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Advanced X-Ray Astrophysics Facility-Spectrometry Spacecraft Pointing Control System: Conceptual Design

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The Advanced X-Ray Astrophysics Facility-Spectrometry is an orbiting astronomical observatory using grazing-incidence optics to focus x-ray images on a liquid-helium-cooled imaging x-ray spectrometer (XRS). It will be launched on a Delta-II launch vehicle into a 650-km, circular, sun-synchronous orbit and will be operated as a national observatory for an expected period of 5 years. This paper describes the conceptual design of the pointing control system (PCS), whose function is to acquire x-ray targets on the XRS detector and maintain a pointing stability of 5.5″ (rms) per axis over any 10-min period that excludes a day-night or night-day terminator crossing. The proposed PCS design attempts to use the best features of previous PCS designs, but has several new features not found on other spacecraft. Hence, it can serve as a good starting point and reference for PCS designers of future spacecraft. Error budgets and preliminary simulation results show the soundness of the design approach.

Introduction

THIS paper presents a conceptual design for the pointing control system (PCS) on the Advanced X-Ray Astrophysics Facility-Spectrometry (AXAF-S) spacecraft. Figure 1 shows a drawing of the spacecraft with some of the PCS components identified. The body-fixed axes of the vehicle coordinate system are denoted by V_1, V_2, and V_3. The PCS design is driven by a desire to keep the PCS simple, low-cost, and low-risk. Maximum utilization is made of methods proven successful on other spacecraft, like those described in Refs. 1–9. Hence, the design is a combination of the best from the past; but it also has several new features not found on other spacecraft. Thus, the PCS design concept presented here can serve as a good starting point and reference for someone with the task of designing the PCS on a future spacecraft.

The outline of the paper is as follows: first, the PCS requirements are summarized; then an overview of the mission sequence and the PCS design is presented; next, the PCS is described in much more detail; preliminary simulation results are presented; finally, concluding remarks are made.

PCS Requirements

The AXAF-S spacecraft will be launched from Vandenberg Air Force Base into a 650-km circular sun-synchronous orbit by a Delta-II launch vehicle. A sun-synchronous orbit is characterized by the fact that the orbit line of nodes is approximately perpendicular to the sun line and the nodal regression rate exactly matches the orbital rate of the earth around the sun. To achieve such an orbit at this altitude, the orbit inclination needs to be 97.9745 deg. The requirement on the PCS is to point the x-ray spectrometer (XRS), the spacecraft scientific instrument, at celestial targets specified by the operations control center (OCC) to an accuracy of 22″ (3σ) per axis. See Fig. 2 for the pointing-accuracy error budget. However, recalibrating the PCS in orbit to meet this accuracy requirement should not have to be done any more frequently than once every 30 days. The celestial targets are nominally within ±20 deg of the orbit plane; however, they can lie anywhere on the celestial sphere that is 45 deg from the sun line. The observing periods for a given target can reach 60,000 s (17 h), not including the time that the target is occulted by the earth. The pointing stability requirement during the observing periods is 5.5″ (rms) per axis, over any 10-min period that excludes a day-night or night-day terminator crossing. See Fig. 3 for the pointing-stability error budget.

From PCS sensor measurements telemetered to the OCC, the XRS attitude must be reconstructed ex post facto to an accuracy of 10″ (3σ) per axis. To change targets, the spacecraft must be able to slew 90 deg in 30 min. At the end of a slew, a 1-min period is allocated to adjust the vehicle attitude in order to meet the pointing-accuracy requirement. After this, a 2-min settling period is allowed for the solar-array motion and liquid-helium slosh in the XRS Dewar to damp out. Then the pointing-stability requirement should be satisfied.

In the event of a failure of onboard hardware such as the onboard computer (OBC) the AXAF-S safing system must be able to perform the following functions, without any help from the OCC, for 72 h: maintain control of the vehicle and keep the solar arrays pointed at the sun, in order to keep the batteries in the electrical power system charged; maintain the ability to communicate with the OCC; prevent the telescope line of sight from pointing within 20 deg of the sun line, to avoid damage to the telescope or the XRS; and prevent the XRS Dewar, located on the back (i.e., $-V_3$ side) and rear (i.e., $-V_1$ end) of the spacecraft, from pointing toward the sun for extended periods. This prevents rapid boiloff of the liquid helium in the Dewar, which would shorten the lifetime of the AXAF-S mission. The mission lifetime is 5 years or more.

Mission Sequence and an Overview of the PCS Design

After the Delta-II inserts the AXAF-S into the proper 650-km orbit, its second, and last, stage should position the AXAF-S so its $+V_3$ axis is within 1.4 deg (3σ) of the sun line, just before separation. Separation should produce an AXAF-S tipoff rate of no more than 0.2 deg/s (3σ) when the Delta-II uses the secondary latch separation system. This system minimizes the tipoff rates and thus allows the AXAF-S PCS to use smaller reaction-wheel assemblies (RWAs) for attitude control. Upon separation, power is applied to the OBC and the activation sequence begins. The solar arrays begin to deploy, and the PCS is switched to the standby mode, which is basically a free-drift mode. See Table 1 for a summary of the PCS modes of operation. Essential PCS hardware for sun acquisition and safing operation is powered up. This includes the RWAs, the magnetic torquers (MTs), the inertial reference units (IRUs), the coarse sun-sensor assemblies (CSSAs), a magnetic sensing system (MSS), and

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a safemode computer and electronics (SMCE). Essential elements of the communication system are also powered up. These include a pair of low-gain antennas, which allow telemetry data to reach the OCC via the Tracking and Data Relay Satellite System or the Deep-Space Network. If a PCS component fails to respond to a power-up command, the OCC can repeatedly issue a real-time command to try to power it up.

After the rate gyros in the IRUs spin up sufficiently (2 min), the PCS attitude determination system begins reconstructing the spacecraft attitude using the IRUs, the CSSAs, and, if desired, the MSS. The initial estimated attitude is the expected attitude at separation. Next, the OBC switches the PCS to the gyro hold mode, to damp out the tipoff rates caused by separation. Once the solar arrays are completely deployed (15 min), the OBC switches the PCS into the safemode OBC sun-point mode. This causes the spacecraft to execute a two-axis slew that points the $+V_3$ axis toward the sun to an accuracy of 1.0 deg. The solar arrays are now generating maximum power for recharging the batteries in the electrical power system. Now, the aperture door is opened, the primary star tracker (ST) is powered up, and the other onboard systems are activated and verified by the OCC. Measurements telemetered to the OCC allow the reference stars in the ST field of view to be identified and the orientation of the spacecraft line of sight in inertial space to be accurately determined. Then, the OCC uplinks this information to the spacecraft and switches the PCS to the science observation mode. This accurately initializes the vehicle attitude determination system. Next, the OBC switches the PCS into the stellar hold mode, in order to accurately estimate and compensate for the drift rates of the online rate gyros, in both their low- and high-rate modes. From sensor telemetry data, the OCC determines sensor alignment error corrections and uplinks these to the spacecraft.

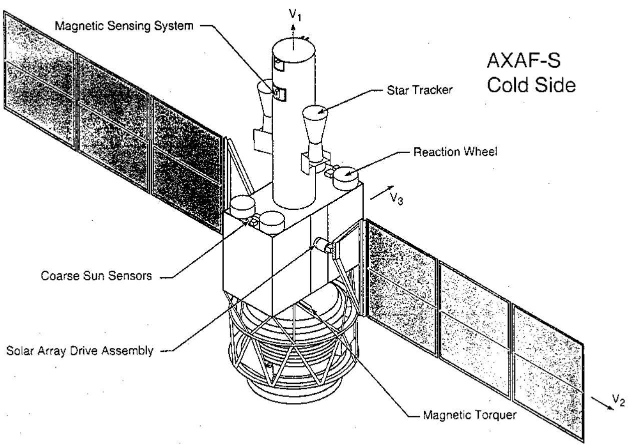

Fig. 1 AXAF-S spacecraft.

Now, AXAF-S is ready for the orbital verification phase of the mission. Here, a series of tests are performed which verify the complete health and performance of the PCS and the other onboard systems. When these are finished, the spacecraft is ready for the science phase of the mission. This begins with the vehicle slew to the celestial target specified in the first target objective load stored in the OBC. PCS parameters in each target objective load include the target right ascension and declination and an associated roll angle, plus ST reference-star magnitudes and coordinates. The OBC switches the PCS into the slew mode, and the PCS slews the spacecraft to this target. When the rate command at the end of the slew is zero, the OBC switches the PCS into the science observation mode. Now, the OBC begins: reading the ST outputs, checking the identity of the stars in the field of view, and updating the estimated attitude with those stars that pass the star identification check. This causes the vehicle attitude control system to adjust the attitude of the vehicle, so the XRS accurately points at the desired target. After the solar-array motion and liquid-helium slosh have damped out, the XRS begins to collect and store science data.

Three safemodes are provided to satisfy the safing requirements. They are the safemode inertial hold mode, the safemode OBC sun-point mode, and the safemode contingency sun-point mode. Table 1 provides a short description of these.

PCS Description

The PCS can be described by its modes of operation, its primary systems, and its hardware. The modes of operation are summarized in Table 1. The others are described in the following sections. Additional information can be found in Ref. 10.

PCS Systems

Vehicle Attitude Determination System

The vehicle attitude determination system is composed of the STs, the CSSAs, the MSSs, the IRUs, and the vehicle attitude determination software routine in the OBC. Its purpose is to determine the AXAF-S attitude relative to the reference coordinate

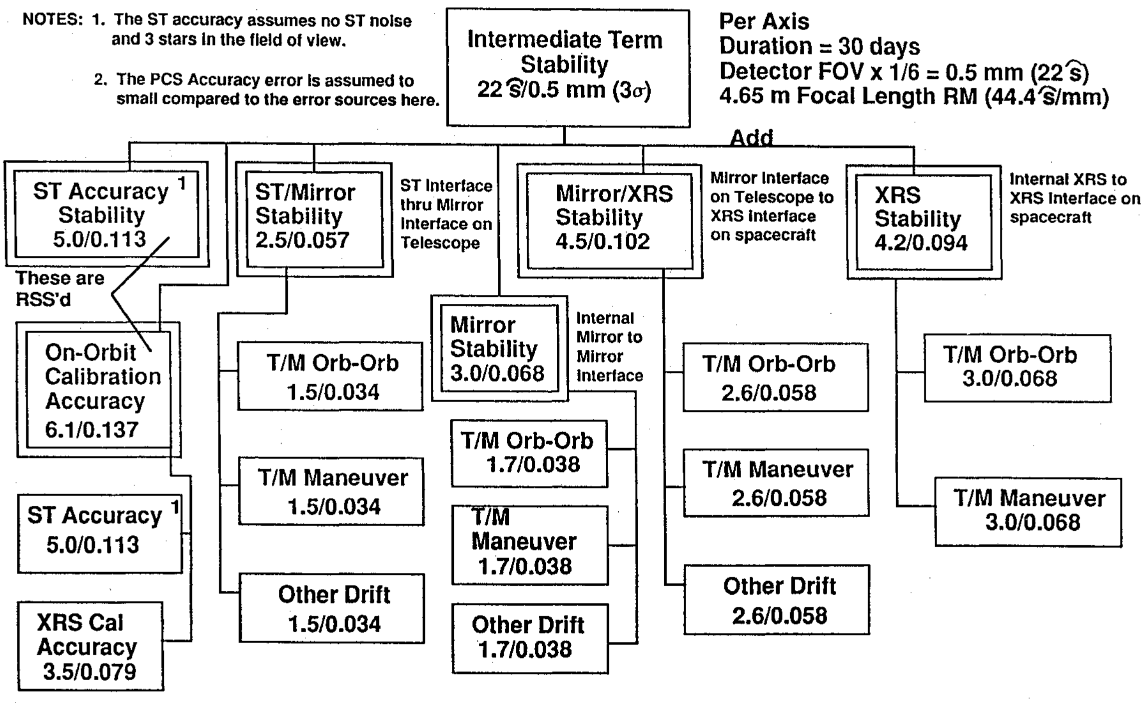

Fig. 2 Intermediate-term stability error budget.

Table 1 PCS modes of operation

Mode	Description	PCS hardware used
Standby	Spacecraft in free drift with attitude determined from CSSAs and/or MSS	OBC, IRUs, CSSAs, MSS
Gyro hold	Attitude control based on integrals of vehicle rate. Attitude determined from CSSAs and/or MSS, but not used for control	OBC, RWAs, MTs, IRUs, CSSAs, MSS
Science observation	Primary fine pointing mode for the spacecraft. Attitude determined from ST	OBC, RWAs, MTs, IRUs, ST, MSS
Stellar hold	Backup to science observation mode for fine pointing. Attitude controlled to keep stars fixed in ST FOV, without star identification. IRU drift rates estimated	OBC, RWAs, MTs, IRUs, ST, MSS
Slew	Spacecraft slewed to new attitude on IRU measurements only. Solar arrays slewed to new position	OBC, RWAs, MTs, IRUs, MSS, SADA, SADE
Safemode inertial hold	First level of safing. Spacecraft attitude trimmed for maximum power without moving solar arrays; then spacecraft holds this inertial attitude	OBC, RWAs, MTs, IRUs, CSSAs, MSS
Safemode OBC sun-point	Second level of safing. Spacecraft $+V_3$ axis points at sun, and solar arrays are slewed to look out $+V_3$ axis	OBC, RWAs, MTs, IRUs, CSSAs, MSS, SADA, SADE
Safemode contingency sun-point	Third level of safing. Like safemode OBC sun-point Mode, except SMCE controls spacecraft	SMCE, RWAs, MTs, IRUs, CSSAs, MSS, SADA, SADE

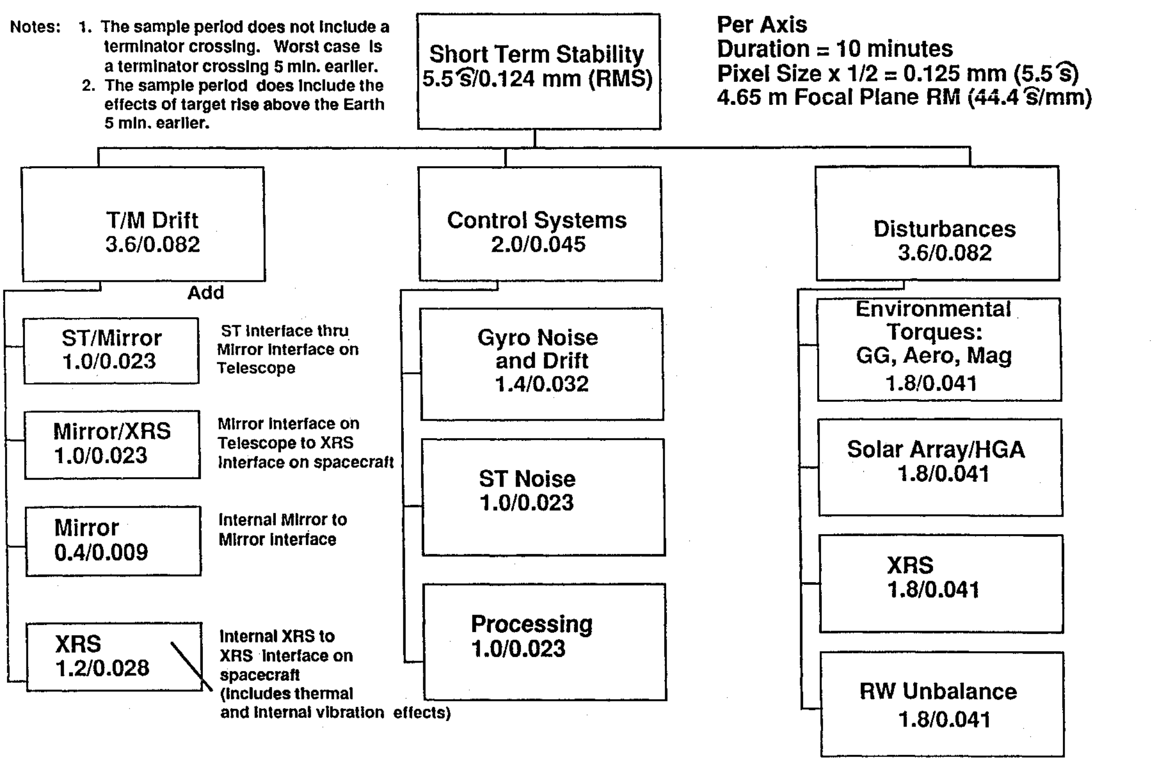

Fig. 3 Short-term stability error budget.

system. Specifically, it determines the orientation of the AXAF-S vehicle coordinate system (V_1, V_2, V_3) relative to the reference coordinate system (R_1, R_2, R_3), based on measurements from these sensors and on computations in the software.

The orientation of the reference coordinate system relative to the Earth-centered inertial axes depends on the PCS mode of operation. In the science observation mode and the safemode inertial hold mode, that orientation is unique for each target. The R_1 axis is in the direction of the AXAF-S target. The R_3 axis is oriented so that the sun line lies near the R_1-R_3 plane, as shown in Fig. 4. This orientation is defined to the OBC by uplinking from the OCC the target right ascension and declination, as well as the desired roll angle. Observe that (R_1, R_2, R_3) is an inertial coordinate system for a given target and observation.

In the stellar hold mode and the gyro hold mode, the previous reference coordinate system is used here. Hence, entering these modes does not affect the choice of the reference coordinate system or its orientation relative to Earth-centered inertial axes.

In the slew mode, the reference coordinate system rotates about the eigenaxis that takes it from its initial orientation to its new orientation. The rate of this rotation is ramped up to some maximum rate, held for a specified period of time, and then ramped down to zero. This is a way of slewing the vehicle from one attitude to another to satisfy the maneuver requirement, but without exceeding the capabilities of the reaction wheels or unduly exciting the solar arrays or the liquid helium in the XRS Dewar.

In the safemode OBC sun-point mode and the standby mode, the orientation of the reference coordinate system relative to Earth-

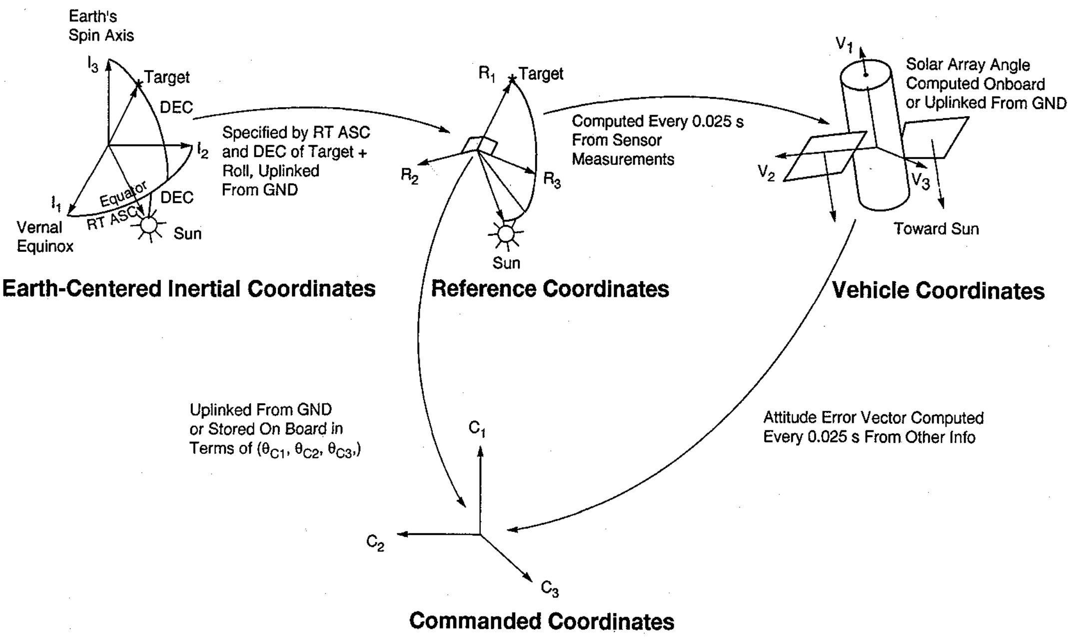

Fig. 4 Coordinate systems for commanding and determining attitude.

centered inertial axes depends on the direction of the sun line in the Earth-centered inertial coordinates, which in turn depends on the time of year. Here, the R_1 axis is aligned with the ecliptic north pole, and the R_3 axis is directed toward the sun. The parameters that describe this relative orientation are computed onboard using the OBC model of the sun line in Earth-centered inertial axes.

In the vehicle attitude determination system, the attitude of the spacecraft relative to the reference coordinate system is calculated every 0.025 s, based on the IRU outputs, which are read by the OBC every 0.025 s. Every 1.0 s, this attitude is updated according to some combination of measurements from the ST, the CSSAs, and the MSS. Mounting matrices for all sensors are stored in the OBC and used in these computations when appropriate. Parameters uplinked from the OCC and the onboard ephemeris models are also used. Uplinked parameters include corrections to the sensor mounting matrices, which are determined at the OCC from telemetered data. The sensor measurements used in the update scheme depend on the PCS mode of operation and commands uplinked from the OCC. In the science observation mode, the ST measurements update the computed vehicle attitude every 1.0 s, whenever two or more reference stars pass the star identification check. Simultaneously, the vehicle attitude is computed every 1.0 s using the CSSAs and the MSS, as a way of checking them against each other.

In the safemode inertial hold mode, the safemode OBC sun-point mode, and the standby mode, the CSSAs and/or MSS measurements provide updates every 1.0 s to the computed attitude propagated by the IRU measurements. The OCC can choose the sensors to be used in the updates by commands in the command time line or real-time commands. In the slew mode, no attitude updates occur. In the stellar hold mode and the gyro hold mode, the measurements and computations from the previous PCS mode are used here, unless the OCC commands differently. However, the computed attitude is not used in the vehicle attitude control system. Instead, it is stored onboard and telemetered to the OCC.

The safemode contingency sun-point mode is implemented in the SMCE processor. The baseline is to use the CSSA and the IRU measurements to point the $+V_3$ axis of the spacecraft at the sun, without determining the vehicle attitude per se. The vehicle attitude determination system is not used in this mode. The update algorithm for the attitude determination system is based on an approach called "observed vectors." This is a completely general approach to spacecraft attitude determination that allows a variety of measurements to be used in determining attitude. Here, knowing the direction cosines of two or more noncollinear observed vectors in the reference coordinate system and measuring them in the vehicle coordinate system is sufficient to compute the direction-cosine matrix relating the two frames. Figure 5 helps to illustrate this.

Vehicle Attitude Command System

The vehicle attitude command system allows small changes in vehicle attitude to be commanded relative to the reference frame used for determining attitude at the time. It issues discrete yaw, pitch, and roll Euler-angle commands that correspond to small-rotations about the 3, 2, and 1 axes of the reference frame, respectively. The order of the rotations is immaterial, since the commanded angles are assumed to be small. The discrete attitude commands can be real-time commands uplinked from the OCC. They give the OCC the ability to adjust the position of the spacecraft in real time. They can also be commands stored in the command time line and issued at a specified time. A number of these issued sequentially can be used for scanning the XRS over a target.

In all PCS modes except the stellar hold mode and the gyro hold mode, vehicle attitude is determined, and consequently commanded, relative to the reference coordinate system. Here, the vehicle attitude command system specifies the orientation of the commanded coordinate system (C_1, C_2, C_3) relative to the reference coordinate system (R_1, R_2, R_3), as shown in Fig. 4. In the stellar hold mode and the gyro hold mode, the vehicle attitude is determined, and consequently commanded, relative to the initial attitude of the vehicle when the PCS entered the mode. Otherwise, the commands are the same as those described above. In the slew mode and the standby mode, the attitude commands relative to the reference coordinate system are set to zero; however, in the standby mode, vehicle attitude is not controlled. In the safemode inertial hold mode, the yaw and pitch commands are set to zero and a roll command is issued that positions the vehicle so the sun line lies in the V_1-V_3 plane. This points the solar arrays close to the sun for maximum power. Now, the vehicle holds this inertial attitude. The roll command must be previously uplinked from the OCC and stored in the target objective load, along with the other target parameters.

Vehicle Attitude Control System

The vehicle attitude control system is the primary feedback control system for attitude control of the AXAF-S spacecraft and is characterized by the fact that the reaction wheels generate the control torques for vehicle attitude control. It is used in all PCS modes controlled by the OBC, except the standby mode.

In the science observation mode, the slew mode, the safemode inertial hold mode, and the safemode OBC sun-point mode, the vehicle attitude determination system is part of the feedback loop, as shown

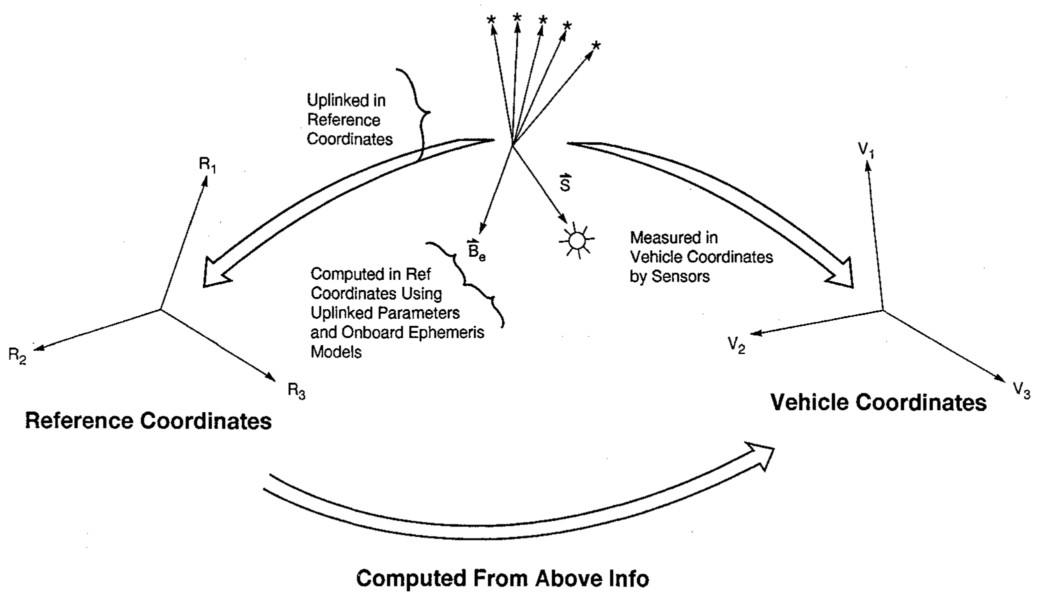

Fig. 5 Concept of observed vectors for attitude determination.

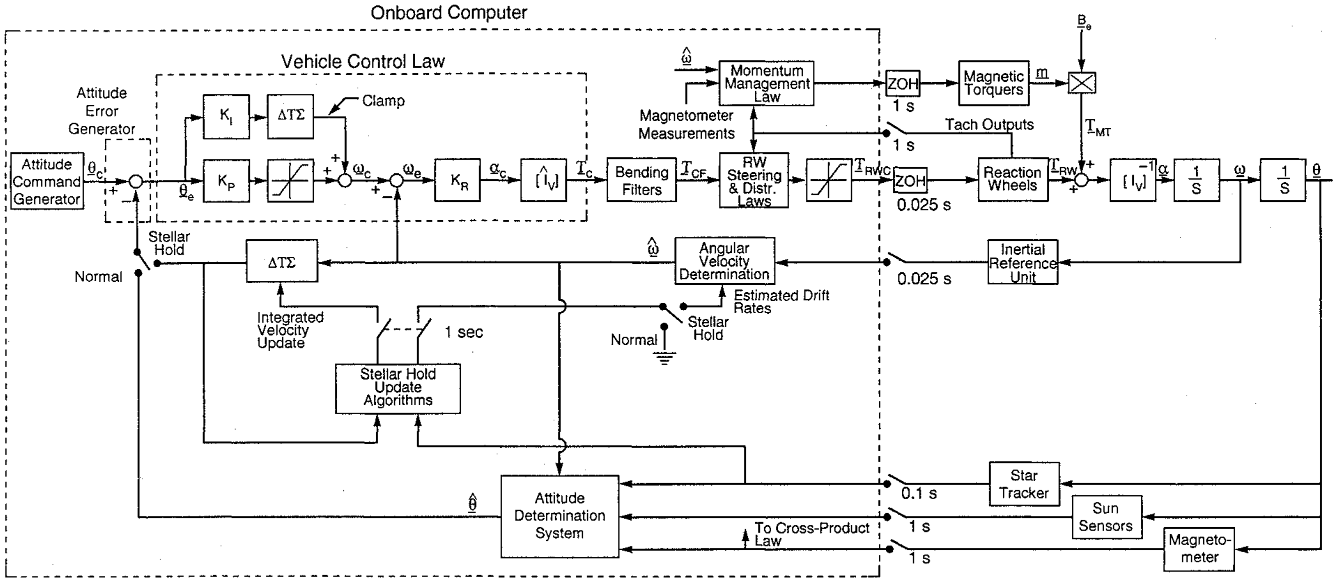

Fig. 6 Vehicle attitude control system and momentum management system.

in Fig. 6. In this configuration, the estimated direction-cosine matrix for the vehicle coordinate system relative to the reference coordinate system, generated every 0.025 s by the vehicle attitude determination system, is input into the vehicle attitude error generator software routine. The attitude commands from the vehicle attitude command system are also input. The vehicle attitude control law is essentially a proportional-integral controller with rate feedback for damping. The software routine for the reaction-wheel steering and distribution laws is the same one used on the Hubble Space Telescope.

The configuration for the vehicle attitude control system employed in the stellar hold mode is also shown in Fig. 6. Observe that the vehicle attitude determination system is operational, but is not used in the feedback loop. Instead, the estimated vehicle rates in vehicle axes are integrated and used for vehicle attitude. Also, ST measurements and integrated vehicle rates are input into a set of algorithms that generate updates to the integrated vehicle rates and estimates for the IRU drift-rate biases in vehicle axes, every 1.0 s.

The configuration for the vehicle attitude control system in the gyro hold mode is the same as that for the stellar hold mode, except there are no updates to the integrated vehicle rates and estimated IRU drift rates using the ST measurements.

The closed-loop bandwidth of each axis of the vehicle attitude control system is expected to be in the neighborhood of 0.10 to 0.5 Hz when the rate gyros are in the low-rate mode. When the rate gyros are in the high-rate mode, the bandwidth will probably be in the neighborhood of 0.01 to 0.05 Hz. Normally, the rate gyros will be in their low-rate mode when the PCS is in the science observation mode and the stellar hold mode; they will be in their high-rate mode in all other PCS modes.

Momentum Management System

The purpose of the momentum management system is to keep the reaction-wheel speeds bounded. This is accomplished by controlling the total angular momentum stored in the reaction wheels and the spacecraft. Torques on the spacecraft resulting from the interaction of the MTs with the Earth's magnetic field are generated, which oppose the total angular momentum vector of the reaction wheel-spacecraft system, on the average. The MT cross-product laws in the OBC and the SMCE processor accomplish this. The angular momentum in the reaction wheels is computed from the reaction-wheel speeds, which are derived from reaction-wheel tachometers. The angular momentum stored in the spacecraft is computed from the vehicle angular velocity, which is derived from the rate-gyro outputs. Based on the computed angular momenta and knowledge of the Earth's magnetic field flux density vector, as measured by the MSS or determined from the OBC's magnetic field model, the dipole-moment commands to the MTs are computed from the cross-product law and issued to the MTs. All operations in the momentum management system are done every 1.0 s, when either the OBC or the SMCE controls the spacecraft. See Fig. 6.

Solar-Array Pointing System

When the OBC controls the PCS, the solar-array pointing system points the solar arrays toward the sun, to an accuracy of 2 deg when the sun line lies in the vehicle V_1-V_3 plane. In this case, the solar-array pointing system is composed of the solar-array drive assembly (SADA), the associated solar-array drive electronics (SADE), and the solar-array pointing software routine in the OBC. The SADA has a torque motor with redundant windings and a harmonic drive that provides gear reduction. There are redundant resolvers in the SADA that measure the position of the SADA shaft.

The scheme for pointing the solar arrays toward the sun is as follows. For each target, the OBC computes the optimum SADA shaft angle for that purpose. This is determined from the onboard model of the sun line in Earth-centered inertial coordinates, plus the transformation from Earth-centered inertial coordinates to reference coordinates. The latter is computed from the right ascension and declination of the target, plus a roll angle that places the sun line near the R_1-R_3 plane. These are uplinked from the OCC with the target objective load. The sun line is not necessarily in the R_1-R_3 plane, because a compromise may be needed between pointing the solar arrays at the sun and pointing the XRS radiator, which looks out the $-V_3$ axis, along the orbit normal and into deep space. The optimum SADA shaft angle computed onboard becomes an input to the solar-array pointing system, unless overridden by an alternative angle uplinked by the OCC. The angle uplinked from the OCC may be part of the command time line, or it may a real-time command. The former allows the OCC to override the onboard computation, if it turns out to be less than optimum. The latter gives the OCC a chance to free a stuck solar array with real-time commands.

In any case, the SADA shaft angle computed onboard or uplinked from the OCC is input into the solar-array pointing software routine in the OBC. The software routine takes this and the actual SADA shaft angle, computed from the SADA resolver outputs that are read by the OBC every 1.0 s, and generates a SADA shaft-angle command profile for gradually moving the solar arrays from their present orientation to their new destination. From this command profile, a commanded shaft angle is generated every 1.0 s and input into the solar-array pointing system control law, along with the actual shaft angle, which is determined by the OBC every 1.0 s from the resolver outputs. The controller computes the torque command for the torque motor each 1.0 s and sends this to the SADE. The SADE generates the current for the torque motor, which produces the required torque. By this process, the solar arrays are moved to their new orientation. Once there, the commanded torque to the torque motor is set to zero, until the spacecraft again changes targets or the OCC issues a real-time command to change the orientation of the arrays. The commanded torque is zero because torque motor friction and cogging torque, amplified by the harmonic drive, are able to hold the solar arrays in position passively.

The SADA and the SADE must be designed to slew the solar arrays 90 deg in 15 min. This is compatible with the requirement to slew the spacecraft 90 deg in 30 min. In many cases, while the spacecraft slews to a new target, the solar arrays can be slewed to their new orientation and properly positioned before the vehicle reaches its new destination.

When the SMCE takes control of the PCS, the solar-array pointing system changes only slightly from when the OBC controls the PCS. First, the SMCE processor, instead of the OBC, reads the resolver measurements every 1.0 s. Second, it computes and issues the commands for the torque motor every 1.0 s. Third, when it first takes control, the solar arrays are driven so they look out the $+V_3$ axis. No other solar-array articulation is required. Otherwise, the two systems are identical.

Safing System

The safing system has three major corrective actions. Two are controlled by the OBC, while the third is controlled by the SMCE.

The first is the least severe of the three and is executed when the PCS enters the safemode inertial hold mode. This happens when the spacecraft is in normal operation and the logic of the safing system monitoring and mode transitions detects an anomaly, such as a failed component. First, the OBC tries to isolate the failed component and continue normal operation. If it cannot do both, it switches the PCS into the safemode inertial hold mode. Then, the vehicle is positioned so the solar arrays point at the sun for maximum power, but without reorienting the solar arrays. The yaw and pitch commands relative to the present reference coordinate system are set to zero, and a stored roll command is issued that rolls the vehicle so that the sun line lies in the V_1-V_3 plane. The vehicle attitude is determined coarsely using the CSSAs and/or the MSS, and the IRUs, but not the STs. If a maneuver to a new commanded attitude is in progress, the maneuver is continued to completion and then the vehicle is repositioned for maximum power. After this, it waits for intervention from the OCC, unless a more severe corrective action is deemed necessary in the meantime.

The second major corrective action occurs when the safing system detects a problem that might be due to the attitude of the spacecraft, the orientation of the solar arrays relative to the vehicle, or the load on the electrical power system. One trigger for this action is when the state of charge of the batteries drops below some threshold level, as set by the OCC. Another is when the spacecraft V_1 axis penerates the sun avoidance cone, that is, when the angle between the V_1 axis and the sun line is less than 45 deg. Then, the OBC switches the PCS to the safemode OBC sun-point mode. This causes the solar arrays to be reoriented so they look out along the $+V_3$ axis, and the spacecraft to be reoriented so the $+V_3$ axis points toward the sun, with an accuracy of 1.0 deg. The aperture door is commanded to close, if this option was previously enabled by the OCC. Also, some of the load on the electrical power system is shed. Attitude is determined in the vehicle attitude determination system using the CSSAs and/or the MSS, and the IRUs. The vehicle rate along the V_3 axis is integrated and used as the attitude error signal for that axis, after a sign change. Once the $+V_3$ axis points toward the sun, the spacecraft waits for intervention from the OCC, unless the third and final, and most severe, corrective action is deemed necessary in the meantime.

The third major corrective action occurs when either one of two situations exists. The first is when the PCS has entered the safemode OBC sun-point mode, but the problem that triggered this action still persists. That is, there is still a problem with the attitude of the vehicle, the orientation of the solar arrays relative to the vehicle, or the load on the electrical power system. The other situation is when the OBC that controls the spacecraft fails. In either case, the OBC in control stops sending keep-alive signals every 1.0 sec to the SMCE. Then, the SMCE takes control of the spacecraft and puts the PCS in the contingency sun-point mode. Like the safemode OBC sun-point mode, this causes the solar arrays to be reoriented so they look out along the $+V_3$ axis and the spacecraft to be reoriented so its $+V_3$ axis points toward the sun, again to an accuracy of 1.0 deg. The aperture door is commanded to close, if this option was previously enabled by the OCC. Severe load shedding on the electrical power system occurs. The vehicle rate about the V_3 axis, derived from the IRU measurements, is integrated and used as an attitude error signal for that axis, after a sign change. Unlike the safemode OBC sun-point mode, these actions are controlled by the SMCE, which solves all control algorithms every 1.0 s. Likewise, it reads all sensors and updates the commands to all actuators every 1.0 s. Once the $+V_3$ axis points toward the sun, the spacecraft waits for intervention from the OCC.

PCS Hardware

A hardware block diagram for the PCS is shown in Fig. 7. The hardware in this figure is described in the following sections.

RWAs

The RWA, with its associated electronics, provides the primary control torque for spacecraft attitude control. The spacecraft has four of these, mounted in a pyramid configuration, with the spin

Table 2 Breakdown of the AXAF-S reaction-wheel torque requirements

Component of T	Value, N-m			Comment
	V_1 axis	V_2 axis	V_3 axis	
Slew accel.	0.015	0.073	0.08	Based on 0.0004-deg/s^2 accel. about one axis at a time. Inertias 2200, 10,400, 11,400 kg-m^2
Grav. grad.	0.0017	0.016	0.014	
Aero	0.01	0.01	0.01	
ADR mag.	0	0.01	0.01	Based on 200 A-m^2 along V_1 in 0.5-G magnetic field
Helium vent	0.00025	0.00025	0.00025	
Other	0.24305	0.16075	0.15575	To counteract remaining disturbances and for feedback control. Based on 0.27 N-m/axis available with 3 RWs
Total	0.27	0.27	0.27	

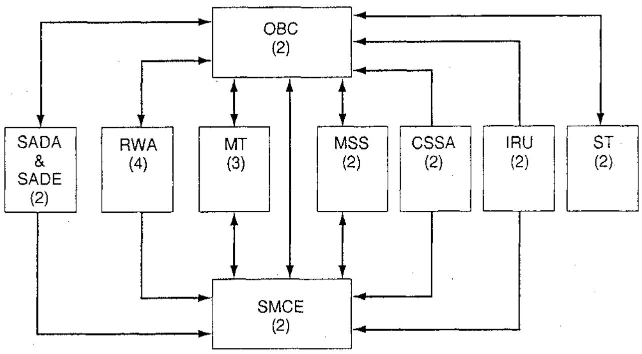

Fig. 7 AXAF-S PCS hardware block diagram.

axis of each inclined 20 deg to the base. This is the same mounting arrangement that was successfully used on the Hubble Space Telescope. With this configuration, any three of the four RWAs can control the attitude of the spacecraft. Normally, all four are operating; but if any one fails, the other three can still control the vehicle and meet the pointing requirements. It just takes longer to slew from target to target. With three or four RWAs operational, they are able to generate at least ±0.27 N-m of control torque and store at least ±75 N-m-s of angular momentum along any of the vehicle axes (V_1, V_2, V_3). The torque and angular momentum storage requirements are broken down in Tables 2 and 3, respectively.

MTs

The PCS has three mutually orthogonal iron-core MTs mounted near the rear of the observatory. These interact with the Earth's magnetic field to produce a torque on the spacecraft that is used for desaturating the angular momentum stored in the reaction wheels. Each torquer has a linear range of ±1000 A-m^2 and a total range of ±1100 A-m^2. On the average, they can desaturate 88 N-m-s of reaction-wheel stored angular momentum per axis per orbit. See Table 4 for the sources of reaction-wheel stored momentum that require desaturation. Each MT has redundant coils, and there are two sets of drive electronics. By uplink command, the OCC can select one set or the other to drive the MTs.

STs

The PCS has two STs; one is for redundancy. Each tracker has an 8-deg × 8-deg square field of view and is aligned close to the spacecraft $+V_1$ axis, which is the telescope line of sight. Actually, they are skewed ±4.0 deg from the $+V_1$ axis in the V_1-V_2 plane; then each is rotated 45 deg about its boresight. This mounting arrangement has these advantages. First, it prevents glints off the spacecraft from corrupting the ST measurements. Second, the trackers' fields of view overlap the XRS field of view. The latter fact allows the OCC

Table 3 Breakdown of the AXAF-S reaction-wheel angular momentum storage requirements

Component of H	Value, N-m-s			Comment
	V_1 axis	V_2 axis	V_3 axis	
Slew	2	11	12	Based on 0.06 deg/s about one axis at a time Inertias 2200, 10,400, 11,400 kg-m^2
Grav. grad. bias residual	3	23	21	Assumes MTs contain H_{bias} to half orbit's worth
Grav. grad. cyc.	2	8	8	Assumes RWs store all of H_{cyc}
Aero. bias res.	10	10	10	Assumes $T_{\text{bias}} = 0.0033$ N-m and MTs contain H_{bias} to half orbit's worth
Aero cyc.	6	6	6	Assumes $T_{\text{cyc}} = 0.0067$ N-m and RWs store all of H_{cyc}
Helium vent res.	1	1	1	Assumes $T_{hv} = 0.00025$ N-m and MT's contain H_{hv} to half orbit's worth
Margin	52	17	18	Based on 76 N-m-s/axis available with 3 RWs
Total	76	76	76	

Table 4 Sources of reaction-wheel stored angular momentum that require desaturation by magnetic torquers

Source	Value, N-m-s			Comment
	V_1 axis	V_2 axis	V_3 axis	
Grav. grad. bias/orb.	5	46	41	
Aero. bias/orb.	19	19	19	Assumes $T_{\text{bias}} = 0.0033$ N-m
ADR mag.	0	9	9	Based on 200 A-m^2 along V_1, in 0.5-G magnetic field, for 900 s, every 3–4 hr
Helium vent	2	2	2	Assumes $T_{hv} = 0.00025$ N-m constant
Margin	62	12	17	Based on MTs desaturating at 88 N-m-s/axis/orb[a]
Total	88	88	88	

[a] $H_{\text{MT/axis/orb}} = 0.5 \times (1000\ \text{A-m}^2) \times (0.3 \times 10^{-4}\ \text{w/m}^2) \times 5864\ \text{s} = 88$ N-m-s. 0.5 is the inefficiency factor, because MTs can only generate torque normal to $\mathbf{B}_e$.

to accurately determine the relative alignment between the XRS and the STs by simultaneously observing a visible x-ray source with all three devices. Also, the OCC can obtain XRS and ST measurements on some x-ray sources simultaneously.

This ST mounting arrangement is like the one proposed in Ref. 2. The ST designated as primary by the OBC provides the reference-star position and magnitude information to the OBC for attitude updates in the science observation mode. It is also used for updating the integrals of the vehicle rates and estimating IRU drift rates in the stellar hold mode.

Each ST can track up to five reference stars from 6 to 1 M_v, concurrently. The tracking accuracy with stars of this magnitude must be 3″ (1σ) or better. This is the allowable spatial error and does not include the noise-equivalent angle. Tracking three or more reference stars concurrently reduces the effective spatial error to 1.7″ (1σ) or less. This is equivalent to 5.1″ (3σ) and is consistent with the PCS error budget for pointing accuracy. The ST two-axis outputs for all stars are read by the OBC every 0.1 s. The noise-equivalent-angle tracking one 6-M_v star must be 3″ (1σ) or better, with an output every 0.1 s. Hence, averaging 10 consecutive outputs in the OBC over each 1-s interval reduces the noise-equivalent angle to approximately 1.0″ (1σ). This is consistent with the PCS error budget for pointing stability.

CSSAs

The PCS has two sets of CSSAs for coarse attitude determination; the second set is needed for redundancy. Each set has six sensor heads, provides 4π sr of coverage, and allows the line of sight to the sun to be determined relative to the vehicle coordinate system to an accuracy of 1.0 deg per axis. Each sensor has a 128×128-deg field of view and an accuracy of 0.35 deg per axis. In each set, one sensor looks out the $+V_1$ axis and a second looks out the $-V_1$ axis. The other four lie in the V_2-V_3 plane. Of these, one makes an angle of 38 deg with the $+V_3$ axis and 52 deg with the $+V_2$ axis. A second makes an angle of 38 deg with the $+V_3$ and 52 deg with the $-V_2$ axis. A third makes an angle of 38 deg with the $-V_3$ and 52 deg with the $+V_2$ axis. The fourth makes an angle of 38 deg with the $-V_3$ and 52 deg with the $-V_2$ axis. This mounting arrangement gives 4π-sr coverage and avoids problems with glints and obstructions from the solar arrays.

IRUs

The spacecraft has redundant IRUs. Each IRU consists of three two-axis rate gyros, mounted so that the vehicle angular velocity can be reconstructed from any two. Figure 8 helps to illustrate this new mounting configuration. Hence, the PCS can function properly even after four rate-gyro failures. Normally, outputs from any three of the six are sent to the OBC and are used to reconstruct the vehicle angular velocity in the vehicle coordinate system. The ones to be used are selected by the OCC. Each rate gyro has a low- and a high-rate mode. The low-rate mode has a range of $\pm 400''$/s and a scale factor of $0.05''$ per pulse. The high-rate mode has a range of ± 2 deg/s and a scale factor of $0.8''$ per pulse. Each rate gyro has a residual drift rate, after in-flight calibration, that is less than or equal to $0.01''$/s (1σ). This means the vehicle attitude drift is approximately $1'$ (1σ) about any vehicle axis, after one orbit in the gyro hold mode, or in the stellar hold mode with no ST updates. This is well within the ST field of view (8×8 deg). It also means that the drift of the vehicle about any axis is $0.01''$ (1σ) after each 1-s interval, just before the ST updates in the science observation mode or the stellar hold mode. This is small compared to the pointing-stability requirement. The residual scale-factor nonlinearity of each rate gyro after in flight calibration, if required, is less than or equal to 350 ppm (1σ). This means that a 90-deg maneuver in 30 min, about a vehicle axis, can be made in the slew mode on IRU control, to an accuracy of approximately $2'$ (1σ), considering residual scale-factor error and residual drift-rate error. This is well within the ST field of view (8×8 deg).

MSSs

The spacecraft has redundant three-axis MSSs for measuring the flux density vector of the Earth's magnetic field. The range is ± 0.6 G in each axis. They are used in the angular momentum management system, when either the OBC or the SMCE controls the vehicle. The MSS measurements can also be used in the attitude determination system, whose software resides in the OBC. The MSSs must be mounted away from any sources of onboard magnetic fields, such as the MTs, to avoid corrupted measurements of the Earth's magnetic field. Hence, they are mounted on the front end of the spacecraft, since the MTs are mounted near the back end.

OBCs

The OBC contains and executes the software routines for normal PCS operation. There are two completely independent OBCs for redundancy. They have no physical link between them, and there is no capability for automatic switchover between them. Switchover can occur only by uplinked command from the OCC. This prevents computer switchover before the OCC can determine what caused the problem with the online OBC.

There are three computation cycle rates for the PCS software. The fast loop is done 40 times per second, the intermediate 10 times per second, and the slow once per second.

Basically, the fast loop involves reading the rate gyros every 0.025 s and determining the attitude from these, generating the commanded attitude, computing the attitude error, solving the control-law equations to generate reaction-wheel torque commands, and issuing those to the reaction wheels every 0.025 s. The delay between the time the rate gyros are read until the reaction-wheel torque commands are issued is less than 0.01 s in every fast-loop computation cycle.

The intermediate loop involves reading the ST outputs every 0.1 s and computing a weighted average of these over 1-s intervals.

The slow loop involves reading the CSSA's and the MSS outputs every 1.0 s and generating attitude updates using either the ST information or else the CSSA and/or MSS measurements. The slow loop also involves reading the reaction-wheel tachometer outputs every 1.0 s and computing the reaction-wheel stored angular momentum, computing the vehicle angular momentum from the estimated vehicle angular velocity every 1.0 s, using these and the MSS measurements to compute the MT dipole moment commands, and then issuing those every 1.0 s. Also, the slow loop involves the measurements, control-law computations, and commands associated with the solar-array pointing system. Finally, the slow loop involves all other computations, such as those for the safing-system monitoring and redundancy management.

SMCE

The SMCE is highly reliable, redundant, and independent of the OBC. It has redundant processors that execute the software routines

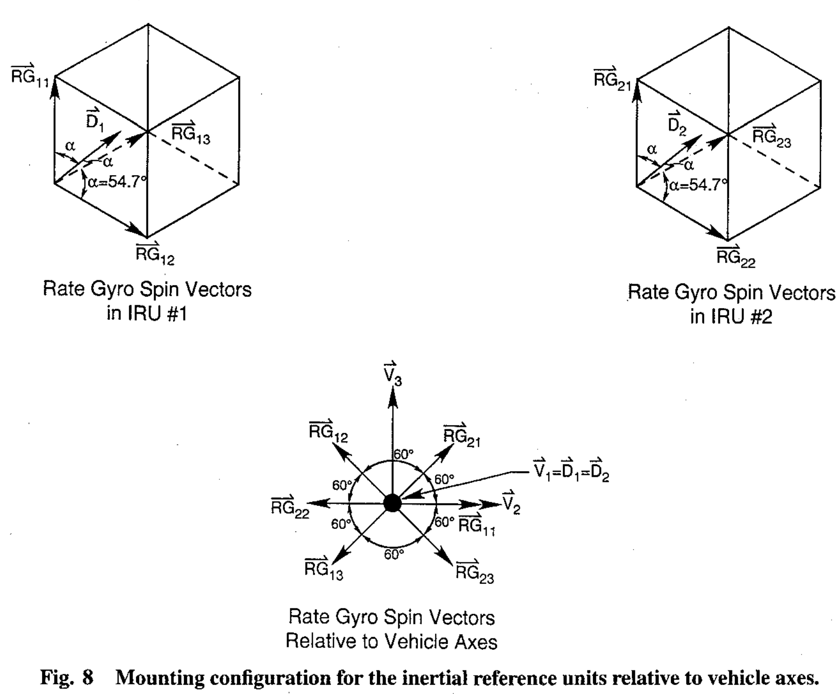

Fig. 8 Mounting configuration for the inertial reference units relative to vehicle axes.

for the contingency sun-point system at a computation rate of once per second. This system is activated upon entry to the contingency sun-point mode, which can be enabled in one of two ways. First, it can be enabled by the OCC; second, it can be enabled autonomously onboard when the OBC stops sending keep-alive signals to the SMCE every 1.0 s. This is a signal to the SMCE to take control of the spacecraft. When the SMCE controls the PCS, the processor designated as primary executes the monitoring software routine. If this detects a problem with the primary processor, control is transferred to the backup.

SADA and SADE

The SADA has a torque motor, with redundant windings, that is connected to a harmonic drive. The harmonic drive attaches to the solar-array drive shaft and provides gear reduction to the system. The solar-array drive shaft rotates through angles in the range of −90 to +45 deg. Here, −90 deg is when the solar arrays look out along the $-V_1$ axis, and 0 deg is when they look out along the $+V_3$ axis. There are redundant resolvers in the SADA that are used to determine the position of the solar-array drive shaft.

The SADE provides the electrical interface between the SADA and both the OBC and the SMCE. It is fully redundant.

Simulation Results

A simplified three-axis computer simulation model of the proposed AXAF-S PCS design was developed. The model includes spacecraft rigid-body dynamics, plus simplified models for the solar array and the liquid-helium slosh dynamics. The star-tracker and rate-gyro noise was modeled; so were the gravity gradient and aerodynamics torques. The control-law gains were chosen for 0.1-Hz bandwidth. The results, shown in Fig. 9, show a pointing stability that is well below the required 5.5″ (rms) per axis over 10 min and consistent with the PCS error budget for stability. This establishes the soundness of the PCS design concept for meeting its most fundamental requirement.

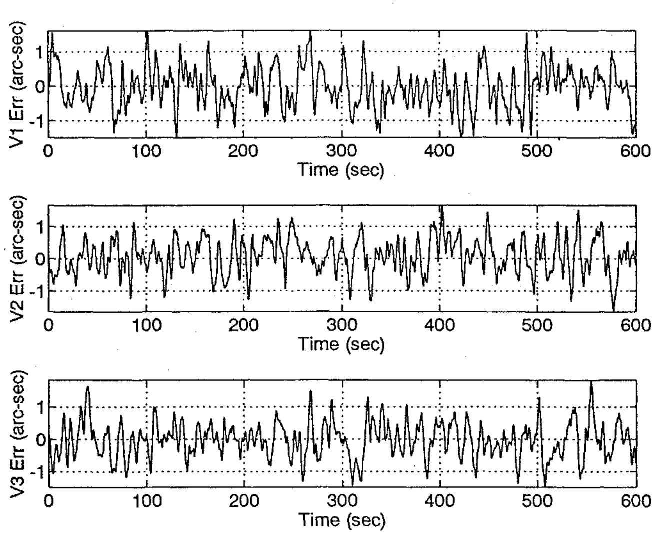

Fig. 9 Preliminary simulation results.

Concluding Remarks

This paper has presented a conceptual design for the AXAF-S spacecraft PCS. This design attempts to utilize the best from past spacecraft PCSs, while adding several new features. The simplified interface with the OCC, and the proposed mounting configurations for the IRUs and the CSSAs, are examples of the latter. Consequently, the PCS design presented here can serve as a good starting point and reference for designers of future spacecraft PCSs.

References

[1]Anderson, G., Quinn, D., Beals, G., Nelson, J., and Nurre, G., "An Overview of the Hubble Space Telescope Pointing Control System Design and Operation," AIAA Paper 92-4616, Aug. 1992.

[2]Bauer, F., Femiano, M., and Mosier, G., "Attitude Control System Conceptual Design for the X-Ray Timing Explorer," AIAA Paper 92-4334-CP, Aug. 1992.

[3]Wider, J., McLaughlin, K., and Schauwecker, C., "Gamma Ray Observatory Attitude Control and Determination On-Orbit Performance," AIAA Paper 92-1148, Feb. 1992.

[4]Williams, R., Dennehy, C., Lee, D., and Welch, R., "An Augmented MMS MACS for the Topex/Poseidon Mission and Beyond," American Astronomical Society, Paper 90-013, Feb. 1990.

[5]Dennehy, C., Kia, T., and Welch, R., "Attitude Determination and Control Subsystem for the Topex Satellite," AIAA Paper 88-4129-CP, Aug. 1988.

[6]Class, B., Bauer, F., Strohbehn, K., and Welch, R., "Space Infrared Telescope Facility/Mutimission Modular Spacecraft Attitude Control System Conceptual Design," American Astronomical Society, Paper 86-031, Feb. 1985.

[7]Bromberg, B., and Croft, J., "On-Orbit Attitude Control of the Cosmic Background Explorer (COBE)," American Astronomical Society, Paper 85-040, Feb. 1985.

[8]Das, A., "The On-Orbit Attitude Determination and Control System for the Landsat-D Spacecraft," AIAA Paper 82-0310, Jan. 1982.

[9]Murrell, J., "The NASA Multimission Spacecraft Modular Attitude Control System," American Astronomical Society, Paper 79-025, Feb. 1979.

[10]Polites, M., Sharkey, J., Nurre, G., and Lightsey, W., "Conceptual Design of the Pointing Control System for AXAF-S," AIAA Paper 93-4250, Sept. 1993.

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