16. REAL-TIME ATTITUDE DETERMINATION WITH ZERO GYROS

In response to the continuing deterioration of all gyros, the zero-gyro operations and software were developed, in principle allowing the real-time attitude control to function with perturbing torque models in place of the gyro data. Catastrophic failure of the telecommunications subsystem terminated the mission before this mode was used. The procedures used to handle these contingencies are nevertheless described.

16.1. Activities during Suspended Operations

On 18 March 1993, the total failure of gyro 2 prompted an emergency sun reacquisition and a spin-up to 0.25 rpm of the spacecraft. This state was maintained until 11 June 1993, when attempts began to operate under the zero-gyro real-time attitude determination control. There were two principal reasons for the delay:

(1) the software for on-board and on-ground real-time attitude determination without gyros was still under development. However, operations resumed when sufficient (if not optimised) ground software was in place for support;

(2) there were concerns that solar panel degradation was too great to allow the spacecraft to be fully operated during the long eclipse season (ending on 21 May 1993). In practice, the degradation was less than that predicted.

The return to full operations was further hampered on 5 and 6 May 1993 by the discovery that both star mappers were partially damaged (no output on the B_T channel of star mapper 1 and the V_T channel of star mapper 2). The only operational solution was to use two different telemetry formats. The first (prepared on-ground and loaded on-board) combined both star mapper outputs but at the cost of the image dissector tube data. This format was required for use in the ground real-time attitude determination system to perform initial star pattern recognition. The second (available on-board) was combining star mapper 2 and image dissector tube 2 data for use once real-time attitude determination convergence was achieved and good science had to be collected. Because the ground software had to work with both formats, a further re-design was necessary.

Event History during Suspended Operations

The progressive failures occurring during the last months of attempted operations were attributed to radiation damage of the associated electronics. A summary of the problems encountered during the period in which zero-gyro operations were attempted is given hereafter, with further details of the various anomalies given in Appendix B.

93-03-18: Gyro 2 failure (Anomaly 77): the on-board system automatically triggered an emergency sun reacquisition.

93-03-19: The spacecraft was spun-up to 0.25 rpm and the gyros switched off to save power. Nutation damping was then performed.

93-03-23: The decision was taken to suspend spacecraft operations. The payload was switched off, with the thermal control electronics branch 2 left on.

93-03-24: Thermal control electronics 2 was configured to wide range.

93-03-28: The nominal antenna switching command failed (Anomaly 78).

93-03-30: All thermal control electronics 2 heaters were set to fixed power with value zero, i.e. switched off, due to power constraints.

93-04-01: Both antenna switching commands failed.

93-04-05: Commands 62 and 65 were used to switch antennas.

93-04-23: Due to the orbital rotation of the Earth around the Sun, the solar aspect angle of the spin axis had been steadily increasing at approximately 1°/day. Power efficiency of the solar panels was highest near the nominal 43° solar aspect angle. This prompted the design and implementation of a series of manoeuvres to reduce the solar aspect angle, which was already larger than 43°, without de-spinning the satellite. This was performed by direct thruster commanding, using the sun acquisition sensor as guidance. Nutation damping was also performed at this time.

93-04-27: The increased efficiency of the panels at 43° combined with the decreasing eclipse durations, allowed the payload thermal control electronics 1 and 2 to be switched on and tested. Consequently, the thermal control electronics 2 heaters were set to fixed power value 15.

93-05-04: The payload was switched on. A preliminary check showed that the image dissector tubes were undamaged by the suspended operations.

93-05-05: Star mapper 2 V_T channel had ceased to function (Anomaly 79).

93-05-06: Star mapper 1 B_T channel had ceased to function (Anomaly 79).

93-05-10: Further manoeuvres were performed to reduced the solar aspect angle and perform nutation damping.

93-05-11: The control law electronics branch A failed (Anomaly 80). The switch to the redundant branch B was made.

93-05-12: A new telemetry was introduced (format 4) combining the star mapper 1 and star mapper 2 telemetry.

93-05-13: Further manoeuvres were performed to reduced the solar aspect angle and perform nutation damping.

93-05-14: The battery discharge regulator 1 module 2 overvoltage status changed.

93-05-17: Further manoeuvres were performed to reduced the solar aspect angle and perform nutation damping.

93-05-24: The eclipse season ended. Gyros 1 and 4 were switched on to increase the gyro electronics box temperature. No usable output was obtained.

93-05-25: Further manoeuvres were performed to reduced the solar aspect angle and perform nutation damping. Gyros 2 and 5 were switched on to increase the gyro electronics box temperature. No usable output was obtained.

93-05-26: Gyro 3 was switched on but within a few hours was showing very high noise (Anomaly 81). The gyro was declared unusable and switched off.

93-05-27: The zero-gyro real-time attitude determination software patch was loaded on-board.

93-06-02: Further manoeuvres were performed to reduced the solar aspect angle and perform nutation damping. The new sun acquisition manoeuvre 4 software patch was loaded on-board.

93-06-04: Further tests were performed with gyros 1, 3 and 5.

93-06-05: The inertial reference unit was completely switch-off.

93-06-06: The attitude control software crashed when the new sun acquisition manoeuvre 4 software was enabled. The problem was traced to an invalid checksum and corrected.

93-06-09: Further manoeuvres were performed to reduced the solar aspect angle and perform nutation damping.

93-06-11: Prior to the resumption of scientific operations, the spacecraft was de-spun from 0.25 rpm to 0.15 rpm.

93-06-12: The spacecraft was further de-spun to 0.05 rpm, and an emergency sun reacquisition was commanded from ground to bring the spacecraft to the safe sun-pointing mode. Operations with the zero-gyro real-time attitude determination system began.

16.2. On-Board Software

Zero-gyro Real-Time Attitude Determination

A full description of the design concepts of real-time attitude determination was given in Chapter 14. Although the basic concepts of filtering were the same in three-, two- and zero-gyro systems (see Figure 14.2), the zero-gyro real-time attitude determination system constituted a complete re-design of the on-board real-time attitude determination.

The body rates derived from gyro measurements were replaced on all three axes by angular accelerations which were integrated using discretised perturbation equations on-board. The result of these calculations was an on-board attitude state vector containing Tait-Bryan attitude error angles and body rate estimates which were corrected using Kalman filter techniques by the star mapper measurements. Further parameters were introduced to account for thruster misalignments and differences in performance.

As with the two-gyro system, the design had to take into account the limited memory space for the on-board software.

A critical new component of the system was an on-board acceleration model for all three spacecraft axes. Although the two-gyro system could work with a sinusoidal on-board disturbance torque model (see Chapter 15), the disturbance torques around the x and y axes were far from sinusoidal. The most accurate solution possible was to compute the predicted angular accelerations arising from external disturbance torques by using empirical tables of points spanning on average two hours and refreshed by ground command (two hours was the maximum allowable given the memory space and the resolution of points required). Each table was capable of being refreshed in two halves thereby avoiding discontinuities when the tables were reset. The on-board angular acceleration tables were computed on-ground from predicted disturbance torques, calibrated from the most recent history of the attitude.

The following revised state vector had to be initialised using the modified ground real-time attitude determination system:

- ϕ: Tait-Bryan error angle around the spacecraft x-axis
- θ: Tait-Bryan error angle around the spacecraft y-axis
- ψ: Tait-Bryan error angle around the spacecraft z-axis
- $\dot{\phi}$: error rate around the spacecraft x-axis
- $\dot{\theta}$: error rate around the spacecraft y-axis
- $\dot{\psi}$: error rate around the spacecraft z-axis
- γ_x: angular acceleration around the spacecraft x-axis
- γ_y: angular acceleration around the spacecraft y-axis
- γ_z: angular acceleration around the spacecraft z-axis

Each component of the state vector v was broken down into a predicted (deterministic) component v_d taking the spacecraft dynamics into account, and the other was a corrective term v_1 taking into account uncertainties in the modelling:

(a) nominal scanning law rates ω_X, ω_Y, ω_Z were computed as for the three-gyro real-time attitude determination (see Equation 8.14);

(b) the deterministic components of the angular accelerations ($\gamma_{d,x}$, $\gamma_{d,y}$, $\gamma_{d,z}$) were derived from the on-board acceleration tables, via quadratic interpolation based on the neighbouring points at the time of computation;

(c) the other deterministic components of the state vector, namely the Tait-Bryan error angles (ϕ_d, θ_d, ψ_d) and error rates ($\dot{\phi}_d$, $\dot{\theta}_d$, $\dot{\psi}_d$) were propagated forward from time t_k to time t_{k+1}. Defining the time step $t_i = 16/15$ s, then the propagation equations were:

$$\begin{pmatrix} \phi_d(k+1) \\ \theta_d(k+1) \\ \psi_d(k+1) \\ \dot{\phi}_d(k+1) \\ \dot{\theta}_d(k+1) \\ \dot{\psi}_d(k+1) \end{pmatrix} = \begin{pmatrix} 1 & \omega_Z\ t_i & 0 & t_i & (1+\Gamma_1)\omega_Z\ t_i^2/2 & 0 \\ -\omega_Z\ t_i & 1 & 0 & -(1+\Gamma_1)\omega_Z\ t_i^2/2 & t_i & 0 \\ 0 & 0 & 1 & 0 & 0 & t_i \\ 0 & 0 & 0 & 1 & \Gamma_1\ \omega_Z\ t_i & 0 \\ 0 & 0 & 0 & \Gamma_2\ \omega_Z\ t_i & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 \end{pmatrix} \begin{pmatrix} \phi_d(k) \\ \theta_d(k) \\ \psi_d(k) \\ \dot{\phi}_d(k) \\ \dot{\theta}_d(k) \\ \dot{\psi}_d(k) \end{pmatrix}$$
$$+ \begin{pmatrix} \gamma_{d,x}\ t_i^2/2 - \omega_X\ t_i \\ \gamma_{d,y}\ t_i^2/2 - \omega_Y\ t_i \\ \gamma_{d,z}\ t_i^2/2 \\ \gamma_{d,x}\ t_i \\ \gamma_{d,y}\ t_i \\ \gamma_{d,z}\ t_i \end{pmatrix} \qquad [16.1]$$

(d) the correction components of the state vector, namely the Tait-Bryan error angles $(\phi_1,\ \theta_1,\ \psi_1)$, error rates $(\dot{\phi}_1,\ \dot{\theta}_1,\ \dot{\psi}_1)$ and angular accelerations $\gamma_{1,x},\ \gamma_{1,y},\ \gamma_{1,z}$ were propagated forward in a similar manner to above:

$$\begin{pmatrix} \phi_1(k+1) \\ \theta_1(k+1) \\ \psi_1(k+1) \\ \dot{\phi}_1(k+1) \\ \dot{\theta}_1(k+1) \\ \dot{\psi}_1(k+1) \\ \gamma_{1,x}(k+1) \\ \gamma_{1,y}(k+1) \\ \gamma_{1,z}(k+1) \end{pmatrix} = \qquad [16.2]$$
$$\begin{pmatrix} 1 & \omega_Z\ t_i & 0 & t_i & (1+\Gamma_1)\omega_Z\ t_i^2/2 & 0 & t_i^2/2 & 0 & 0 \\ -\omega_Z\ t_i & 1 & 0 & -(1+\Gamma_1)\omega_Z\ t_i^2/2 & t_i & 0 & 0 & t_i^2/2 & 0 \\ 0 & 0 & 1 & 0 & 0 & t_i & 0 & 0 & t_i^2/2 \\ 0 & 0 & 0 & 1 & \Gamma_1\ \omega_Z\ t_i & 0 & t_i & \Gamma_1\omega_Z\ t_i^2/2 & 0 \\ 0 & 0 & 0 & -\Gamma_2\ \omega_Z\ t_i & 1 & 0 & -\Gamma_2\ \omega_Z\ t_i & t_i & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 & t_i \\ 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 \end{pmatrix} \begin{pmatrix} \phi_1(k) \\ \theta_1(k) \\ \psi_1(k) \\ \dot{\phi}_1(k) \\ \dot{\theta}_1(k) \\ \dot{\psi}_1(k) \\ \gamma_{1,x}(k) \\ \gamma_{1,y}(k) \\ \gamma_{1,z}(k) \end{pmatrix}$$

where:

$$\Gamma_1 = \frac{I_{yy} - I_{zz}}{I_{xx}}$$
$$\Gamma_2 = \frac{I_{xx} - I_{zz}}{I_{yy}} \qquad [16.3]$$

are ratios involving the principal spacecraft moments of inertia $(I_{xx},\ I_{yy},\ I_{zz})$;

(e) corrections for actuations are performed on the vector $(\phi_d,\ \theta_d,\ \psi_d,\ \dot{\phi}_d,\ \dot{\theta}_d,\ \dot{\psi}_d)$ using:

$$\phi'_d = \phi_d + \lambda_x(T_{ax}\delta_{tx} + T_{ax}\ t_{\mathrm{on},x})(1 - \tfrac{1}{2}(t_{\mathrm{on},x} + \delta_{tx}))$$
$$+ \lambda_{xz}(T_{az}\delta_{tz} + T_{az}\ t_{\mathrm{on},z})(1 - \tfrac{1}{2}(t_{\mathrm{on},z} + \delta_{tz}))$$
$$\theta'_d = \theta_d + \lambda_y(T_{ay}\delta_{ty} + T_{ay}\ t_{\mathrm{on},y})(1 - \tfrac{1}{2}(t_{\mathrm{on},y} + \delta_{ty}))$$
$$+ \lambda_{yz}(T_{az}\delta_{tz} + T_{az}\ t_{\mathrm{on},z})(1 - \tfrac{1}{2}(t_{\mathrm{on},z} + \delta_{t_z}))$$

$$
\begin{aligned}
\psi_d' &= \psi_d + \lambda_z(T_{az}\delta_{tz} + T_{az}\ t_{\mathrm{on},z})(1 - \tfrac{1}{2}(t_{\mathrm{on},z} + \delta_{tz})) \\
\dot{\phi}_d' &= \dot{\phi}_d + \lambda_x(T_{ax}\delta_{tx} + T_{ax}\ t_{\mathrm{on},x}) + \lambda_{xz}(T_{az}\delta_{tz} + T_{az}\ t_{\mathrm{on},z}) \\
\dot{\theta}_d' &= \dot{\theta}_d + \lambda_y(T_{ay}\delta_{ty} + T_{ay}\ t_{\mathrm{on},y}) + \lambda_{yz}(T_{az}\delta_{tz} + T_{az}\ t_{\mathrm{on},z}) \\
\dot{\psi}_d' &= \dot{\psi}_d + \lambda_z(T_{az}\delta_{tz} + T_{az}\ t_{\mathrm{on},z})
\end{aligned} \qquad [16.4]
$$

where:

- T_{ax}, T_{ay}, T_{az} are the calibrated accelerations of the $+x$, $+y$, and $+z$ thrusters respectively;
- λ_x, λ_y, λ_z are set to 1 if the actuation is on a positive thruster. If the actuation occurs on the negative thruster, $\lambda = T_a(-)/T_a(+)$ (the ratio of the negative thruster to the positive thruster);
- λ_{xy}, λ_{yz} are corrective coupling terms;
- δ_{tx}, δ_{ty}, δ_{tz} are thruster time offsets which are independent of the length of any actuation;
- $t_{\mathrm{on},x}$, $t_{\mathrm{on},y}$, $t_{\mathrm{on},z}$ are the computed thruster on-times;

(f) the new updated Tait-Bryan error angles and error rates combined the deterministic and corrective terms:

$$
\begin{aligned}
\phi(k+1) &= \phi_d(k+1) + \phi_1(k+1) \\
\theta(k+1) &= \theta_d(k+1) + \theta_1(k+1) \\
\psi(k+1) &= \psi_d(k+1) + \psi_1(k+1) \\
\dot{\phi}(k+1) &= \frac{(\phi(k+1) - \phi(k))}{t_i} \\
\dot{\theta}(k+1) &= \frac{(\theta(k+1) - \theta(k))}{t_i} \\
\dot{\psi}(k+1) &= \frac{(\psi(k+1) - \psi(k))}{t_i}
\end{aligned} \qquad [16.5]
$$

(g) the measurement extrapolation and innovation computation were similar to the filter used for three-gyro real-time attitude determination. However the state vector (ϕ, θ, ψ, $\dot{\phi}$, $\dot{\theta}$, $\dot{\psi}$, $\gamma_{1,x}$, $\gamma_{1,y}$, $\gamma_{1,z}$) was now updated using a (9×6) array of fixed gains (9 elements to the state vector and 6 different star mapper slits). As before, separate gain sets were included for Quality 1 and Quality 2 stars;

(h) after update, the corrective terms had to be recalculated before the next cycle:

$$
\begin{aligned}
\phi_1 &= \phi - \phi_d \\
\theta_1 &= \theta - \theta_d \\
\psi_1 &= \psi - \psi_d
\end{aligned} \qquad [16.6]
$$

Sun Acquisition Sensor Controller

There was a need to maintain the spacecraft stability and achieve a certain operational safety with a zero-gyro system. The questions how to manoeuvre the spacecraft and how to maintain the attitude of the spacecraft at any solar aspect angle still needed an answer. Looking for alternative ways of operations, the spacecraft equipment used only for launch and early orbit phase was examined and the sensing and measuring devices still operational in the spacecraft were scrutinised. Finally, using existing features, a way to maintain and correct the measured angles with respect to the Sun was achieved using the sun acquisition sensor aligned with the z-axis. The resulting modification was called the sun acquisition sensor controller (modified sun acquisition manoeuvre 4).

The sun acquisition sensor was capable of sensing the direction of the Sun along two perpendicular axes, provided that the solar aspect angle, (the angle between the Sun and the spacecraft z-axis) never exceeded approximately 70°. Each of the two aspect angles output from the sun acquisition sensor formed a sine wave whose frequency and amplitude were a function of the spacecraft spin rate and of the solar aspect angle. The faster the spacecraft was spinning, the higher the frequency of the sine waves; the further the Sun was from the spacecraft z-axis, the bigger the amplitude. If the spacecraft could have two reference sine waves representing together the nominal spin-rate and the nominal spacecraft angle with respect to the Sun, those references could be compared with the actual waves seen by the solar aspect angle. Then, if deviations induced by external factors could be corrected, the sun acquisition sensor and spin rate would be maintained under control.

The new sine models were propagated using the same orbital oscillator coefficients that controlled the precession of the z-axis around the 43° cone centred on the Sun. Also, the attitude and orbit control system supported an operational mode used for early orbit sun acquisition able to correct deviations in x and y axes by firing the relevant thrusters, with the property that it was triggered by error signals coming from the solar aspect angle. When combining both of the above, an alternative way to control the spacecraft attitude could be derived.

16.3. On-Ground Software

Ground Real-Time Attitude Determination

The major changes in the ground real-time attitude determination chain (described more fully in Chapter 14 and Chapter 15) are now summarised. One new task was developed to support the ground real-time attitude determination: ‘full sky matching’ which is described below.

In order to perform initial star pattern recognition, it was necessary to have information from both B_T and V_T channels to provide not only brightness but colour information. Without both, the range of candidate stars for each match would have been too large to achieve a successful match sufficiently quickly. Once a match was achieved however and the on-board real-time attitude determination was successfully initialised, the knowledge of which stars should be observable in the immediate future meant that the ground system should be able to continue to run in monitoring mode with only one star mapper channel, thus freeing up the space in telemetry for the necessary science data from the image dissector tube (on-board processing actually used only one star mapper channel).

Telemetry processing and filtering: The software needed to be able to work with star mapper 1 and/or star mapper 2 depending on the selected telemetry format with independent filtering of B_T and V_T channels. In addition a more responsive background noise estimation algorithm was added which additionally filtered out the occasional 'spikes' seen in the telemetry stream leading to a more accurate estimate of the star brightnesses. The opportunity was also taken at this time to install an improved graphical interface for initial scan rate determination.

Slit distinction and gyro correction: The two star mappers worked with two distinct sets of slits (Chapter 2) with the stars crossing first star mapper 2 (B_T channel) followed by star mapper 1 (V_T channel). The time separation between the vertical and inclined slits for one star mapper was between 0.3 s and 7.2 s (at the nominal scan rate) depending on the position of the star with the respect to the scan plane. In matching stars across two star mappers, separations of up to 38 s had to be allowed for. The ground software needed the ability to perform slit distinction for two non-synchronised star mapper channels (B_T and V_T). This involved a complete redesign of this element of the system to allow the determination of the times for each star to cross all four star mapper slits (vertical or inclined) as well as correct the crossing times for actuations.

Since the separation between the two vertical slit transits in each channel were well defined, the algorithm was based on matching the vertical V_T channel transits to the vertical B_T channel transits. When a matching pair of vertical transits had been identified, the inclined transit partners had to be searched among the past transits of the vertical B_T channel transit and the future partners of the V_T channel transit. A suitably long time gap had to be maintained to ensure that all the transits used in a match could only occur for one star and that there was no possibility for ambiguity.

The time to transit all four slits of the vertical slit system gave a coarse estimate of the scan rate which could be used to correct the transit times for scan-rate variation before star pattern matching was attempted.

Full sky matching: The lack of gyro information around any of the three axes meant that significant excursions of the spin axis from the nominal scanning law could be expected after every perigee passage. During three- and two-gyro operations, initial star pattern matching had assumed a known solar aspect angle ξ and initial precession angle $\overline{\nu}_0$ (see the nominal scanning law definitions in Chapter 8). This new off-line program was developed to determine ξ and $\overline{\nu}_0$ to within 1° accuracy, using statistical star pattern recognition techniques similar to that employed for the 'star pattern offset matching' task (Chapter 15).

The software required a list of successful star mapper transits from the previous 'slit distinction and gyro correction' task. Both star mapper channels had to be in telemetry. The full sky matching algorithm then pre-selected only rarer stars which were above a certain brightness threshold or colour index to limit the list of potential candidate matches. By way of preparation, a specially filtered version of the Hipparcos Input Catalogue was prepared during the suspended operations phase, with corresponding brightness and colour thresholds.

The algorithm proceeded to match the reduced list of star transits against the filtered input catalogue to produce, for each transit, a list of potential matching stars. Individual candidates for different transits were then considered in a pairwise manner. If the combination of transit times and angular separation were such that the star could have been observed at those times in one or other of the two fields of view, the cross product of the two star vectors (giving the spin axis orientation) was calculated and stored. Once all possible combinations had been considered, the resulting cross product vectors were binned in small areas (typically $2° \times 2°$) of the celestial sphere and displayed graphically as a colour coded density map. A clear peak around the true spin axis orientation was invariably apparent.

The algorithm used for the two-gyro operations to determine the initial heliotropic angle Ω_0 was incorporated into the new software. Once ξ and $\overline{\nu}_0$ had been identified in this way, the same transits could be used to determine Ω_0. With this knowledge of the 'best fit' nominal scanning law parameters to the current attitude, the real-time star pattern monitoring was able to be initialised.

Typically a good match was obtained with 10 to 15 rare transits, which took between 3 and 5 minutes to collect, assuming that the spacecraft z-axis did not drift by more than 3° in that period.

Star pattern monitoring: The original algorithm for star pattern matching was capable of working with one star mapper channel information or two by suitable changing of on-line parameters. It was however fully expected that the number of ambiguities in matching would rise when only one star mapper channel was available.

Fine attitude estimation: The on-ground Kalman filter was adapted to replace the gyro information by the same theoretical disturbance torque model as was used on-board. Additional thruster misalignments and calibration results were modelled in the attitude and body rate prediction. It was possible to obtain a coarse estimate of the spacecraft body rates by calculating the time for a star to transit all four slits of the vertical and inclined slit systems of one of the star mappers. The time to cross the vertical slits gave the scan rate around the z-axis. The angular speed across the inclined slits, less the scan rate, gave a body rate about an axis perpendicular to the plane containing the z-axis and the star mapper field of view direction. These two measurements were input to the Kalman filter.

Real-time initialisation: The final task was required to reset the new format on-board state vector. Previous versions of the task had performed automatic uplink of the state vector command as soon as a star transit had provided a recent update to the on-ground attitude estimate. This was to prevent accumulation of error due to extrapolation. A manual override was now added to allow the operator to manually command within a few seconds of receiving an on-ground update, after verifying its accuracy. The task was also capable of finding the best fitting nominal scanning law parameters before the programme star file was generated. This would minimise attitude corrections with a subsequent saving in fuel and time.

On-Ground Attitude Reconstitution

A very significant step was the inclusion of the NDAC Consortium's on-ground attitude reconstitution software at ESOC for use in calibrating external disturbance torques and thruster performance. The experience of the Royal Greenwich Observatory (RGO) in calibrating the external torques and thruster throughout the mission provided considerable insight into the problems confronting the development of the zero-gyro systems. Studies with Matra showed that it was likely that the disturbance torques would require calibration every orbit to provide accurate enough information for the on-board acceleration tables.

The RGO collaborated closely with ESOC to establish a way of installing an adapted version of their system within the ground control system using an additional high performance work-station. They also provided test data for both the on-board and on-ground systems, through the system.

Discrete Disturbance Torque Model

In addition to the off-line on-ground attitude reconstitution software described above, a new program was developed to uplink the commands necessary to refresh the on-board acceleration tables at regular intervals, using input from on-ground attitude reconstitution results. This software had to note when one of the two tables had expired and uplink the next set of data while the other table was active. These tables were obviously dependent on the nominal scanning law and could only be uplinked once the first coarse attitude estimates had been achieved on-ground.

Modified Sun Acquisition Sensor Support

The two new sine wave models for the sun acquisition sensor controller required to be calculated and refreshed once the current nominal scanning law parameters had been defined. These coefficients were calculated off-line and uplinked by manual command.

Additional changes

The nominal focus monitoring task had used gyro derived rate information. This information was replaced by the telemetered real-time attitude determination error rates.

The real-time telemetry and telecommanding chain underwent numerous changes for zero-gyro operations, including:

- additional routing of telemetry to the new workstation required for the ground real-time attitude determination and on-ground attitude reconstitution software and receipt of commands;
- development of a telemetry template and format to allow the recovery of both star mapper streams;
- development of a filing system for the new workstation, keeping a local copy of all required telemetry. Details of on-board time and frame number were also stored;
- conversion of the data reduction consortia tape production software to produce disk format output for transfer to the workstation. The data fed the on-ground attitude reconstitution software. This data reduction consortia tape software incorporated flags for the solar aspect angle and the telemetry mode;
- modification of the uplink system to decompose all commands before transmission.

16.4. Operational Experience

The spacecraft remained spinning at 0.25 rpm until 11 June 1993 when the on-ground software was completed. An initial de-spin to 0.15 rpm was then performed with nutation damping. The next day the spacecraft was de-spun to 0.05 rpm and emergency sun reacquisition was triggered to bring the spacecraft to sun-pointing mode. Earlier fears that the power margin at sun-pointing would be so low that a quick manoeuvre to the nominal 43° solar aspect angle would be required, were unfounded. It was therefore agreed to remain sun-pointing and to attempt to converge real-time attitude determination. The ground real-time attitude determination software worked as expected and was able to perform star mapper filtering with the two semi-functioning star mappers. The attitude was subsequently determined and initialisation of real-time attitude determination on-board attempted. It was then found that the on-board software was incapable of responding accurately to the frequent thruster firings caused by the initialisation command, resulting in an unpredictable divergence after each command.

Within a few hours the ground software was modified to limit changes in the uplinked on-board state vector in such a way as to reduce thruster on-times at the normal mode boundary. In the following days further modifications were made to the on-board and ground real-time attitude determination software.

On-board real-time attitude determination: Convergence of real-time attitude determination under simulations showed sensitivity to star mapper transit frequency. Some attempts were made to incorporate Tycho catalogue stars into the program star file at sparse areas. This was not tested with the spacecraft however. Having verified the performance of the ground real-time attitude determination software in determining body rates, some changes to the on-board Kalman gain set were made to increase the sensitivity to innovations of the transverse body rates on-board. This helped by allowing the real-time attitude determination to stay converged after initialisation from ground, making it possible to collect real-time attitude determination converged data every pass for further analysis. From this it was possible to correlate later divergences with firings, leading to the conclusion that a new thruster calibration was required. Insufficient data for use with calibration software on-ground, was obtained before the end of the mission.

Ground real-time attitude determination: After real-time attitude determination operations resumed, some significant changes were made when it was found that a less strict approach to using three out of four sets of slit crossings still maintained star pattern monitoring and allowed more measurements for on-ground attitude convergence. A further modification was made which limited the commanded change to the on-board state vector to lie just outside the normal mode control bounds. In this way, convergence of the on-board body rate estimates could be maintained. An additional command was added to allow only the on-board body rates to be initialised, while maintaining the old attitude.

Short periods of converged real-time attitude determination were achieved, ranging from 15 minutes to 1 hour, over the next days in sun-pointing mode. The frequency with which these periods of convergence could be achieved was steadily increasing, with the further tuning and familiarisation with the new operations mode.

Throughout this period, investigations by ESOC, Matra Marconi Space and the RGO were on-going concerning the fragility of the on-board real-time attitude determination. It was concluded that the *a priori* thruster calibration made by RGO did not adequately reflect the true thruster performance after such a long sun-pointing with a very different thermal environment. ESOC and RGO worked toward a revised thruster calibration by using the NDAC Consortium's on-ground attitude reconstitution software on the short intervals of real-time attitude determination-converged data obtained.

On 15 June 1993, the spacecraft was manoeuvred to the nominal 43° solar aspect angle having spun down the spacecraft to zero, before invoking the sun acquisition manoeuvre controller with fixed 43° sun acquisition sensor bias. The manoeuvre had an overshoot of 4 – 5° due to initial inaccuracies in the sun acquisition sensor calibration and the spacecraft was returned to sun-pointing. A second attempt some hours later was successful. The modified sun acquisition manoeuvre controller with oscillating sun acquisition sensor biases was successfully used and maintained the spacecraft safely through the next perigee.

Ground real-time attitude determination attempts were unsuccessful due to the high frequency of firings and the very high transverse rates (20 – 30 arcsec s^{-1}) experienced by the spacecraft at that time. Moreover, the fuel consumption under this controller proved to be unexpectedly heavy (30 grams per day). The reason for both phenomena was that the simple sinusoidal model of the sun acquisition sensor bias on-board did not adequately reflect the real sun acquisition sensor output for the spacecraft spinning at 43°. This was a recognised problem beforehand, however neither the sun acquisition sensor calibration nor the simulations had shown that the effect would be so severe.

On 16 June 1993, the spacecraft was manoeuvred back to sun-pointing mode, to improve overall real-time attitude determination performance there. A revised strategy was devised for operating at 43°, using various combinations of control thresholds timed with the phase of the sun acquisition sensor which would have reduced both fuel consumption and transverse rates. This was not attempted however until the overall real-time attitude determination performance could be improved.