites over the 10-year mission and only about one chance in one-hundred million for the GPS satellites in ten years.

Once in final orbit, the observatory is to undergo a 40-day checkout period of all of its systems and subsystems by the engineering team that oversaw the construction and ground tests and checkouts. Once it is determined that all systems are functioning normally, the observatory will be turned over to the science operating team, located at Cambridge, MA, which will then plan and execute the science observing program over the next 5-10 year period. It will be operated like any large observing facility with guest astronomers proposing and executing observing programs based on the merit of their proposals as judged by their peers. CHANDRA should be able to remain operational until all of its on-board hydrazine is depleted at which time attitude control of the observatory may be lost.

Brief History of AXAF (CHANDRA)

The Advanced X-ray Astrophysics Facility (AXAF), recently renamed CHANDRA in honor of the late Indian astrophysicist S. Chandrasekhar, was conceived in the late 1970's and early 1980's (1) as a long-lived orbiting national observatory for X-ray astronomy. It was to be one of NASA's Great Observatory series of missions of which the Hubble Space Telescope (HST) and the Compton Gamma Ray Observatory (GRO) are members. Phase A (conceptual design) studies were completed in 1978 and Phase B (detailed design) studies were completed in 1985. 'New Start' funding was provided in 1988 and TRW was selected as the contractor to build the spacecraft bus and integrate the science instruments. The management oversight of the fabrication and construction of the spacecraft bus, the optics and optical bench and the integration of the science instruments has been provided by the Observatory Projects Office at the Marshall Space Flight Center (MSFC) which also performed the same function for both Hubble and Compton.

As initially conceived, AXAF was to be a single mission to do both high resolution spatial imaging and moderate to high resolution spectroscopy. In 1992, the mission was broken into two smaller missions: AXAF-I for imaging and AXAF-S for spectroscopy. In 1993 Congress terminated the AXAF-S mission for budgetary reasons. The AXAF-I survived. Subsequently, the AXAF-I mission became known simply as the AXAF mission. After the spacecraft construction was completed in 1998, a national contest was held by NASA to rename the spacecraft before launch. From thousands of entries the name CHANDRA was chosen, in honor of the late Nobel Prize winning Indian astrophysicist S. Chandrasekhar who taught astrophysics at the University of Chicago for more than 50 years. The original schedule was for launch of the AXAF (CHANDRA) to have occured in late August, 1998 but due to unforseen problems which occured during integration, test and checkout of the spacecraft, the launch was delayed until July 1999.

The CHANDRA X-ray Observatory

The total spacecraft system, which provides the support structure and environment necessary for the telescope and the science instruments to function as an observatory, consists of the spacecraft module, the telescope and the Integrated Science Instrument Module (ISIM). The ISIM contains the two science instruments, the AXAF CCD Imaging Spectrometer (ACIS) and the High Resolution Camera (HRC). This total system is depicted in Figure 1.

The telescope consists of four concentric pairs of grazing incidence mirrors fabricated from glass manfactured by Schott Glaswerke of Germany and built into the telescope mirrors by Hughes Danbury Optical Systems. They are coated with iridium to provide high reflection efficiency for x-rays. The assembly and alignment of the mirror elements was done by Eastman Kodak. Testing of the mirrors was accomplished in the X-ray Calibration Facility at the MSFC.

The science instruments were integrated into the ISIM at Ball Aerospace before being shipped to TRW for integration into the AXAF observatory. The spacecraft module, built by TRW, consists of many parts and functions. It contains the power system with its solar panels and batteries, a thermal control system, the pointing control and attitude determination system (PCAD) for executing attitude maneuvers and holding attitudes, the Integral Propulsion System (IPS) for orbital maneuvering and the antennas and command and data management system for receiving uplinked commands, storing data and downlinking data (through the JPL DSN network).

Attitude maneuvers and attitude holds are accomplished using six sets of reaction wheel~~s~~ assemblies (RWA) arranged in a hexagonal configuration. Normal operation will be to use all six RWAs. In case of a failure, the opposing RWA will also be shut down and the remaining four will be used. Excess momentum in the wheels can be unloaded by the MUPS (Momentum Unloading Propulsion System) which consists of four MUPS assemblies, each with primary and secondary reaction jets powered by hydrazine. There is enough hydrazine on-board for a minimum 5-year mission and probably a 10-year mission. Pointing can be held with an accuracy of **(the relative pointing stability of the line-of-sight with respect to the commanded direction should <25 arsec (rms) half-cone angle over 95% of all 10 second periods)** arc seconds. The angular resolution of the telescope is approximately 0.5 arc seconds which is eight times better than its predecessor, the Einstein Observatory (1978 - 1981).

The initial weight in the final operating orbit is approximately 4600 kg. The length of the observatory is approximately 12 meters (39.5 ft), the span of the solar panels is 19.5 meters (64.0 ft) and they generate approximately 2,300 watts of power under full sun (Note: they were sized to provide 2,100 watts at 5 years). The 3 Ni-Cd batteries are 40 amp-hour batteries which can power the observatory for up to 2-hours under the condition that only 2 of the batteries are operational and the depth of discharge will not exceed 80% which is approximately ½ power for normal operations (Note: eclipse power requirement cannot exceed 64 amp-hour).

The body coordinate system is shown in Figure 1. The X body axis is along the long axis of the telescope pointing in the direction of the target to be observed. The Y body axis is along the axis of the solar panels and the solar panels can rotate +,- 90° about this axis. The Z body axis is in the direction opposite to the active side of the solar panels.

Hardware Restrictions

There are not many hardware restrictions imposed on the mission but there are a few. Because of the battery limitations, eclipses of the Sun by the Earth or by the Moon cannot exceed 2 hours. The X-axis (the long axis of the telescope) cannot be pointed within 45 degrees of the Sun or Moon which means that no targets to be observed can be picked within 45° of the Sun or Moon. Once the X body axis is pointed at the target the telescope is rolled about this axis until the Sun lies in the observatory X-Z plane in the -Z half of that plane. The solar panels can then be gimbaled about the Y axis until the Sun is incident normally onto the solar panels.

insertion (10K km x 140K km) and will be above the LEO population; however, it still will pass through the 12-hour orbit shell (r = 4.17 earth radii) twice per orbit, where the GPS satellite population resides, and it will cross the equator plane twice per orbit where the geosynchronous satellite population resides (r = 6.61 earth radii). There will be a collision possibility on each intersection of the 12-hour orbit shell , whenever the inclination of the Chandra orbit is below 63°. [There are 10 Block I (non-operational) GPS satellites in 63° inclined orbits and 25 Block II (operational) GPS satellites in 55° inclined orbits (9).] Chandra will not always be at the proper altitude when it crosses the equator plane, however, and so will only at times have collision possibilities there. The two burned out solid rocket motor casings will have low perigees, initially around 300 km, and so will pass through the LEO population on every orbit of their lifetimes; SRM2 about once per day and SRM1 about six times per day. SRM2 will also pass through the 12-hour orbit shell and possibly the GEO ring on each of its orbits. (These objects are not unique, however, because each TDRSS launch has left a burned out motor casing in a LEO crossing orbit.)

We are not concerned per se about the fate of the spent casings but they can endanger other satellites and a collision would also possibly increase the existing debris population which is a concern. The spent SRM casings will be considered orbital debris as soon as they have burned out and separated from Chandra. Chandra will not be considered debris until its lifetime or mission is over. NASA has issued guidelines to be followed (10,11) to prevent the aculumlation of orbital debris. These guidelines essentially call for the removal of debris objects within 25 years of the completion of their mission. Because there is no active control of the spent casings, there is little that can be done to insure that the guidelines are followed in this case. Only natural forces such as atmospheric drag and gravitational perturbations from the Sun and Moon can be counted on to eventually remove these objects from orbit. Long term numerical integrations can give some indication of how long this might be but it is known to be highly dependent on the launch date and launch time because of the phasing with the Sun and Moon. The probabilities of collision between the spent SRM casings and the existing LEO population have not been made. The probabilities should be comparable to those of the spent casings from previous IUS missions.

We are concerned about the fate of the Chandra, however, as well as any other active satellite it might encounter. Because of this we have made some estimates of the collision probilities between the Chandra and the existing satellite population that it has any chance of encountering. These probabilities do not go directly into the mission planning unless they should turn out to be very high in which case consideration would be given to changing the basic mission plan to reduce the probabilities.

Orbital Transfer Scheme

The orbital transfer scheme for getting the Observatory to its final operating orbit was to launch the Shuttle due East from KSC and put the payload into a circular 28.°5 inclined orbit at an altitude of about 153 N.Mi. (283 km). This was about the maximum altitude that the Shuttle could put this payload weight. The transfer from the shuttle parking orbit to the final orbit was accomplished using an Inertial Upper Stage (IUS) and the Integral Propulsion System (IPS) of the Chandra. The IUS propelled the Chandra into a transfer orbit and the IPS provided the additional delta velocity to transfer to the final operational orbit. Normally the IUS targets to a specific orbit orientation and energy. However it was decided for this mission, that the IUS would use all the energy to maximize the transfer orbit apogee rather than trim to a specific orbit and all maneuvers would be in-plane. This would require that an acceptable orbit plane orientation be provided by the shuttle which would determine the allowable launch window.

The desired argument of perigee location within the orbit plane determined the IUS ignition time which occurred near the lowest declination of the shuttle park orbit. Based on the shuttle park orbit achieved, the IUS determined the desired ignition time and deployment occurred 3,600 seconds prior to that time. Rev 6 was selected to provide coverage of the Chandra solar array deploy from a ground station located at Diego Garcia. On the 6th rev the Payload was deployed from the bay by ejection springs. The Shuttle then backed away from the payload and one hour later the IUS first stage solid rocket motor (SRM1) burned for about 2 minutes putting the payload into an elliptical orbit of 290 km perigee and 14,000 km apogee. The SRM1 then separated and two minutes later the IUS second stage solid rocket motor (SRM2) burned for about 2 minutes putting the payload in an elliptical transfer orbit of approximately 300 km altitude perigee and 74,000 km altitude apogee. Following the SRM2 burn, the IUS also burned all remaining RCS propellant to increase the apogee on the transfer orbit, reserving only the amount to perform the collision/contamination avoidance maneuver (CCAM) following separation. The CCAM maneuver was designed to provide an acceptable separation distance between the Chandra and SRM2 during the coast until the first IPS burn. Following the RCS burn the IUS remained attached to the Chandra providing attitude control until the solar arrays were deployed and the MUPS was activated. At that time separation occurred and the CCAM maneuver took place. The timing of the IUS events were determined by Boeing to provide maximum performance.

At this point the Observatory was on the outbound leg of the quoted elliptical orbit with a period of about 25 hours. Since the IUS transfer orbit is highly eccentric and the IPS is relatively low thrust, the IPS burns are most efficient if done near the apsides. The original IPS transfer plan (known as Low Intermediate Perigee Plan or LIPP) was to complete the transfer in a sequence of 3 coast-burn periods. Shortly thereafter a decision was made to include a very short demonstration of operation burn. First there would be a 36.7 hour coast from the IUS separation to the second apogee of the IUS transfer orbit where the demo burn (IPS1) would occurr raising perigee to about 425 km. Then following a 12.4 hour coast to perigee, a burn (IPS2) would occur expending approximately ½ of the delta velocity required to raise apogee to 140,000 km. This would raise apogee to about 97,430 km. Then following a 35.9 hour coast to the next perigee a third burn (IPS3) would occur to finish raising apogee to 140,000 km. Finally after a 29.1 hour coast to apogee the final burn (IPS4) would be made to raise the perigee to 10,000 km. This profile was the most efficient use of the delta velocity available. As the concern grew that lunar eclipses would become a problem later in the mission any effort made to avoid these events was deemed worthwhile. At that time a decision was made to split IPS4 into two burns. The first of these burns would provide approximately 90% of the delta velocity with 10% reserved to target to a specific perigee to avoid lunar eclipses. The process used to determine the specific perigee will be discussed later.

Many variations on this original scheme were considered before the final transfer orbit was chosen and many iterations were caused by continual updating of weights and updating of estimates of Shuttle and IPS performance. Initially the planned transfer was barely able to achieve the desired orbit of 140,000 km apogee and 10,000 km perigee. As the program matured and changes were implemented a final orbit of 140,000 km apogee and 16,725 km perigee was achievable. Some of these changes were: shuttle park orbit was raised from 130 nmi to 153 nmi; the shuttle cargo weight was increased from 49,800 to 50,228 pounds ; and finally the Chandra weight was reduced from the design weight of 12,960 pounds to 12,495.

In the original plan there was no requirement that the IPS burns occur over a Deep Space Network (DSN) site. In fact while the apogee burns would occur in view of a DSN site, the perigee burns did not. However as performance became less an issue, a decision was made to shift the IPS burns so they would occur in view of a DSN site. This would be an inefficient use of the IPS system and would reduce the perigee from the 16,725 km that could be achieved back toward the 10,000 km minimum. The reason that performance became less an issue was that an early measurement of the goodness of the final orbit was the per cent of the orbit that was above 60,000 km. For the apogee of 140,000 km as the perigee increases from 10,000 km, the per cent time above 60,000 km decreases slightly (0.10s %) until it exceeds 30,000 km. Figure XX shows this relationship. Therefore no reduction in the mission measurement of goodness was encountered if the perigee was reduced from 16,725 km back toward 10,000 km.

Shifting the IPS burns so that they would occur in view of a DSN [illegible] done in either of two methods. Either by moving the burns away from the apsides which would also shift the argument of perigee or by rem[illegible] transfer orbit while the orbit drifted so that the apsides were in view of [illegible] DSN site. Since shifting the argument of perigee was undesirable and at the time this method was proposed there was insufficient performance [illegible] burns and still achieve an acceptable perigee. Therefore the second method was selected and a revised transfer plan was developed. Along with remaining in the transfer orbit, raising perigee of the transfer orbit to an intermediate value enabled the mission designer to control the orbit drift. The revised plan (known as High Intermediate Perigee Plan or HIPP [illegible] proposed by TRW.

After separation from the IUS and coasting 36.6 hours to the first apogee, a short burn (IPS1) was made which raised perigee to 1,215 km. After coasting 25.4 hours to the next apogee, a second burn (IPS2) was made to raise perigee to approximately 3,500 km. Then after coasting 92.6 hours to the second perigee, the apogee was raised to 140,000 km in a single burn (IPS3). Finally following coasts of 30.0 hours and 63.1 hours the perigee was raised to the final altitude in burns (IPS4 and IPS5) at the next two apogee pa[illegible] The first of the apogee burns was approximately 90% of the total, reserving 10% for the final targeting maneuver. This revised transfer strategy could only be used if a nominal IUS transfer orbit was achieved. A decision regarding which transfer method to use was made following confirmation of the IUS transfer orbit. Figure YY shows the Chandra operational orbit that could be achieved as a function of the IUS transfer orbit apogee for both [illegible] methods. It shows that in order to achieve a perigee of 10,000 km using the revised transfer plan, the IUS transfer apogee had to exceed 72,000 km.

The targeted perigee was determined for both the LIPP and the HIPP in the same manner. Prior to the making the 90% apogee burn, the maximum perigee that could be achieved was determined. At this time all the other orbit parameters (apogee, inclination, argument of perigee, and right ascension of the ascending node) were known. Then beginning with the maximum perigee, simulations were run to determine a perigee that would give the best chance of avoiding a debilitating eclipse event. These simulations propagated the Chandra orbit for 10 years, calculating the lunar and solar eclipses. If at any time during the 10 year mission an unacceptable eclipse event occurred, that orbit was rejected as an operational orbit. Perigee was incremented downward from the maximum with 10 year simulations run for each candidate perigee until a satisfactory perigee was found. That perigee had to have a band both above and below so that the IPS uncertainty in accuracy [illegible] into the band. Since the IPS used an open loop control system [illegible] control system aligned the Chandra to the desired attitude and the IPS was ignited for a predetermined duration with no attitude correction [illegible] resultant orbit could only be determined from tracking/telemetry at the conclusion of the burn. That's why the targeted perigee could not be [illegible] determined because until after the last perigee burn (IPS3) the predicted apogee had a large uncertainty.

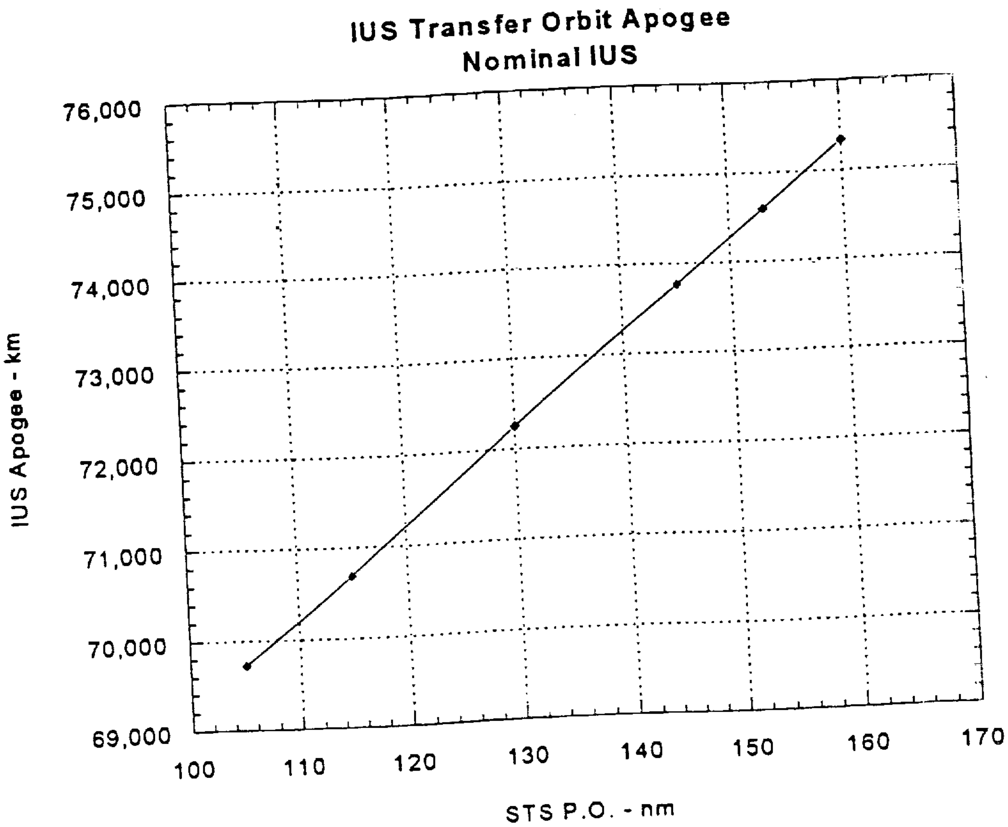

Figure 1 (Not sure if this figure is needed)

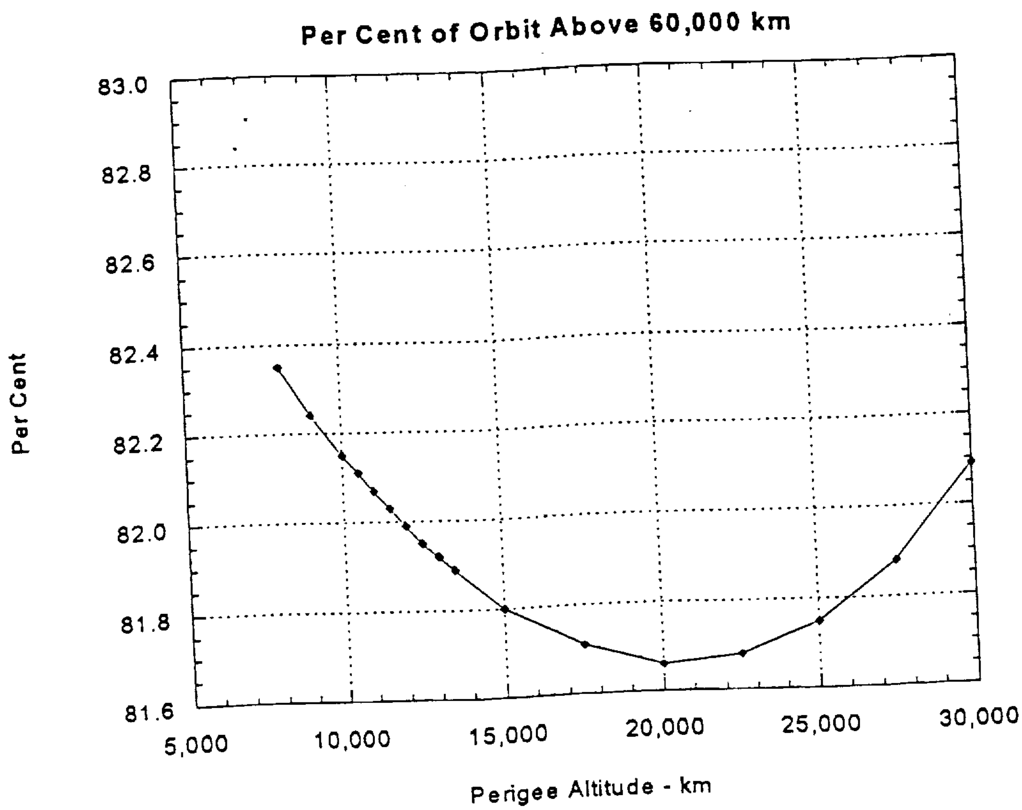
Per Cent of Orbit Above 60,000 km
83.0
82.8
82.6
82.4
82.2
82.0
81.8
81.6
Per Cent
5,000
10,000
15,000
20,000
25,000
30,000
Perigee Altitude - km

Figure XX

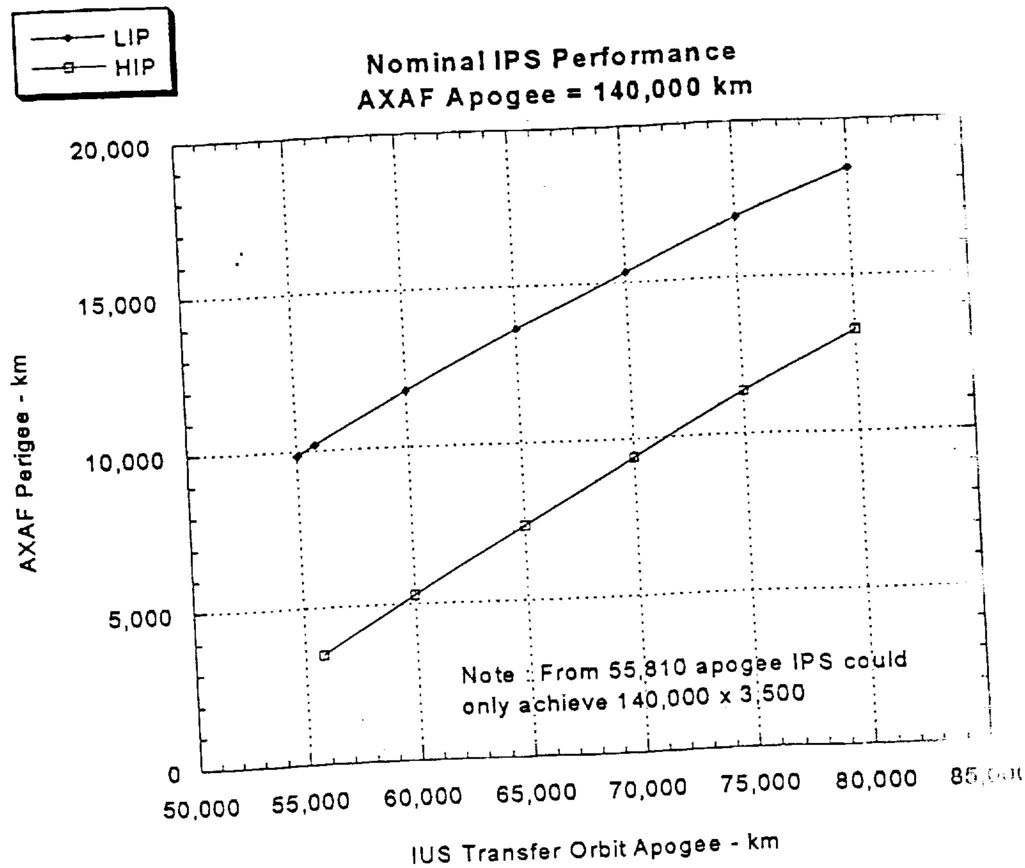

Figure YY

Communications Requirements; Choosing .

The communication between the Observatory, up and down, for the entirety of the mission (5 or 10 years) is to be through the JPL's Deep Space Network (DSN) which consists of three ground stations at Goldstone, Madrid and [illegible]. Since the Observatory will spend most of its time near apogee and [illegible] two of the three ground stations are in the Northern hemisphere, it was [illegible], from a communications point of view, that the apogee of the orbit should be placed in the Northern hemisphere. This would put the observatory in [illegible] one of the two northern stations most of the time. Since initially there was little other reason for choosing the initial argument-of-perigee, ., the communication requirement was sufficient reason to choose the initial argument-of-perigee to be at or near 270°. This orientation was achieved by proper timing of the IUS burn.

Free Variable ....Launch Window

This still left the initial value of the right ascension of the ascending node, ., as a free variable. Whatever that turned out to be would, in turn, define the launch window. It was necessary to examine the long term behavior of the orbit in order to have some insight into how to choose ..

Problem Formulation

This orbit is considerably larger than most ever considered before by NASA **(Question: by NASA or by MSFC? I think there have been some rather highly elliptical orbits with high apogees but they have not had the stability or pointing requirements)** for a long-term mission like this. The apogee distance is nearly 40% of the mean distance to the Moon and orbits this far away from the Earth will be perturbed considerably by the lunar and solar gravitation fields, in fact, much more so than by the oblateness of the Earth. Mission analysis tools routinely used previously for near Earth missions, most of which ignored solar and lunar gravity, were inadequate for the required long term integrations for this mission. Since eclipse considerations were important was crucial that these integrations be as accurate as possible.

The problem at hand is a restricted 4-body problem, the Earth, Sun, Moon and spacecraft. Restricted because the fourth body, the spacecraft, is affected gravitationally by the three massive bodies (Earth, Sun and Moon) but its mass is so infinitesimal compared to the other three that it does not affect their motion. Thus, the motions of the three massive bodies are taken as given or known and do not have to be integrated. Only the equations of motion of the Observatory, relative to the Earth, are integrated. The solar and lunar positions are calculated from analytic theories, the Sun from Simon Newcomb's theory (2) with the epoch coefficients updated from the original B1900.0 to the epoch J2000.0 (3). The Moon's position is calculated from a truncated version of E. W. Brown's theory as summarized in Escobal (4). This version is claimed to give an accuracy of 30 arc seconds in the calculated lunar position (and when compared to the lunar positions listed in The Astronomical Almanac this appeared to be true). The first three zonal harmonics of the Earth's gravitational field were also included in the force calculations. Solar radiation pressure was considered briefly but soon discarded as being insignificant. Atmospheric drag was not a factor because of the altitude of the orbit.

Integration Methods

The integration method originally chosen was a 7th order Runge-Kutta-Felhberg method (5) with step size control, integrating an Encke formulation of the equations of motion. It took some experimentation and comparison with other results to determine an adequate tolerance level for the step size control to produce acceptable results. Later a Cowell formulation of the equations of motion was implemented with an Adams-Bashforth predictor-corrector integration method, called the PECE method in (6). This proved to be faster and slightly more accurate than the Runge-Kutta as compared to other integrations (7).

The integrations soon revealed that the lunar-solar perturbations could produce some very large changes to the initial orbit. The oscillations in the apogee and perigee altitudes, depending on the initial orientation of the orbit relative to the Sun and Moon, could be as large as 30,000 km over time spans of 10 years. These oscillations, in some cases, can cause early impact with the Earth thus ending the mission prematurely. The inclination of the orbit also can undergo large oscillations. Starting from the nominal 28.°5, it can, in some cases, increase to 80° or more and then plunge to near 0° in time spans of a few years. These changes were much greater than those normally encountered in near Earth mission planning and the length of the mission was also much greater than most (those without reservicing capability). Once the Observatory was on-orbit and ready for operations there would be no more orbit control or orbit adjust capability (there is only attitude control) so we had to be very careful in choosing the initial orbital orientation so that the orbit would evolve in an acceptable manner over the life of the mission.

Eclipse Calculations; Earth and Lunar

surface that was occulted. The percentage occultation of the Sun as a function of time was a "V'-shaped curve and the maximum occultation was usually less than 50%. There were a few 'double' eclipses with the percentage occulation being a 'W'-shaped curve. These usually occurred near perigee of the AXAF orbit where the AXAF went through the shadow cone near an apsis, came back out of the shadow and then went back through the shadow cone on the other side of the apsis. We can't recall any Lunar eclipses occurring on consecutive orbits. There would be just one isolated Lunar eclipse and then usually another one wouldn't occur for several months or more. Sometimes there would be periods of 3 or 4 years between Lunar eclipses.

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