Attitude Sensor Calibration for the Topography Experiment (TOPEX) Spacecraft

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ABSTRACT

A ground-based software system to calibrate the attitude control sensors for the Ocean Topography Experiment (TOPEX) spacecraft is described. The algorithm determines sensor misalignment, bias and scale factor errors from gyro, sun sensor and star tracker measurements. The inherent yaw slew motion of the spacecraft during Normal Mission Mode is exploited to make the error parameters observable,

A two loop recursive least-squares algorithm is implemented with the feature of inhibiting the update of the error parameter state until all of the available data has been processed. This feature eliminates the estimation state feedback typical of other algorithms which can cause instability and convergence problems. The two loop algorithm updates the error parameter state in the outer loop once the error parameter increment has been determined from processing all of the data within the inner loop.

The guidance and control file set (GCFS) interface is introduced. This menu driven user interface has been developed to facilitate running the calibration algorithm, telemetry processing program and other analysis software within a mission operations environment.

Calibration results from actual spacecraft flight test data and the required telemetry data processing arc discussed.

1 INTRODUCTION

The TOPEX/POSEIDON experiment objective is to make accurate, measurements of the world's ocean surface elevation from a space-based perspective. Th is information will benefit the study of global climate and weather prediction, for coastal storm warning and maritime safely. The ocean plays an important role in affecting and controlling global climate due to temperature regulating effect of great bodies of water. By understanding the ocean surface elevation, distribution of water and current circulation pattern at different locations on the globe can be deduced. When integrated with sub-surface data, an improved knowledge of ocean dynamics can be achieved thereby leading to a more comprehensive understanding of the relationship between ocean dynamics and global weather pattern.

The primary sensors for this experiment consist of two types of radar altimeters which are provided by NASA and CNES respectively. These instruments measure the height of the spacecraft with respect to the ocean surface. In addition, information generated from the Precision Orbit Determination process provides the reference of the altimeter data in a geodetic frame. These sensors and other instruments arc housed in the instrument Module (IM) which is attached to the TOPEX Multi-mission Modular Spacecraft (MMS) bus. The MMS bus is made up of four modular subsystems: the Command and Data Handling (C& DH) module, the Modular Attitude Control Subsystem (MACS), the Modular Power Subsystem (MPS) and the Propulsion Module (PM) which arc all mounted to a triangular Module Support Structure (MSS). The MACS consists of all the attitude sensors and actuators responsible for achieving attitude and delta velocity control and other mission payload operations. The purpose of this paper is to describe the ground-based in-flight calibration algorithm design, software implementation and calibration results of tile attitude sensors which arc located at the MSS.

There arc four types of attitude determination sensors; sun sensor, 2 star trackers, gyros and horizon sensors. The star trackers, sun sensor and gyros arc used primarily during normal mission mode. A more detailed description of the spacecraft hardware configuration and the on-board attitude determination algorithm is provided in reference (1) and (2). The horizon sensors arc used only during safe hold mode when in a fault situation. The calibration algorithm developed concerns only the star tracker, sun sensor and gyros and dots not include the horizon sensors.

2 MI SSION REQUIREMENT

The space.craft is launched, using the ARIANE - 42P expandable launch vehicle, into a non-Sun synchronous circular Earth orbit of 1336 Km with an inclination of 66 degrees. It takes about 112 minutes to complete one orbit. During normal mission mode with a fixed yaw Configuration for experiment data collection, the satellite orients its altimeter boresight or yaw axis (Z) to be nadir pointed. The roll axis (X) is aligned with the velocity 'vector of the satellite and the pitch axis (Y) is directed opposite to the orbit normal. A diagram corresponding to the coord inate system definition is presented in figure i.

in order to provide electrical power for satellite operation, a solar array is mounted on the Instrument Module. This array is attached to a solar array drive assembly which allows one rotational degree of freedom along the satellite pitch axis. For maximum sun exposure on to the array, it is desirable to maintain the pitch axis to be orthogonal to the sun vector at all limes. Thus by controlling the solar array pitch angle, the sun vector can be made normal to the array plane. To align the pitch axis to be perpendicular to the sun vector, maneuvering of the spacecraft is required along the yaw axis, The exact yaw angle profile which is defined as the angle, ψ, between the nominal velocity vector and roll axis is provided in figure 2. The actual implementation by the on-board software is, however, a simple sinusoidal function as given in figure 3. The error introduced is considered acceptable for the purpose of solar array control. The yaw steering maneuver which is discussed in depth in reference (2) is implemented using a set of reaction wheels during normal mission mode for science data collection. A side benefit of this maneuver provides the necessary rate stimulus to the gyros along all three axes for proper calibration of gyro relate.d errors. Otherwise, the only excitation is along the pitch axis due to orbital rate. Consequently, all in-flight calibrations are usually performed while in the yaw steering mode,

To achieve the required control and knowledge accuracy of the satellite attitude, it is necessary to perform in-flight calibration of the attitude sensors on a regular basis. The estimated error parameters consisting of star tracker and sun sensor misalignments, gyro scale factor, bias, misalignments and non-orthogonality errors arc then uplinked to the satellite (o bc compensated by the on-board attitude determination and control software for performance enhance.ment,

3 ON-BOARD ATTITUDE DETERMINATION

To provide the background for the calibration algorithm design, it is helpful to describe briefly the on-board attitude determination process and sensors.

The TOPEX attitude determination design is inherited from MMS. It is based on propagating measured gyro rates to determine the inertia] attitude of the spacecraft while periodically updating the inertial attitude using celestial sensors for absolute reference. TOPEX's attitude control sensor components consist of a Dry Rotor Inertial Reference Unit (DRIRU 11), a Digital Fine Sun Sensor (DFSS) and two Advanced Star '1'rackers (ASTRA). These components arc located in the spacecraft as shown in figure 4.

During nominal mission mode, only the star trackers arc used for celestial updates. They provide the line and pixel coordinates of the stars detected within the sensor fic]d-o(-view, An on-board star identification algorithm is employed to associate the star images as sensed by each tracker to the star information stored in a star catalog. A total of 326 stars are used for the entire mission. Star updates are performed every 32 seconds when identified stars arc available. In addition,.the digital fine sun sensor also provides X - Y components of the sun vector, The sun vector pointing information referenced to an inertial frame is provided by an on-board solar ephemeris data base which is updated from the ground on a periodic basis. Finally, the DRIRU II is a strapdown inertial reference unit containing three, dry-tuned, two degree-of-freedom gyroscopes. Output from each gyro channel represents incremental angular rotation about the input axis of the gyro. These channels can be selected to operate in a low rate mode for finer resolution or in a high rate mode for coarser resolution and greater dynamic range.

During the normal attitude determination process, the gyro incremental information is used to propagate, every 0.512 seconds, the spacecraft orientation which is expressed as a quaternion. The rate information is compensated for drift, scale factor, misalignments and non-orthogonality errors prior to attitude propagation. These errors arc represented as on-board parameters which can be updated from the ground upon successful completion of the ground based in-flight calibration process. Every 32 seconds, the on-board Update Filter is invoked to estimate the propagated attitude error and gyro drift based on the available stellar information. Misalignment errors between the gyros and celestial sensors arc compensated for by mapping the measured star or sun vector to the gyro coordinate system. Covariance matrices corresponding to altitude errors and drift rate biases arc also computed by the Update Filter which is based on the Kalman Filter approach, The Update Filter can bc enabled or disabled from the ground independent of the gyro propagation routine.

A more detailed comprehensive discussion of the attitude determination process is presented in reference (2).

4 ALGORITHM DISCUSSION

4.1 OVERVIEW

A batch post processor is implemented as a multiple pass, two loop, recursive least squares parameter estimation algorithm.

Specifically,

- initial attitude errors
- gyro rate scale factor errors
- gyro rate biases
- gyro misalignments and non-orthogonalities
- digital fine sun sensor (DFSS) misalignments and
- advanced star tracker (ASTRA) misalignments

arc determined from the avai lable telemetry and ground based data consisting of:

- initial satellite attitude
- nominal sensor to spacecraft orientation matrices
- gyro measurements
- sun sensor measurements
- star tracker measurements
- solar and satellite ephemeris files
- star catalog and star identification number.

All estimated parameters arc assumed to be time invariant.

The algorithm inner loop (k index) sequentially processes all of the data collected over the selected calibration segment. When propagating the attitude quaternion and evaluating the" measurement function within the inner loop, the parameter vector is fixed. A parameter increment vector is, however, recursively updated based on a calculated measurement gradient matrix and the difference between predicted and actual measured data.

Upon completion of the inner loop processing, the parameter vector is updated in the outer loop (j index) by the estimated parameter increment. The inner loop is then restarted and the process is repeated for several (3-5 typical) iterations until the parameter increment converges to an acceptabl y small value.

4.2 INNER LOOP PROCESSING

Quaternion Propagation

The quaternion representing the orientation of the satellite with respect to the Earth Centered Inertial (ECI) coordinate frame is estimated by integrating the quaternion differential equation:

$$\dot{\hat{Q}} = \frac{1}{2}\begin{bmatrix}\omega_m(x_j, t_k) \\ 0\end{bmatrix}\hat{Q} \qquad (2)$$

where ω_m is the measured rate gyro vector, x_j is the estimated parameter vector for the jth pass of the outer loop and t_k is the inner loop time parameter. The nominal gyro axes arc assumed to be attached to the. spacecraft axes.

Predicted Sun Sensor and Star I'racker Outputs

The predicted sun sensor and star tracker outputs defined in sensor coordinates can be calculated solely from geometric considerations. Data required for the prediction include: I) nominal rotation matrices defining the sensors' orientations relative to the satellite body; 2) solar and satellite ephemeris files for the sun sensor; and, 3) a star catalog and star identification number for the two star trackers.

The measurer-nent function, H_k, relates the known celestial line-of-sight vector, A^i, the estimated satellite quaternion and the error parameter vector (in inertial coordinates) to the predicted sun sensor and star tracker outputs, $\hat{z}_{m_k}$:

$$\hat{z}_{m_k} = H_k(\hat{Q}, A^i, x_j, t_k) \qquad (2)$$

Measurement Function Gradient

The change in the predicted sun sensor and star tracker outputs duc to changes in the error parameter vector can be expressed in terms of the partial of the measurement function:

$$d\hat{z}_{m_k} = \frac{\partial H_k}{\partial x_j}\, dx_j \approx z_{m_k} - \hat{z}_{m_k} \qquad (3)$$

The $d\hat{z}_{m_k}$ increment is approximately equal to the difference between the measured output, z_{m_k}, and the predicted output as shown. The measured output is equal to the sum of the actual output, z_k, and a white noise sequence, v_k, with covariance R_k:

$$z_{m_k} = z_k + v_k \qquad (4)$$

Recursive Least Squares Minimization

The sum of the squares of the differences between the two right sides of equation (3) can be minimized with respect to the error parameter increment by standard recursive techniques [3, pg. 110].

The update gain matrix, K_k, is defined in terms of the parameter increment covariance, P_k, the measurement gradient matrix and the measurement noise covariance:

$$K_k = P_k\left(\frac{\partial H_k}{\partial x_j}\right)^T\left[\left(\frac{\partial H_k}{\partial x_j}\right)P_k\left(\frac{\partial H_k}{\partial x_j}\right)^T + R_k\right]^{-1} \qquad (5)$$

The covariance matrix is updated based on the gain matrix and the measurement gradient matrix according to the computationally desirable form:

$$P_{k+1} = \left[I - K_k\left(\frac{\partial H_k}{\partial x_j}\right)\right]P_k\left[I - K_k\left(\frac{\partial H_k}{\partial x_j}\right)\right]^T + K_k R_k K_k^T \qquad (6)$$

Finally, the parameter increment is updated by incorporating the difference between the measured outputs and the predicted outputs:

$$dx_{k+1} = dx_k + K_k\left[(z_{m_k} - \hat{z}_{m_k}) - \left(\frac{\partial H_k}{\partial x_j}\right)dx_k\right] \qquad (7)$$

4.3 OUTER LOOP PROCESSING

The error parameter vector is update-d from the final parameter increment computed from a complete inner loop pass over the calibration data segment:

$$\underline{x}_{j+1} = \underline{x}_j + d\underline{x}_j \tag{8}$$

The outer loop processing is complete when the final parameter increment, calculated within the inner loop, converges to an acceptably small value.

4.4 MEASUREMENT FUNCTION GRADIENT

To complete, the measurement function gradient, equation (3), must be determined. First skew symmetric matrix concepts and conventions are defined to eliminate any notational confusion. Next, equations for the attitude error and its gradient are derived. Finally, the measurement function and its gradient are the derived,

4.4.1 SKEW-SYMMETRIC MATRIX PROPERTIES

Define the vector operator, F, which operates on a three dimensional vector, y, to form the skew-symmetric matrix:

$$F(\underline{v}) = \begin{bmatrix} 0 & -v_z & v_y \\ v_z & 0 & -v_x \\ -v_y & v_x & 0 \end{bmatrix} \tag{9}$$

The vector cross product of two vectors, $\underline{A}$ and $\underline{B}$ can the.n be defined as a matrix multiplication operation:

$$\underline{A} \times \underline{B} = F(\underline{A})\ \underline{B} \tag{10}$$

Additional properties include:

$$F(\underline{A})\underline{B} = -F(\underline{B})\underline{A}$$

$$and \tag{11}$$

$$F(\underline{A})F(\underline{B}) - F(\underline{B})F(\underline{A}) = F(\underline{A} \times \underline{B})$$

Further, a three axis small angle rotation matrix, relating two nearly aligned coordinate frames I and 2, can be define.d in terms of the skew-symmetric operator:

$${}^2T^1 = [I - F(\underline{\alpha})] \tag{12}$$

where, $\underline{\alpha}$, contains the misalignment angles about the x, y and z axes and ${}^2T^1$ transfers a vector from coordinate system I to 2.

4.4.2 ATTITUDE ERROR AND ATTITUDE ERROR GRADIENT

The difference between the actual satellite attitude and the estimated attitude is defined as the attitude error. The attitude error and its gradient are required to predict sensor outputs and to compute the measurement gradient as described later.

The attitude error vector, $\underline{\theta}$, can be found by manipulating the differential forms of the actual and estimated direction cosine matrices. The actual, T, and estimated, $\hat{T}$, direction cosine matrices, which transform vectors from the inertial to the body coordinate frame, are related to the actual, $\underline{\omega}$, and measured, $\hat{\underline{\omega}}$, rates as shown:

$$\dot{T} = -F(\underline{\omega})\ T \tag{13}$$

$$\dot{\hat{T}} = -F(\underline{\omega}_m)\ \hat{T} \tag{14}$$

Assume a small angle rotation, of the skew-symmetric form, relates the actual and estimated direction cosine matrices:

$$T = [I - F(\underline{\theta})]\ \hat{T} \tag{15}$$

Additionally, assume that the error between the actual rates and the measured rates due to gyro errors are related by a time varying matrix, Ω, and the parameter vector such that

$$\underline{\omega} = \underline{\omega}_m + \Omega\ \underline{x}_j \tag{16}$$

It can then be shown (eliminating products of small terms) that:

$$\dot{\underline{\theta}} \approx \underline{\theta} \times \underline{\omega}_m + \Omega \underline{x}_j = -F(\underline{\omega}_m)\underline{\theta} + \Omega \underline{x}_j \tag{17}$$

Obtain the attitude error gradient with respect to the parameter vector by partial differentiation of equation (17):

$$\left(\frac{\partial \dot{\underline{\theta}}}{\partial \underline{x}_j}\right) = -F(\underline{\omega}_m)\left(\frac{\partial \underline{\theta}}{\partial \underline{x}_j}\right) + \Omega \tag{18}$$

4.4.3 MEASUREMENT FUNCTION

Position Measurements from Line of Sight

To compute the measurement function, H_k, solar and stellar position measurements can be calculated by projecting the line of sight vector, from the satellite to the celestial body, onto the sensor focal plane.

Assume that the horizontal and vertical celestial position measurements are made in the sensor x-y plane and the sensor boresight is along the z axis. Then, designating the. horizontal and vertical focal plane positions as a and b respectively, the sensor focal length as, k_f, and the line of sight vector expressed in the sensor coordinate frame as, $\underline{A}^s$, the measurement function for a single sensor, H_k, becomes:

$$\begin{bmatrix} a \\ b \end{bmatrix} = H_k(\underline{A}^s) = \frac{k_f}{A_3^s}\begin{bmatrix} A_1^s \\ A_2^s \end{bmatrix} \tag{19}$$

Or in strict matrix form:

$$\begin{bmatrix} a \\ b \end{bmatrix} = k_f \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \end{bmatrix} \{[0\ 0\ 1]\, A^s\}^{-1} A^s \tag{20}$$

Line of Sight Vector in Sensor Frame

A^s is determined by transforming the inertial line of sight vector of the sun or star into to the sensor coordinate frame. The inertial line of sight vector itself, A^i, is determined either from solar/satellite ephemeris data or a star catalog and star identification number.

Six individual rotations are required for TOPEX. They include all of the known errors making up the error parameter vector:

$$A^s = {}^{s}T^{s'}\ {}^{s'}T^{b}\ {}^{b}T^{b'}\ {}^{b'}T^{o}\ {}^{o}T^{o'}\ {}^{o'}T^{i}\ A^i \tag{21}$$

The transformations in equation (21) are described below, from right to left. ${}^{o'}T^{i}$ rotates the inertial line of sight vector from inertial coordinates to the nominal initial satellite coordinate frame. ${}^{o}T^{o'}$ is a small angle, initial attitude error matrix relating the nominal initial satellite attitude to the actual initial attitude. ${}^{b'}T^{o}$ corresponds to the transformation from the actual initial attitude to a new arbitrary attitude as defined by propagation of the estimated quaternion. ${}^{b}T^{b'}$ is another small angle matrix. It represents the attitude error between the estimated body frame and the actual body frame. ${}^{s'}T^{b}$ is the fixed nominal rotation from the body frame to the sensor (sun or star tracker) frame. Finally, ${}^{s}T^{s'}$ is the sensor's small angle misalignment matrix from its nominal to its actual position.

Replacing the three small angle matrices in equation (21) by equivalent skew-symmetric forms yields:

$$\begin{aligned} A^s = [I - F(c_s x_j)]\ {}^{s'}T^{b}\ [I - F(\theta)]\ \cdot \\ {}^{b'}T^{o}\ [I - F(c_o x_j)]\ {}^{o'}T^{i}\ A^i \end{aligned} \tag{22}$$

where x_j is the error parameter vector, c_s is a constant matrix which selects out the appropriate sensor misalignment angles from the parameter vector, θ is the attitude error vector and c_o is another constant matrix which selects out the initial attitude errors from x_j.

Successive expansions of equation (22) and elimination of second-order terms produces:

$$\begin{aligned} A^s \approx &+ {}^{s'}T^{b}\ {}^{b'}T^{o}\ {}^{o'}T^{i}A^i \\ &+ F({}^{s'}T^{b}\ {}^{b'}T^{o}\ {}^{o'}T^{i}A^i)\ c_s\ x_j \\ &+ {}^{s'}T^{b}\ F({}^{b'}T^{o}\ {}^{o'}T^{i}A^i)\ \theta \\ &+ {}^{s'}T^{b}\ {}^{b'}T^{o}\ F({}^{o'}T^{i}A^i)\ c_o\ x_j \end{aligned} \tag{23}$$

Measurement Function Gradient

The measurement function gradient with respect to the error parameter vector is now determined using these expressions. First note that the attitude error vector, θ, is itself a function of the parameter vector as derived in equation (17). Thus, the chain rule must used to find the gradient matrix:

$$\frac{\partial H_k}{\partial x_j} = \frac{\partial H_k}{\partial A^s}\cdot\frac{dA^s}{dx_j} = \frac{\partial H_k}{\partial A^s}\left[\frac{\partial A^s}{\partial x_j} + \frac{\partial A^s}{\partial \theta}\cdot\frac{\partial \theta}{\partial x_j}\right] \tag{24}$$

Combining previous results and differentiating:

$$\frac{\partial H_k}{\partial A^s} = k_f \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \end{bmatrix} \{-\{[0\ 0\ 1]\, A^s\}^{-2} A^s\, [0\ 0\ 1] + [0\ 0\ 1]\, A^s\}^{-1} I\}$$

$$\frac{\partial A^s}{\partial x_j} = F({}^{s'}T^{b}\ {}^{b'}T^{o}\ {}^{o'}T^{i}A^i)\ c_s + {}^{s'}T^{b}\ {}^{b'}T^{o}F({}^{o'}T^{i}A^i)\ c_o \tag{25}$$

$$\frac{\partial A^s}{\partial \theta} = {}^{s'}T^{b}\ F({}^{b'}T^{o}\ {}^{o'}T^{i}A^i)$$

The measurement gradient in equation (24) is now readily calculated from the constituents in equation (25) and the integration of the attitude gradient differential equation in equation (18). This measurement gradient is for a single celestial sensor.

4.5 ALGORITHM SUMMARY

Collecting all the results, the steps for completing the algorithm are summarized below:

Between Measurements:

1) Propagate the quaternion, $\hat{q}$, according to equation (1) using the measured rate gyro outputs and error parameter vector, x_j.
2) Propagate the attitude error gradient, $\frac{\partial \theta}{\partial x_j}$, according to equation (18).

Incorporate Measurements at Observation Time:

1) Calculate the valid celestial line of sight vectors, A^i, in the ECI coordinate frame from solar/satellite ephemeris data or from a star catalog by looking up on the star identification number.
2) Predict the celestial line of sight vectors, A^s, in the sensor coordinate frame using equation (21) and estimated

transformation matrices.

3) Compute the predicted sensor outputs, $\hat{z}_{m_k}$, from equation(19).

4) Compute the measurement gradient matrix, $\frac{\partial H_k}{\partial x_j}$, using (24) and (25).

5) Update the gain matrix, parameter increment covariance and parameter increment according to equations (5)-(7).

Repeat the above steps over the entire calibration data segment. Update the parameter vector in the outer loop and repeat for each celestial measurement.

4,6 MATRIX DEFINITIONS

The 24 x 1 clement error parameter vector is defined as:

$x_j =$

ϕ_o	initial roll attitude error
θ_o	initial pitch attitude error
ψ_o	initial yaw attitude error
sfe_x	roll gyro rate scale factor error
sfe_y	pitch gyro rate scale factor error
sfe_z	yaw gyro rate scale factor error
b_x	roll rate bias
b_y	pitch rate bias
b_z	yaw rate bias
$\Delta\phi$	gyro axes roll misalignment
$\Delta\theta$	gyro axes pitch misalignment
$\Delta\psi$	gyro axes yaw misalignment(31)
yzg_x	pitch or yaw gyro non-orthogonality about roll axis
xzg_y	roll or yaw gyro non-orthogonality about pitch axis
xyg_z	roll or pitch gyro non-orthogonality about yaw
ϕ_{SS}	sun sensor angular misalignment about x axis
θ_{SS}	sun sensor angular misalignment about y axis
ψ_{SS}	sun sensor angular misalignment about z axis
ϕ_{ST1}	star tracker 1 angular misalignment about x axis
θ_{ST1}	star tracker 1 angular misalignment about y axis
ψ_{ST1}	star tracker I angular misalignment about z axis
ϕ_{ST2}	star tracker 2 angular misalignment about x axis
θ_{ST2}	star tracker 2 angular misalignment about y axis
ψ_{ST2}	star tracker 2 angular misalignment about z axis

The 3 x 24 time varying matrix which when multiplied by x_j, produces the angular velocity error is defined as:

$$\Omega = \begin{bmatrix} 0 & 0 & 0 & \omega_{x_m} & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & \omega_{y_m} & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & \omega_{z_m} & 0 & 0 & 1 \end{bmatrix} \cdots$$

$$\cdots \begin{bmatrix} 0 & \omega_{z_m} & -\omega_{y_m} & 0 & -\omega_{z_m} & \omega_{y_m} & 0 & . & 0 \\ -\omega_{z_m} & 0 & \omega_{x_m} & \omega_{z_m} & 0 & \omega_{x_m} & 0 & . . & 0 \\ \omega_{y_m} & -\omega_{x_m} & 0 & \omega_{y_m} & -\omega_{x_m} & 0 & 0 & \cdots & 0 \end{bmatrix}$$

5 SOFTWARE IMPLEMENTATION

The Attitude Sensor Calibration Program Set (ASCAL) consists of multiple programs integrated within the TOPEX Guidance and Control ground operations software system (GCFS) which is implemented on the VAX 6410 mainframe computer. This software system provides an integrated environment and uniform user interface for execution of the various application program sets including ASCAL. In addition, it also provides the functions of data base management, input/output file archival and management as well as a variety of utility tools for the implementation of these functions.

The user interface is menu driven with data entry screens provided for user specified inputs. The menu screens are designed in a hierarchical architecture for case of understanding and selection of the desired operation. Data entry screens arc employed for specifying input values for program execution by the user. Values retrieved from a defaulted file arc already incorporated in the data entry screens to minimize the amount of editing required. On-screen description of the selected input variable is presented along with the valid range and data type. Each input is checked for proper data type and range before it is accepted. For enumerated variable.s, the possible inputs arc specified in a sub-menu for case of selection. If the input variable is an input file name, the corresponding file will be checked to ascertain its availability within the computer system. Otherwise, the input will be rejected.

A central data base is implemented for storing parameters which arc common and used by the various programs. Accessing or updating records in the data base arc transparent to the user. A variety of data base management tools arc also provided such as browsing, printing, initializing and selective updates. In order to avoid inadvertently corrupting the data base, only the system administrator with the appropriate priority is allowed to access the tools for modifying the data base contents.

Another essential feature of this environment is the archival capability of the input/output files for each execution session. The files utilized for each session arc automatically archived in separate folders upon execution. This feature allows ease of re-execution of a previous session with minor modification to the previous set-up. All output files arc also archived for later review or hardcopy generation.

The basic menu provided for ASCAL is shown in figure 5. The first step in this process is the retrieval of telemetry data required for calibration. This is accomplished by executing a telemetry processing program (TLMPRO) which accesses the primary telemetry data file to retrieve the appropriate channel data (gyros, star tracker, sun sensor and spacecraft attitude) for the period of interest, Additional processing of the raw data is performed such as unit conversion, calculation of the celestial vector in an inertial frame (J2000) from the X -Y components in the sensor frame, curve fitting of the gyro incremental position data and evaluating the numerically differentiated rates. Primary outputs of this program arc three binary files containing the spacecraft attitude quaternion, gyro rate data for the three axes, and celestial sensor data. These files arc generated in a format compatible with the calibration program. To ascertain the quality of tbc telemetry data prior to calibration, it is required to review the data manually either by plots or tabulated display. The intention of this process is to avoid data gaps, data hits, out of range data, etc which may corrupt the performance of the calibration process. A program is thus provided [o convert [hc binary files into ASCII format which can bc viewed directly on-screen or to generate hardcopy, A generic plotting program is also availability for viewing the overall pattern of the gyro data for consistency. This ASCII file can also be processed by a Word Processing program to perform data edition, This is not recommended as a routine procedure. However, the utility [001 is available if necessary. Once the telemetry data file has been edited, it mus[be converted back to a binary format for processing by [be. calibration program. A corresponding routine is also available from the menu selection for this function.

The core of ASCAL is the calibration program which utilizes the binary telemetry files as input. Detailed algorithm design is presented in the previous section. For initialization, the last estimated error parameters arc retrieved from the central data base 10 start the current calibration. All pertinent data such as covariances, error estimates, residuals, etc. can be saved for plotting. This is essential for evaluating the accuracy and quality of the estimates. Upon completion of the calibration process, a utility tool is provided to update the central data base for the latest error estimates and covariances. These data can be retrieved for trend analysis after multiple calibrations have been performed over a period of twelve months. It is intended to perform Ibis calibration process on a bi-monthly basis throughout the mission. Attitude reconstruction is also a requirement of ASCAL. This program propagate.s the gyro data, using the latest estimates of the gyro errors to generate a reconstructed spacecraft attitude file. This file is compared against the on-board attitude estimates obtained from telemetry. Statistical analysis of tbc differences is performed to evaluate the performance of the on-board attitude determination process.

6 CALIBRATION RESULTS

Following launch and after orbit adjustment maneuvers, TOPEX exhibited time varying off-nadir pointing errors symptomatic of an un-calibrated spacecraft. The pointing error was large (-.3), as measured by radar data and collaborated by the horizon sensors, and rendered any science data useless which requires stable, accurate pointing.

Using the ASCAL program set, sensor error parameters were determined using flight data from 3 consecutive orbits on days 271 and 303. The calibration results for each of the 3 orbits for each error parameter (X, Y and Z axes) and for both days arc shown in figures 6-12. The vertical lines indicate the estimated parameter and its' 3σ error bounds for each calibrated orbit. The horizontal lines indicate the parameter values which were actually uploaded to the spacecraft on 8 Dec 92. These also arc the ground measured values for the gyro parameters as they were not modified. The overlap in the error bounds as well as the consistency from day 271 to 303 indicate reasonable results. Note that the gyro biases and scale factor errors shifted little from their pre-launch measured values. In contrast, the ASTRAs' and sun sensor misalignment appear to be quite large relative to the ground measurements (zero on the plot). The horizontal line.s on these plots represent the 8 Dec 92 uploads.

On 8 Dec 92 calibration parameters were uploaded to the TOPEX spacecraft, Following the parameter upload, on-board attitude estimation residuals reduced substantially from about 7 mini-radian extrema down to 1 milli-radian as shown in figure 13. Subsequent radar and horizon sensor measurements confirmed near elimination of the time varying pointing error. Pointing error biases were later corrected by commanding a bias quaternion.

7 SUMMARY

An algorithm and software program system is presented which was used to calibrate attitude determination and control system sensors on the TOPEX spacecraft. Parameters determined by ASCAL were uploaded to the spacecraft and improved overall attitude pointing by a factor of 7.

ACKNOWLEDGEMENTS

This work was performed at the Jet Propulsion Laboratory, California Institute of Technology, under contract with the

National Aeronautics and Space Administration.

The spacecraft coordinate systems and yaw steering profile drawings are extracted from documents originated from Fairchild Space Co., the prime contractor of the TOPEX project

The authors would like to thank the following for their substantial contribution to the development of the algorithm and software: W. Brekenridge, K. Hioc, T. Kia and M. San Martin.

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2) Dennehy, C. J., R. Welch, T. Kia, "Attitude Determination and control Subsystem for the TOPEX Satellite", AIAA Paper 88-4129, August 1988.

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4) Modular Attitude Control System Specification, Fairchild Space Company Document 968-PF1011, March 1989.

5) Widnall, W. S., Grundy, P. A., "Inertial Navigation System Error Models", Intermetrics, Inc., TR-03-73. Project 688G, contract #F29601-72-C-0162. 6585th Test Group, Air Force Special Weapons, Center, Holloman AFB, New Mexico 88330.

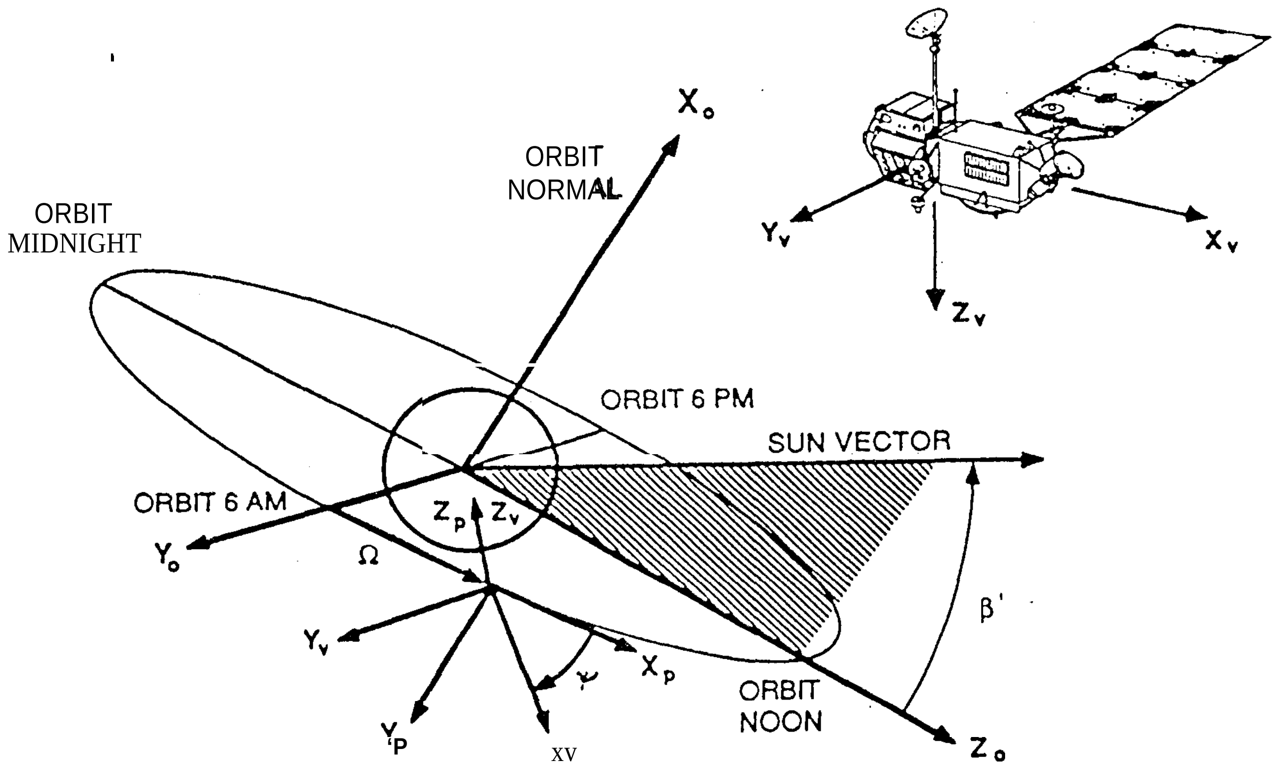

FIGURE 1. Illustration of Coordinate Systems

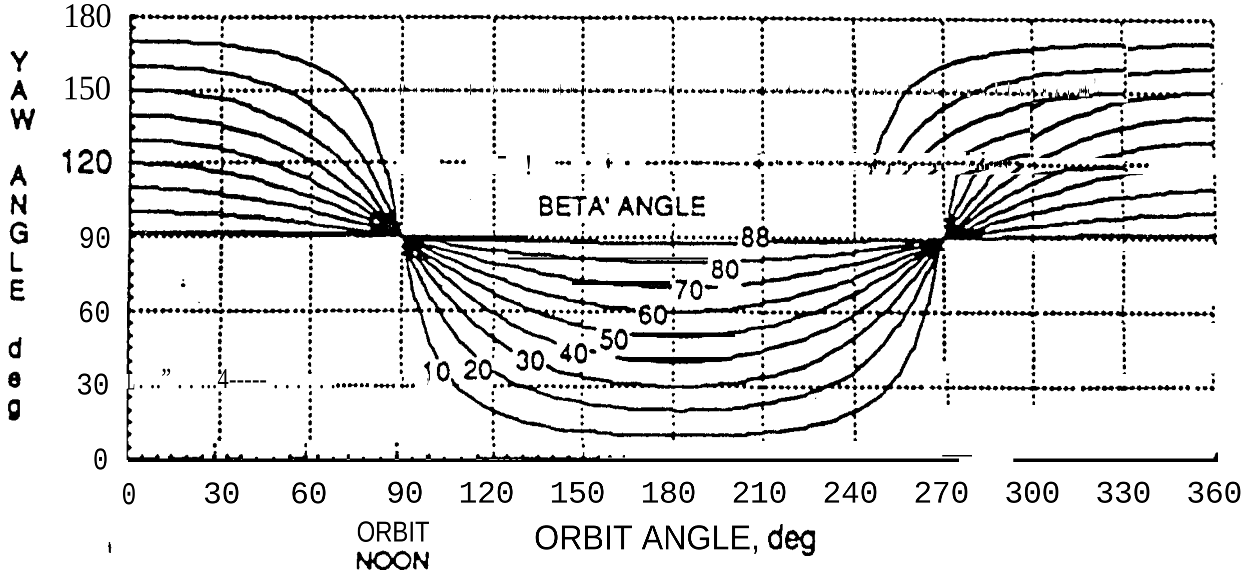

FIGURE 2. Satellite Perfect Yaw angle as a Function of Orbit Angle and Sun Angle, β'

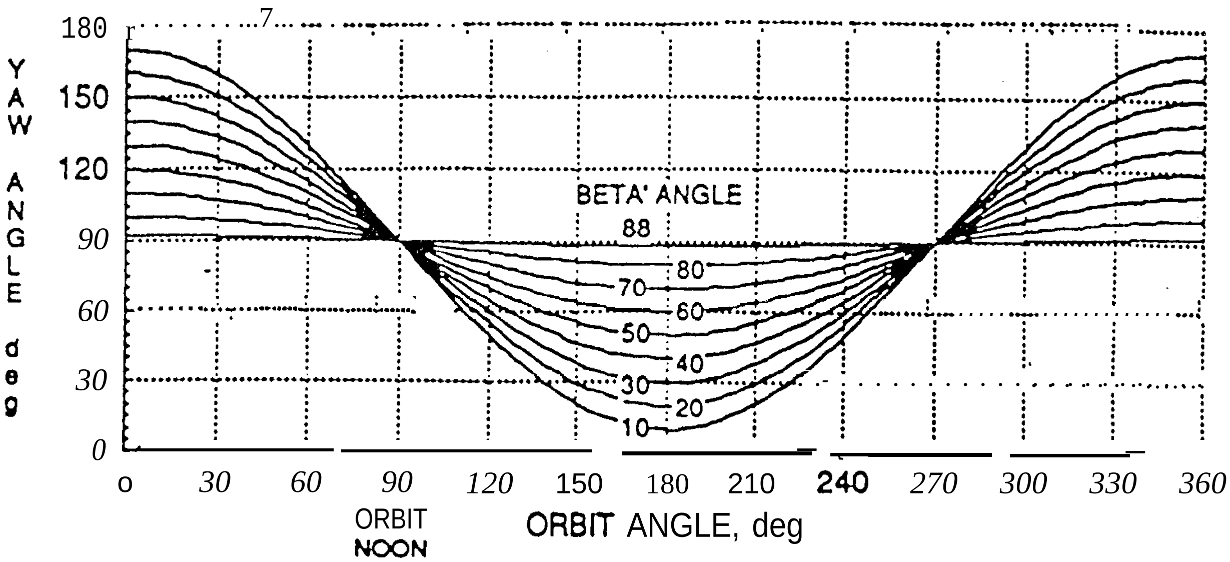

FIGURE 3. Spacecraft Sinusoidal Yaw Angle as a Function of Orbit Angle and Sun Angle β'

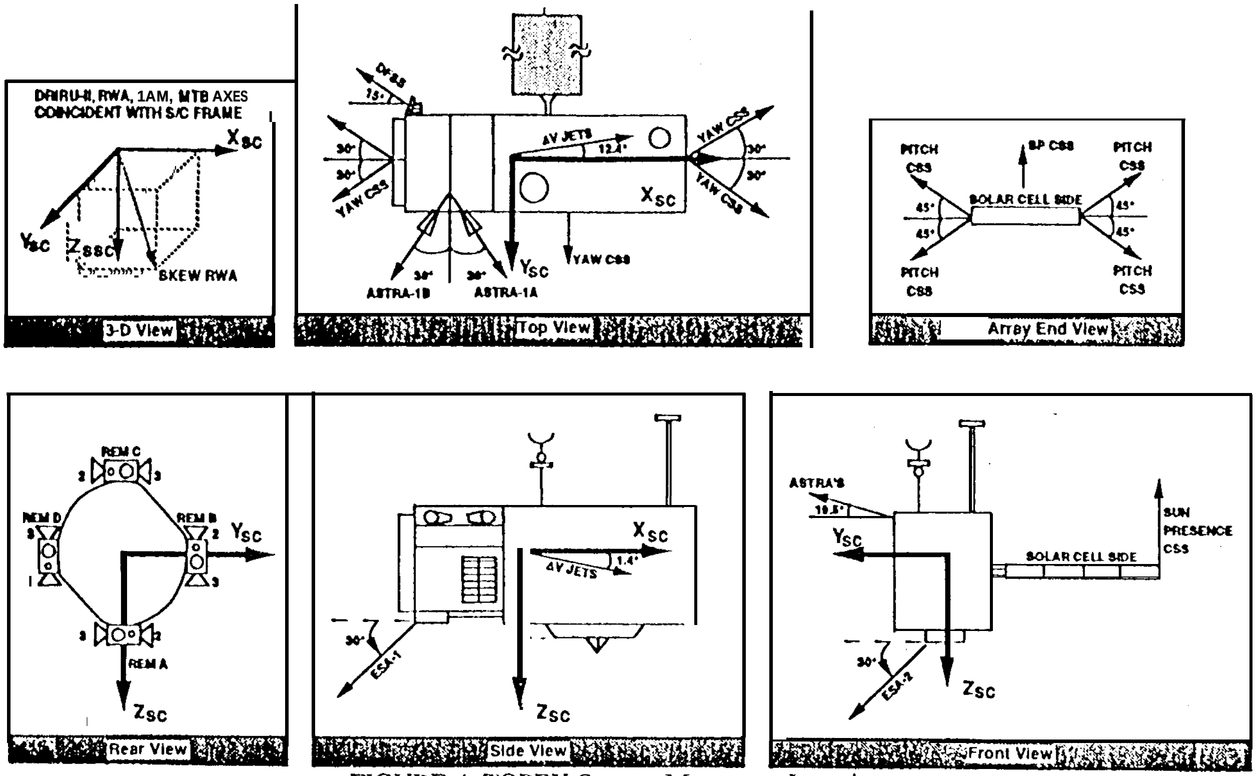

FIGURE 4. TOPEX Sensor Moun ng Locations

Commands Edit Customize

Attitude Sensor Calibration Program File Set
(ASCAL MENU)

1. Telemetry Processing. Note: Press return.

2. Attitude Calibration Program.

3. Attitude Reconstruction Program.

4. Plot/Graph data program.

5. File Editor/Browser.

6. Print file.

7. Update Database.

Select this option to do telemetry processing for ASCAL program.

FIGURE 5. Attitude Sensor Calibration Program File Set Menu

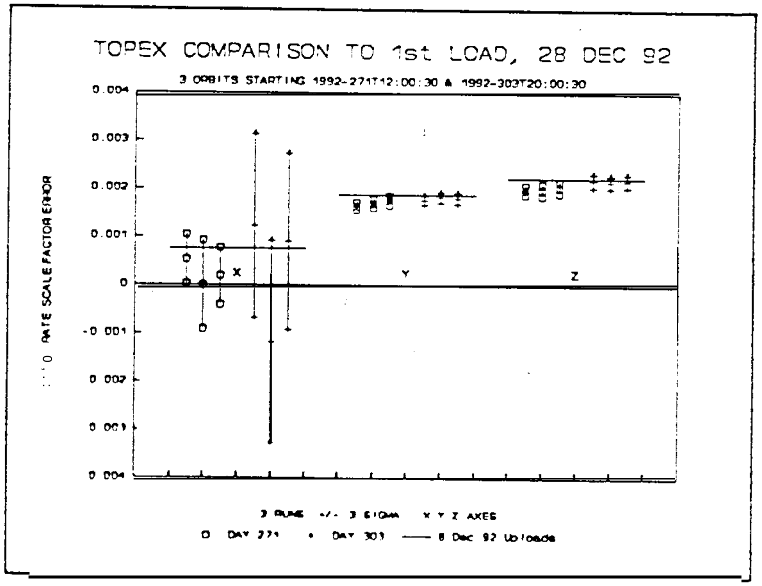

FIGURE 6. Gyro Rate Scale Factor

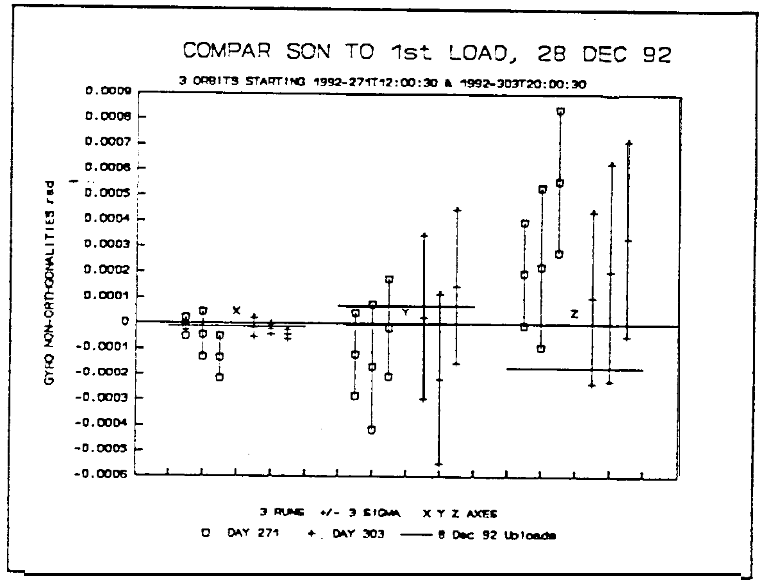

FIGURE 8. Gyro Non-Orthogonalities

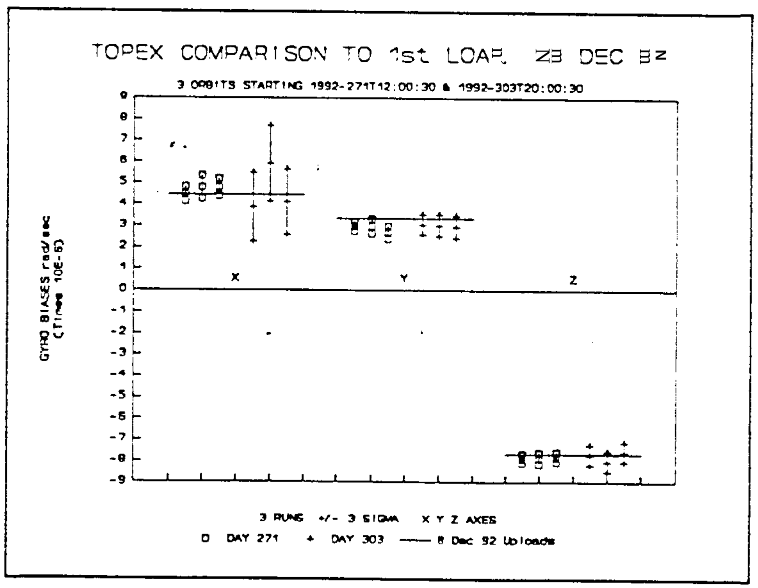

FIGURE 7. Gyro Rate Biases

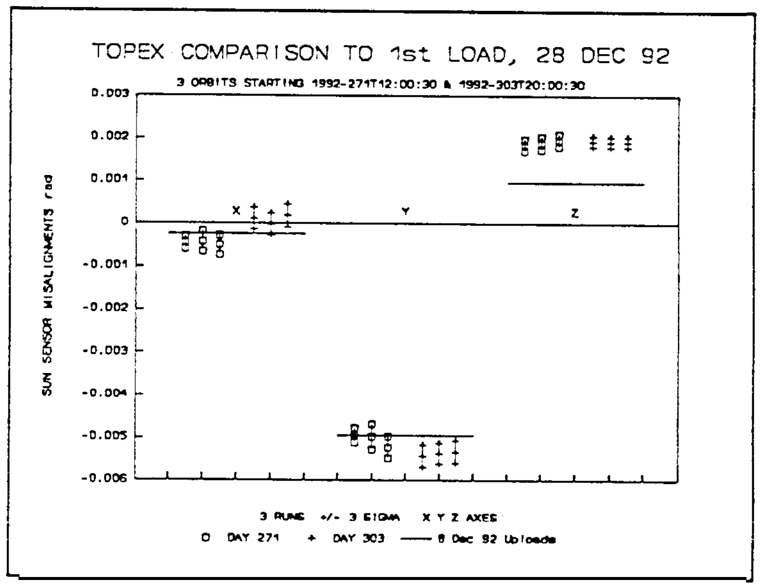

FIGURE 9. Sun Sensor Misalignments

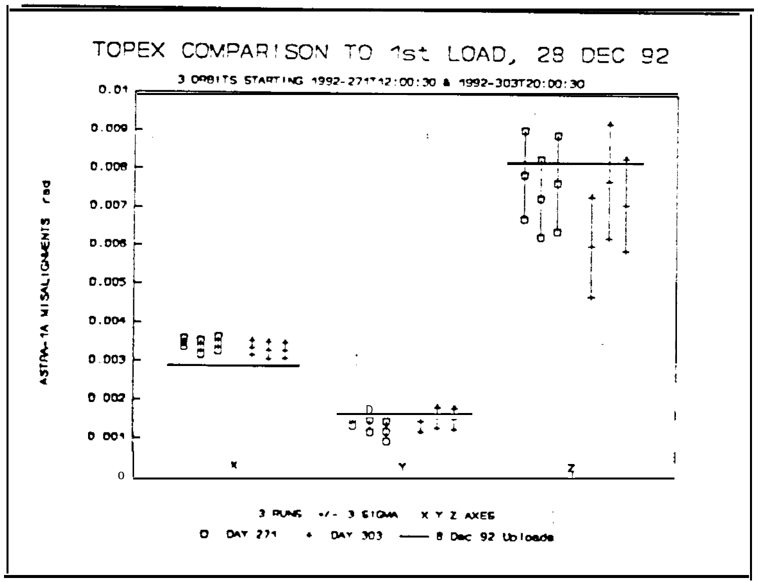

FIGURE 10. ASTRA-1A Misalignments

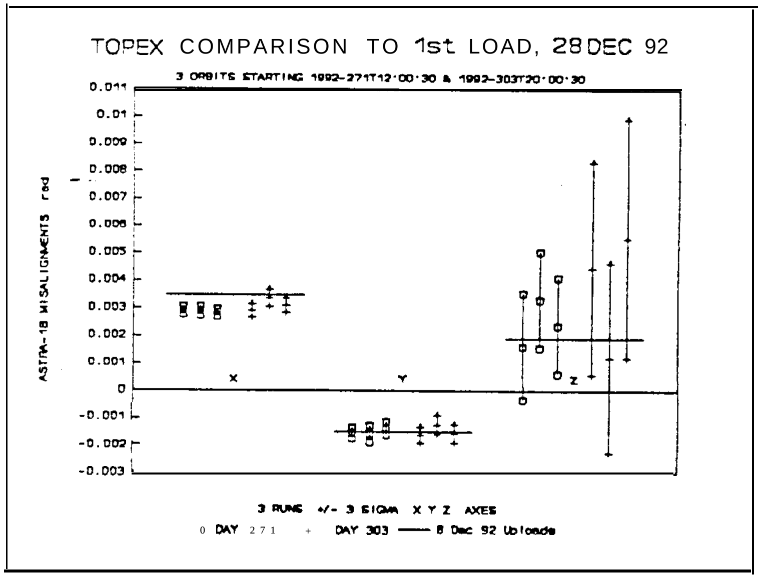

FIGURE 11. ASTRA-1B Misalignments

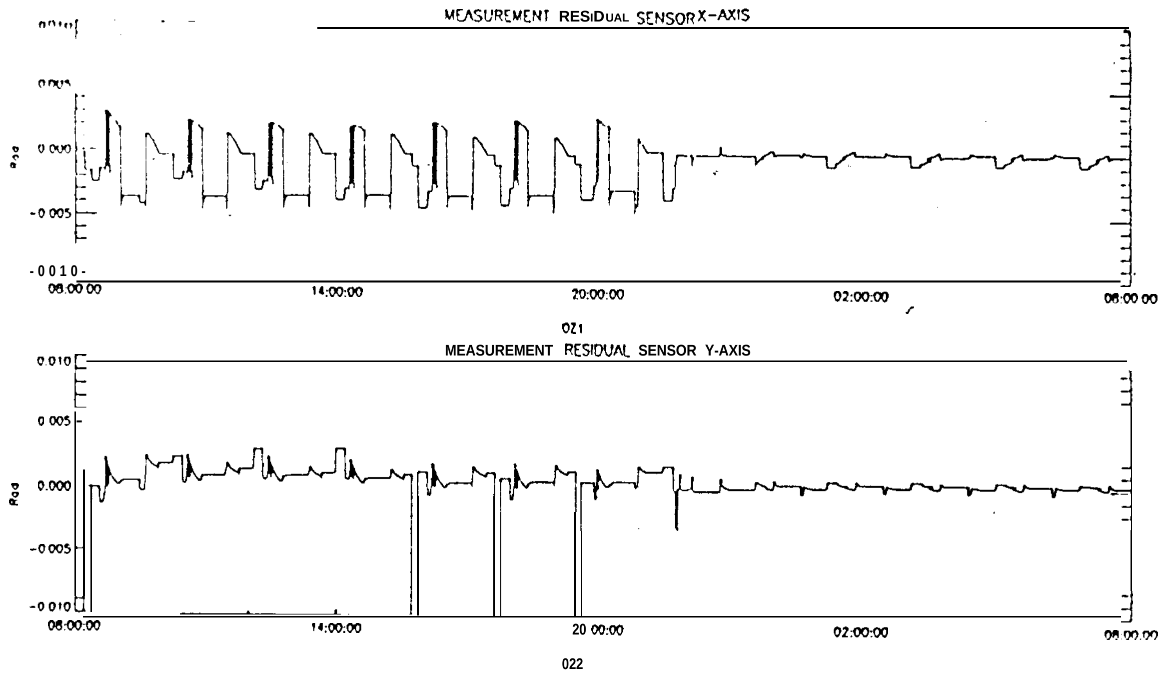

FIGURE 12. On-Board Attitude Measurement Residuals

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