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ORBIT AND ATTITUDE DETERMINATION RESULTS DURING LAUNCH SUPPORT OPERATIONS FOR SBS-5

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ABSTRACT

Presented are orbit and attitude determination results from the launch of Satellite Business Systems (SBS)-5 satellite on September 8, 1988 by Arianespace. SBS-5 is a Hughes Aircraft Corporation HS-376 spin-stabilized spacecraft. The launch vehicle injected the spacecraft into a low inclination transfer orbit. Apogee motor firing (AMF) attitude was achieved with trim maneuvers. An apogee kick motor placed the spacecraft into drift orbit. Postburn, reorientation and spindown maneuvers were performed during the next 25 hours. The spacecraft was on-station 19 days later.

The orbit and attitude were determined by both an extended Kalman filter and a weighted least squares batch processor. Although the orbit inclination was low and the launch was near equinox, post-AMF analysis indicated an attitude declination error of 0.034 degree, resulting in a saving of 8.5 pounds of fuel. The AMF velocity error was 0.4 percent below nominal.

The post-AMF drift rate was determined with the filter only 2.5 hours after motor firing. The filter was used to monitor and retarget the reorientation to orbit normal in real-time.

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1. INTRODUCTION

SBS-5 is an HS-376 spin-stabilized communications spacecraft built by Hughes Aircraft Corporation (Reference 1). It is designed to provide voice, video, and data traffic to the United States on 14 channels in the Ku-band frequency, i.e. in the 14/12 GHz range.

Figure 1 shows an exploded view of the spacecraft components. The spacecraft remains in the stowed position until drift orbit, when the communications platform is despun and the solar panel is extended. The main sections of the spacecraft are the spinning rotor and the despun Earth-oriented platform.

The reaction control subsystem (RCS), which uses hydrazine propellant, is located on the spinning rotor. It consists of two independent systems joined by an interconnect manifold. Each system contains two conispherical tanks, one radial, and one axial thruster. A Thiokol Star 30 apogee kick motor (AKM) provides the impulse to inject the spacecraft into drift orbit.

The attitude control subsystem (ACS) provides velocity control, spin-axis attitude control and antenna pointing control. Data for attitude determination are provided by spinning Sun and Earth sensors. A thruster actuated active nutation control (ANC) subsystem is used after apogee motor firing to dampen nutation. After the communications reflector is deployed, the despin active nutation damping electronics (DANDE) controls nutation with despin motor torques. Accelerometers in the rotor sense nutation for both ANC and DANDE nutation damping. Figure 2 shows the ACS functional block diagram (Reference 1).

Telemetry, command/track, and ranging subsystems provide command capability and spacecraft information. The slant range from the ground station to the spacecraft is determined by a multiple-tone ranging system.

The Flight Dynamics group and Mission Control Facility (MCF) are located at the Clarksburg, Maryland ground station. This facility is used for tracking, telemetry and commanding (TT&C) support and contains the real-time computers for data collection and storage. In addition, the following ground stations provided support: Castle Rock, Colorado; Allan Park, Canada; Perth, Australia; Sydney, Australia; and Betzdorf, Luxembourg.

The software used to support the SBS-5 mission was written by the Flight Dynamics Department at Telesat Canada (Reference 2). The software system was used to support several Telesat missions and was modified for the SBS mission. Two Hewlett-Packard (HP)-1000 minicomputers were used as primary and backup systems for the mission. Telesat personnel also provided support during the SBS-5 mission.

SBS-5 was assigned a longitude of 123 degrees West by the Federal Communications Commission (FCC). Prelaunch mission analysis was conducted to optimize the transfer orbit and drift orbit trajectories based on a 10-year mission lifetime requirement.

SBS-5 and the GSTAR-3 spacecraft of GTE Spacenet Corporation were copassengers on the Ariane 3 launch vehicle. The launch site was Kourou, French Guiana. A combined launch window is shown in Figure 3. The SBS-5 constraints are based on Sun angle and eclipse duration restrictions.

The spacecraft sensors provide three basic measurements: Sun angle, Earth chord, and separation data, i.e. the angle between the spin axis-Sun plane and the spin axis-Earth plane. Using only Sun angle and Earth chord measurements, attitude determination errors place stringent constraints on the launch window for low inclination orbits during periods near the equinoxes. However, the spin-axis attitude is sensitive to separation angle measurements except when the Earth, the spacecraft, and the Sun are colinear. Since colinearity occurs at only one instance in the transfer orbit, and sensor data are collected for approximately two hours during each apogee pass, separation angle measurements restore attitude determination accuracy during periods near equinox (Referrence 3).

Liftoff occurred at the opening of the launch window (23:00 GMT), resulting in a nominal separation time of 23:20:52 GMT. The separation Sun angle of 72.1 degrees and spin rate of 6.94 rpm were near the nominal predicted values of 70.0 degrees and 7.0 rpm respectively. The postlaunch orbit vector delivered by Ariane showed the semimajor axis 2.6-sigma below nominal. After processing tracking data through the first and second revolutions (revs), the semimajor axis was determined to be 4.8-sigma high.

A spinup maneuver was performed during the first rev to increase the spin rate to 50 rpm. Thruster functionals and ANC test firings were performed. The first attitude trim maneuver was postponed until the fourth rev due to the uncertainty in the orbit and attitude state. An attitude touchup was necessary before the apogee motor firing (AMF) on rev 7.

On September 11 at 20:17:39 GMT, the apogee motor was fired at a longitude of 150.5 degrees West. The orbit solution from the Kalman filter 2.5 hours after AMF showed that the orbital plane was achieved, but the drift rate was 3 degrees per day East instead of the nominal 1 degree per day East. This indicated a 0.4-percent underperformance of the apogee motor with negligible attitude pointing errors. The drift orbit maneuver sequence was redesigned.

At the perigee following AMF, a postburn maneuver and a reorientation (reor) maneuver to near orbit normal attitude were performed. The attitude target was planned to account for the precession expected during the spindown maneuver. The reor was monitored and retargeted in real-time with the Kalman filter. At the next apogee, the spindown maneuver was performed to reduce the spin rate to 37.4 rpm in preparation for despinning the communications platform. The spin rate increased to 55 rpm as a result of deployment activities which began on on September 13.

On September 18 a series of four maneuvers were executed to reduce drift rate. Three additional orbit maneuvers were performed on September 26 and 27 to stop the drift rate at a longitude of 123 degrees West. Stationkeeping support began on September 27. The first North-South maneuver was executed on October 12.

2. MODELING

Predicting the orbital and attitude motion of a spacecraft during the transfer orbit is very complicated. This is due to the many perturbing forces on the spacecraft such as the Earth's geopotential field, atmospheric drag, and lunar and solar gravity. These forces must be accounted for to accurately predict the motion of the spacecraft. An Earth sensor model is required to predict the output as seen from the spacecraft Earth sensors.

2.1 ORBIT DETERMINATION

An extended Kalman filter and weighted least squares batch processor were both used for orbit determination during launch support (Reference 2). The Mission Kalman Filter (MKAL) Program provided real-time orbit estimation, tracking data monitoring, and automatic deweighting of less accurate or "bad" data samples. The effects of drag at perigee on the orbit are predicted and the state covariances are corrupted appropriately. The filter equations are augmented with considered parameters, i.e. station location and range bias uncertainties to insure stability (Reference 4). The filter estimates the spacecraft position and velocity vectors, and azimuth and elevation biases. When a set of observations are averaged, the orbit state vector and covariance matrix are advanced to the average time of the observations. The data are filtered using the extended Kalman filter algorithm. The method produces a state vector which minimizes the weighted mean square residuals between the model and the measurements. The filter adjusts the covariances to account for modeling, propagation, and maneuver performance errors.

The weighted least squares algorithm with a priori statistics was used to minimize the sum of the square of the weighted residuals between actual and computed observations, while simultaneously constraining the state to satisfy an a priori state within a specified uncertainty (Reference 2). The data are averaged to the mid-interval of the observations. The mean and the root mean square are computed for each residual type. Azimuth and elevation biases for each station were adjusted with this information. Range biases for Allan Park and Perth were set to zero for the mission. Range biases for Castle Rock and Clarksburg were set to values estimated by orbit determination of on-station spacecraft.

The spacecraft position vector was propagated by numerical integration of the Encke formulation of the equations of motion. A fourth order Runge-Kutta-Gill integrator was used with variable stepsize. Stepsizes of 54 seconds at perigee, and 360 seconds at apogee, were used during transfer orbit. A stepsize of approximately 35 minutes was used during drift orbit.

The following forces were modeled: geopotential, lunar and solar gravity, and atmospheric drag. Solar radiation pressure was modeled during drift orbit. The Goddard Earth Model (GEM)-10 was used for nonspherical gravitational perturbations. The geopotential order 6 was used during the transfer orbit. Accelerations due to lunar and solar forces were computed via Encke's method. Drag acceleration was modeled analytically as outlined in Reference 5. The Jacchia-Roberts atmospheric density model was used. Lunar and solar positions were obtained from a Jet Propulsion Laboratory ephemerides tape.

The tracking data rates were as follows: The Perth and Allan Park ground stations sent 25 azimuth, elevation, and range measurements within 25 seconds; Castle Rock sent six azimuth and elevation measurements within 30 seconds, followed by 25 range tone measurements in 25 seconds; Clarksburg sent 5 range tone measurements in 5 seconds (only during the drift orbit).

2.1.1 INITIAL ASSUMPTIONS

The Ariane 3 launch vehicle injects the spacecraft into a transfer orbit with a perigee height of 200 kilometers, an apogee height of 36,206 kilometers, and an inclination of 7 degrees. After transfer orbit injection, the attitude and roll control system (SCAR) of the Ariane vehicle begins orientation maneuvers required before separation of the GSTAR-3 and SBS-5 satellites. Separation of SBS-5 occurs about four minutes after injection.

The prelaunch nominal separation state (at injection time) was as follows:

Epoch (GMT)	-	88:252:23:16:47:241	1-sigma
	-	September 8, 1988	
Semimajor Axis (km)	-	24554.0830	20.231
Eccentricity	-	0.73210339	0.226E-03
Inclination (deg)	-	6.998977	0.173E-01
Mean Anomaly (deg)	-	0.536041	0.157E-02
Argument of Perigee (deg)	-	177.977117	0.141
Node (deg)	-	146.985053	0.141
Mass (kg)	-	1238.9741	
Spin (rpm)	-	7.0	
RMOI (kg * m**2)	-	500.25623	

This state was used for prelaunch mission analysis and for generating nominal station prediction information. It corresponds to a seventh rev apogee bias of approximately 171.6 kilometers. A correlation matrix in spherical coordinates at spacecraft injection/separation was delivered by Ariane before launch. It contained expected launch vehicle dispersions.

2.1.2 ESTIMATION OF ORBITAL DRAG

Prelaunch analysis was conducted to develop a procedure for estimating the effect of atmospheric drag on the orbit (Reference 6). Because the spacecraft attitude was altered only slightly during transfer orbit, the angle between the spin axis and the velocity vector of the spacecraft relative to the atmosphere was the same each perigee. It was not possible to distinquish between the axial and normal drag force components, so the drag coefficents were equated.

The perigee drag coefficients were estimated with two successive revs of tracking data to independently compute the apogee height of the second of the two revs. Due to perigee drag, the apogee height computed with the second of the two revs of data should be lower than the apogee height computed with the first of the two revs. The difference in the computed apogee heights was expected to vary from a minimum of 540 meters, corresponding to drag coefficients of 1, to a maximum of 2.7 kilometers, corresponding to drag coefficients of 5.

2.2 ATTITUDE DETERMINATION

The MKAL Program was used for real-time attitude estimation and reorientation maneuver retargeting. The Mission Attitude Determination (MAD) Program was used for weighted least squares batch processing of data (Reference 2).

The attitude data were provided by the spacecraft attitude data processor (ADP), which makes time interval measurements (t-times), between occurrences of real-time reference pulses. Ten measurements made up one set of t-times, which were received approximately every 15 seconds. These pulse code modulation (PCM) data were then converted to the types needed for attitude determination. The five data types were: psi data (the spin angle separating the midpoints of the psi and psi-2 Sun pulses); north and south Earth sensor half-chord widths; and north and south Earth separation data (the angles between the psi pulse and the center of the north or south Earth sensor pulse).

Five sets of observations are averaged to the midpoint of the interval. The attitude state vector included the phi and theta angles of the spin-axis attitude vector, psi bias, Earth sensor cant angle biases, and Sun/Earth sensor separation biases. Earth sensor chord width biases were determined with the MAD Program and applied to the observation model of the filter. During large reor maneuvers, the maneuver precession rate and force centroid error are also included in the filter state vector. (The force centroid error is the difference between the desired and actual thrust direction.)

The Phi-Theta coordinate system (Reference 2) is shown in Figure 4. Theta is the angle between the planes formed by the Sun vector/Earth spin axis and the Sun vector/spacecraft spin-axis vector. Theta is positive if measured clockwise looking along the Sun vector. Phi is the angle measured from the spacecraft spin axis to the Sun vector.

Earth sensor scanning and delay models were included in the software (Reference 2). Since the Earth sensors have a 1.5-degree field of view that is diamond shaped, a point source cannot be used to accurately predict the Earth chord width. The scanning model reproduces the discrete digital waveform of the Earth sensor and uses a fixed threshold level of 31 percent to estimate the Earth chord width. Due to thermal capacitance and AC coupling in the Earth sensor amplifiers, the actual sensor output will be distorted (Reference 2). The delay model was developed to account for this effect.

2.2.1 INITIAL ASSUMPTIONS

SBS-5 was injected with an orientation near AMF attitude. This was based on an average AMF attitude over the Ariane two-month launch period. The Ariane 3-sigma error on this injection attitude was 2.8 degrees, with a contractual agreement that it be within six degrees. The prelaunch nominal attitude state was as follows:

Epoch (GMT)	-	88:252:23:16:47:241	
			1-sigma
Spin-axis declination (deg)	-	236.428	2.00
Spin-axis right ascension (deg)	-	-8.3495	2.00

The nominal AMF target attitude had a spin-axis right ascension of 238.148 degrees and declination of -7.777 degrees. An objective of this mission was to target the post-AMF plane so that the inclination was less than 0.05 degree on October 15, 1988, without North-South maneuvers.

The pre-AMF inclination was 7.0 degrees and the right ascension of the ascending node was 146.98 degrees. The target post-AMF inclination was 0.16 degree and the node was 287.57 degrees. The target plane was biased to allow for the plane change effects of the reorientation maneuver to orbit normal and the spindown maneuver.

2.2.2 ESTIMATION OF ATTITUDE PRECESSION DUE TO PERIGEE DRAG

The torque due to atmospheric drag is in the direction of $\hat{A} \times \vec{V}$, where $\vec{V}$ is the velocity of the spacecraft with respect to the atmosphere, and $\hat{A}$ is the spin-axis vector. At perigee, $\vec{V}$ is normal to the spacecraft position vector $\vec{R}$, and $\hat{A}$ is nearly normal to $\vec{R}$. Therefore, $\hat{A} \times \vec{V}$ is nearly in the direction of $\vec{R}$. Since the inclination of the spacecraft orbit is low, $\vec{R}$ is directed primarily in right ascension. The change in right ascension due to perigee precession was estimated from the measured change in the Sun angle, phi, during perigee passage. The Sun angle is estimated from the measured rotation angle, psi, between the midpoints of the two Sun pulses from the spacecraft Sun sensor.

3. RESULTS

A summary of both weighted least squares and real-time orbit determination results are shown in Table 1. A summary of both weighted least squares and real-time attitude determination results are shown in Table 2.

3.1 TRANSFER ORBIT

3.1.1 SEPARATION STATE VECTOR DETERMINATION

Liftoff of the Ariane 3 occurred at the opening of the September 8th launch window (23:00 GMT), resulting in a nominal SBS-5 separation time of 23:20:52 GMT.

Ariane delivered a transfer orbit solution about 23:46 GMT. It was obvious soon after launch that the SBS-5 transfer orbit was not nominal. The semimajor axis of a weighted least squares (WLS) solution was 96 kilometers above the Ariane nominal, whereas the Ariane 3-sigma error was only 61 kilometers. The nominal seventh rev apogee bias was 171.6 kilometers, whereas the WLS apogee bias was 367.1 kilometers (Table 1).

Ariane delivered an attitude solution shortly after launch. This is shown in Table 2 as the GSFC/ARIANE solution. It was 2.2 degrees from the nominal and within the Ariane 3-sigma error. However, the spacecraft attitude could only be verified in the phi direction. The ADP on-board the spacecraft cannot send valid data at spin rates below 29 rpm. Frequency modulated (FM) real-time attitude sensor data were used to calculate a Sun sensor psi to psi period of 8642 milliseconds, corresponding to a spin rate of 6.9 rpm. The phi angle of 72.1 degrees was derived from these psi data, and was identical to the Ariane delivery.

The MKAL Program diverged due to inaccurate starting conditions. The filter was restarted with the WLS solution. At the start of rev 2, the filter diverged again, and it was restarted with a WLS solution based on tracking data just from rev 2. The attitude covariance was reinitialized. This shows the importance of an accurate state vector and covariance matrix for starting the filter.

3.1.2 ORBITAL DRAG ESTIMATION

Since the first attitude trim maneuver was postponed until the fourth rev of the transfer orbit, an estimate of the orbital drag could be made for perigees

2 and 3. The drop in apogee due to perigee 2 was estimated to be 0.592 kilometers, which correponded to a drag coefficient of one. The drop in apogee due to perigee 3 was estimated to be 0.013 kilometers, which corresponded to a drag coefficient of zero. The decision was made to leave the drag coefficients equal to zero throughout the transfer orbit.

3.1.3 ATTITUDE SENSOR AND DATA BIAS DETERMINATION

A procedure was developed during launch simulations to calculate sensor and data biases using the MAD Program. The psi bias was first computed with cant angle biases fixed at zero. Then the cant angle biases were computed with the psi bias fixed at its value computed in the first iteration.

A psi bias of -0.242 degree was estimated at the end of rev 1. A psi bias of 0.088 degree was estimated at the end of rev 2 and -0.067 degree during rev 3. It was apparent that the psi bias was not converging. The decision was made to fix the psi bias at zero and calculate the other biases with WLS processing for the remainder of the mission (Table 2). Chord width biases estimated with WLS processing were input to the filter.

3.1.4 ATTITUDE TOUCHUPS FOR APOGEE MOTOR FIRING

The first attitude touchup was performed on rev 4. The reor target is shown in Table 2. The reor target was biased to account for precession expected due to perigees 5, 6, and 7. The perigee precession was consistent from rev to rev. The delta psi angle was about -0.013 degree, which corresponded to a delta phi angle of 0.016 degree.

Figure 5 shows the results of the first attitude touchup. The filter solutions are designated by KF, and the WLS solutions are designated by AD. Each solution was propagated to rev 4 to account for perigee precession. The difference, 0.82 degree, between the WLS solution for revs 2 and 3 indicates a modeling error. The WLS solutions for revs 4, 5, and 6 in Table 2, show similar but smaller differences. The WLS solution for revs 4, 5, and 6 taken together (not shown in Table 2) is essentially identical to the filter solution at the end of rev 6.

The first attitude touchup was executed with 115 pulses of the axial 1 thruster and a start angle of 172.7 degrees. The expected precession was 1.68 degrees, and the expected precession phase was 334.4 degrees. Table 3 shows the precession calculated from the premaneuver and postmaneuver attitude solutions in Figure 5. The table also shows the percent difference between the measured and nominal precession, the difference between the measured and nominal precession phase, relative to the Sun's azimuth in the spacecraft spin plane, and the corresponding thrust centroid error. In each case, the postmaneuver attitude is the KF/END-REV6 solution. Of the three premaneuver solutions, the KF/END-REV3 solution was probably closest to the true attitude.

Figure 6 shows the results of the second attitude touchup. Only sensor data were provided during rev 6 which was the Luxembourg pass. The decision whether or not to do a touchup on rev 7 was based primarily on the filter solution at the end of rev 6.

As mentioned above, the discrepancy between the weighted least squares attitude state solutions for revs 2 and 3 indicates a modeling error. The attitude solutions differ by 0.82 degree, and sensor biases differ significantly (Table 2). Most of the discrepancy between the attitude solutions can be removed by fixing the biases. Table 4 contains the weighted least squares solutions using fixed biases. For comparison, the corresponding solutions from Table 2 are also shown. To account for perigee precession, all solutions were propagated to rev 4. Table 4 shows that fixing the biases removes most of the difference between weighted least squares solutions that use data from differerent revs.

Figure 7 shows the Earth chord residuals for revs 2 through 6. These residuals were computed with WLS using the fixed biases from solution KF/END-REV6 (Table 2). The north Earth chord data residuals showed a different signature for the even revs than for the odd revs. The south Earth chord residuals showed and even/odd rev pattern to a lesser degree. Table 4 also shows that the weighted least squares solutions for revs 4 and 6 are much closer to each other than the solution for either rev is to the rev 5 solution.

Allan Park executed the AMF maneuver as scheduled on September 11 at 20:17:39 GMT. The apogee motor firing was successful. The burn duration was about 54.9 seconds. The predicted duration was 54.1 seconds.

The AMF delta-velocity vector was determined from pre- and post-AMF orbit solutions. The delta-velocity vector corresponded to an AMF attitude error of 0.18 degree in right ascension and 0.03 degree in declination. The magnitude of the error was well within the 3-sigma value of 0.5 degree used for dispersion fuel allocation. The AKM performance error was about 0.4 percent below nominal.

The small attitude declination error resulted in a saving of approximately 8.5 pounds of fuel, based on a prelaunch study of injection and AMF dispersion errors (Reference 7).

3.2 <u>DRIFT ORBIT</u>

The filter was used to monitor and confirm the results of the apogee motor firing in real-time. Three parameters were closely monitored in an effort to confirm the AMF delta-velocity and attitude, i.e. drift rate, inclination and node. A scan was made on these three parameters by varying the apogee motor delta-velocity and attitude right ascension and declination, i.e. the initial burn vector. By comparing the post-AMF results from the filter with these scans, a preliminary estimate of the initial burn vector was made.

3.2.1 REAL-TIME POST-AMF MONITORING

Allan Park sent azimuth and elevation data following AMF. About one hour later, Allan Park also sent range data. About 1.5 hours after AMF, Castle Rock sent azimuth, elevation, and range data. Coverage continued for 3.7 hours after AMF.

The post-AMF weighted least squares solution showed an inclination of 0.15 degree, in agreement with solutions received from Millstone and NORAD. The MKAL Program showed an inclination of 0.08 degree. A later Millstone solution confirmed this inclination.

Figure 8 shows the drift rate, and inclination, along with the 1-sigma errors as determined by the MKAL Program. The filter had an accurate determination of the drift rate and inclination within 2.5 hours of AMF. The node was not accurately determined until more data were received after the six-hour ground station coverage gap. Due to an apparent 0.4-percent underperformance of the AKM, the observed drift rate, inclination, and node were different than expected.

3.2.2 REAL-TIME MONITORING OF REORIENTATION TO ORBIT NORMAL ATTITUDE

A postburn maneuver and reorientation maneuver to orbit normal attitude were performed near perigee to remove most of the drift orbit apogee bias. The reor was planned in two parts to insure Earth sensor coverage during the burn by at least one sensor. The target attitude was five degrees off orbit normal attitude to account for the precession expected during the spindown maneuver.

The first leg of the reor was performed about 15 minutes after the postburn. The second leg was performed 9 minutes later. The maneuver was terminated early due to the loss of execute pulses caused by incorrect polarization of the transmitting antenna at Allan Park. Commanding was switched to Castle Rock. Leg 3 was performed about 12 minutes later. This part of the maneuver was terminated early due to commanding constraints near Earth shadow. The attitude was about 1.5 degrees from the target. A spindown maneuver was performed about 12 hours later.

The filter was used to monitor and retarget the reor in real-time. The filter calculated a new jet start angle and maneuver duration for each leg of the reor. Because the precession error was slight, the first part of the reor was allowed to continue to its nominal end time. The MKAL Program computed a new jet start angle and duration for the second leg of the maneuver. Figure 9 shows the predicted and observed attitude motion during the reor as it was monitored in real-time with data from the MKAL Program.

3.2.3 TRANSITION FROM MISSION OPERATIONS TO STATIONKEEPING OPERATIONS

The stationkeeping Kalman filter, KALMN, estimates both the orbit vector and the spin-axis vector. KALMN models the force and torque exerted on the spacecraft due to solar radiation. KALMN uses a quadratic fit to the GEM-10 model about a specified central longitude. The force model includes the gravitational forces due to the Sun and Moon. Maneuvers are modeled as instantaneous changes in velocity and spin-axis orientation. The spacecraft state vector includes the spin-axis orientation, a correction to the solar radiation torque and force model, a psi bias, and cant angle biases.

KALMN began processing range and attitude data from 88:258 (September 14). Orbit and attitude vectors were taken from the MKAL Program. The covariances were corrupted to accommodate differences in modeling. The psi and cant angle biases were set to zero and the chord width biases were set to 0.25 degree. The corrections to the solar radiation force and torque models were set to zero. Tracking and attitude data were processed by both KALMN and MKAL until 88:262 (September 18), when MKAL processing was suspended. On 88:270 (September 26), MKAL was restarted using the KALMN state to monitor three drift and two attitude maneuvers on 88:270 and 88:271. MKAL processing was terminated on 88:272 (September 27), and normal stationkeeping operations began.

4. CONCLUSION

The source of the attitude modeling error is unknown and existed in both the Kalman filter and weighted least squares batch processor. The filter continuously estimated the attitude state vector using data from revs 2 through 6; whereas, the weighted least squares processor estimated a state vector one rev at a time, and was more sensitive to the even/odd rev modeling error.

The low orbital inclination and launch time near equinox produced poor attitude determination geometry. Attitude separation data were used to restore attitude determination accuracy. The attitude precession due to perigee drag was determined from rev to rev and this information was used to benefit the AMF attitude targeting. A Kalman filter and weighted least squares processor were used to determine the attitude and orbit. This provided a way to verify the accuracy of state vector solutions. As a result, the AMF attitude error in declination was only 0.034 degree.

Since the Kalman filter continuously updated the orbit and attitude state vector, it was possible to: (1) determine the drift rate and inclination 2.5 hours after AMF, which verified the results of the motor firing, and facilitated prompt planning of the drift orbit maneuver sequence; (2) monitor and retarget the reorientation to orbit normal attitude in real-time, which minimized pointing errors during the large maneuver.

The SBS-5 predicted end of life was extended nearly 4 months as a result of the technical expertise and dedication of the launch support team.

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Table 1. SBS-5 Orbit Determination Summary

SOLN	7TH APOGEE TIME(88:255) HH:MM:SS	BIAS (KM)	INC (DEG)	ARGUMENT OF PERIGEE (DEG)	8TH PERIGEE HEIGHT (KM)
NOMINAL	20:00:37	171.656	7.003	180.333	205.411
GSFC SEP SMA	19:47:26	75.099	7.007	179.912	198.305
R1OD3	20:24:38	366.707	7.024	177.013	200.275
R1OD4	20:24:41	367.094	7.014	176.992	200.365
KF-REV1	20:24:51	370.728	6.842	177.265	197.733
R1OD5	20:25:18	360.324	6.930	177.651	212.400
R12OD1	20:24:48	362.179	6.965	180.448	205.953
R12OD2	20:24:56	362.633	6.977	180.455	206.808
R12OD3	20:24:55	361.074	6.992	180.240	208.301
R123OD1	20:24:57	360.619	7.002	180.177	209.040
R123OD2	20:24:56	360.452	7.000	180.152	209.074
R123OD3	20:24:55	360.353	6.999	180.137	209.080
R123OD4	20:24:54	360.228	6.995	180.118	209.038
R123OD1B	20:24:56	360.195	6.998	180.119	209.592
		TOUCHUP			
R12345 OD1	20:24:53	359.919	6.99	180.148	209.358
R45OD1	20:24:48	356.398	7.012	180.091	211.260
R123 PROPUP	20:24:56	360.220	6.995	180.115	209.596
R12345OD2	20:24:50	359.917	6.985	180.123	208.705
R45OD1B	20:24:47	354.760	7.017	180.090	212.730
		TOUCHUP			
PRE-AMF	20:24:47	357.114	7.019	180.096	210.370
R1-7OD1	20:24:52	359.638	6.996	180.094	209.486

NOTES - The following is an explanation for the solution titles.

Example - RnODx

R - abbreviation for revolution
n - revolution numbers of the tracking data used in the solution
OD - weighted least squares orbit determination
x - solution number
KF - orbit determination from the filter (MKAL)

Table 2. SBS-5 Attitude Determination Summary

SOLN	SARA (DEG)	SADEC (DEG)	CANT ANGLE BIASES (DEG)		CW BIASES (DEG)		SUN-EARTH SEPARATION BIASES (DEG)		PSI BIAS (DEG)
			N	S	N	S	N	S	
NOMINAL	236.429	-8.349							
GSFC/ARIANE	238.632	-8.200							
R1AD1	238.806	-6.442	0.0000	-.0020	0.000	0.000	0.000	1.766	-.242
KF	238.782	-8.751	-.1332	.0210	0.000	0.000	1.605	1.462	.000
R2AD2	238.860	-9.120	.0068	-.0035	-.150	-.164	1.519	1.544	.088
KF/END-REV2	238.880	-8.992	.0347	.0164	-.150	-.164	1.558	1.548	.088
R3AD1	238.800	-8.296	-.0450	.0210	-.074	-.051	1.792	1.663	-.067
KF/END-REV3	238.958	-8.606	-.1724	-.1286	-.074	-.051	1.687	1.595	.093
TOUCHUP									
REOR-TARGET	237.483	-7.419							
R4AD1C	237.400	-7.736	-.0220	.0160	-.096	-.085	1.587	1.560	.000
KF/END-REV4	237.436	-7.603	-.0448	-.0059	-.096	-.085	1.643	1.594	.014
R5AD1_MAD	237.470	-7.343	-.1380	-.0780	-.104	-.082	1.827	1.708	.000
R45AD1_MAD	237.427	-7.606	-.0763	-.0345	-.100	-.084	1.671	1.609	.000
KF/END-REV5	237.462	-7.546	-.1110	-.0680	-.104	-.082	1.749	1.656	.018
R6AD1B_MAD	237.405	-7.922	-.0020	.0596	-.051	-.045	1.468	1.457	.000
KF/END-REV6	237.478	-7.608	-.0800	.0040	-.051	-.045	1.633	1.555	.025
TOUCHUP									
PREAMF-KF	237.439	-7.419	-.0084	.0421	-.051	-.045	1.694	1.579	.008
AMF-TARGET	237.47	-7.400							
AMF									
DV-VECTOR	237.292	-7.434							
POSTAMF-KFS1	238.074	-7.895	-.0330	-.0051	-.051	-.045	1.709	1.657	.019
POSTAMF-KFS3	238.070	-7.712	-.0226	-.0019	-.051	-.045	1.741	1.421	-.0096
AMF+AD1	238.149	-7.164	-.1337	.0120	-.051	-.045	1.968	1.826	-.0009

NOTES - The following is an explanation for the solution titles.

Example - RnADx

R - abbreviation for revolution
n - revolution numbers of the sensor data used in the solution
AD - weighted least squares attitude determination (MAD)
x - solution number
KF - attitude determination from the filter (MKAL)

Table 3. Precession and Phase Errors Relative to Nominal

ATTITUDE SOLUTION PREMANEUVER	POSTMANEUVER	DELTA-P (DEG)	DELTA-P ERROR (%)	PHASE ERROR (DEG)	CENTROID ERROR (MS)
R2AD2	END-RV6-KF	2.08	24	- 12.7	- 42
R3AD1	END-RV6-KF	1.54	- 8	6.8	23
END-RV3-KF	END-RV6-KF	1.82	8	0.3	1
R3AD1	R5AD1	1.65	-2	-1.6	-5

<u>NOTES</u>

DELTA-P is the angular separation between the intial and final solutions.
DELTA-P ERROR is the difference between DELTA-P and the nominal precession.
PHASE ERROR is the difference between the measured and nominal precession phase relative to the Sun's azimuth in the spacecraft spin plane.
CENTROID ERROR is PHASE ERROR divided by the spin rate.
R3AD1 and R5AD1 were the solutions used to calibrate the first reor during the mission.

Table 4. Comparison of WLS Attitude Solutions Generated with Fixed and Free Biases. (All angles are in degrees.)

	FREE BIASES			FIXED BIASES		
REV	SARA	SADEC	ANG. SEP.	SARA	SADEC	ANG. SEP.
2	238.89	-9.12		238.87	-8.76	
			0.82			0.06
3	238.82	-8.30		238.88	-8.70	
			TOUCHUP			
4	237.40	-7.74		237.44	-7.66	
			0.40			0.13
5	237.45	-7.34	0.18	237.42	-7.79	0.04
			0.59			0.17
6	237.37	-7.92		237.44	-7.62	

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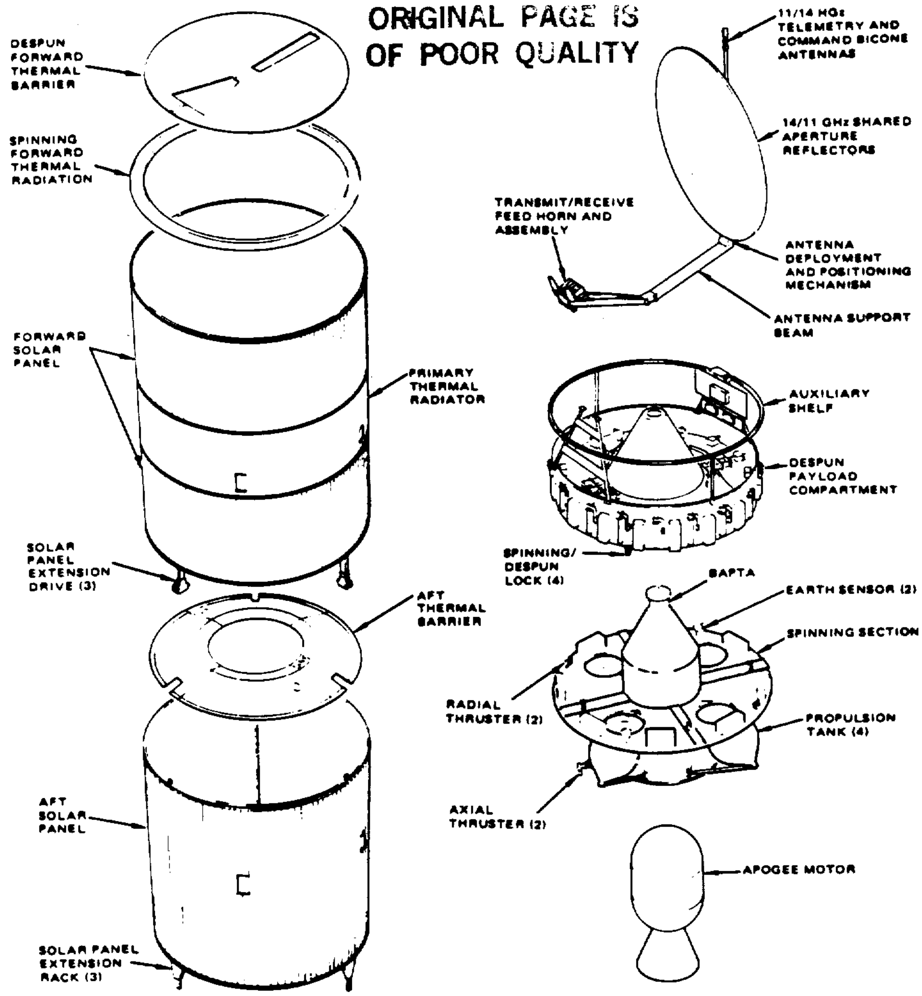

Figure 1. SBS-5 Spacecraft Components

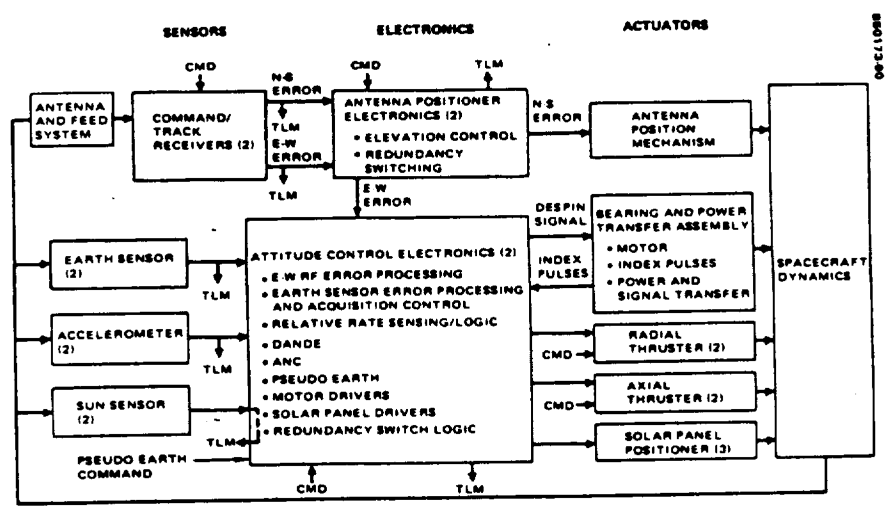

Figure 2. ACS Functional Block Diagram

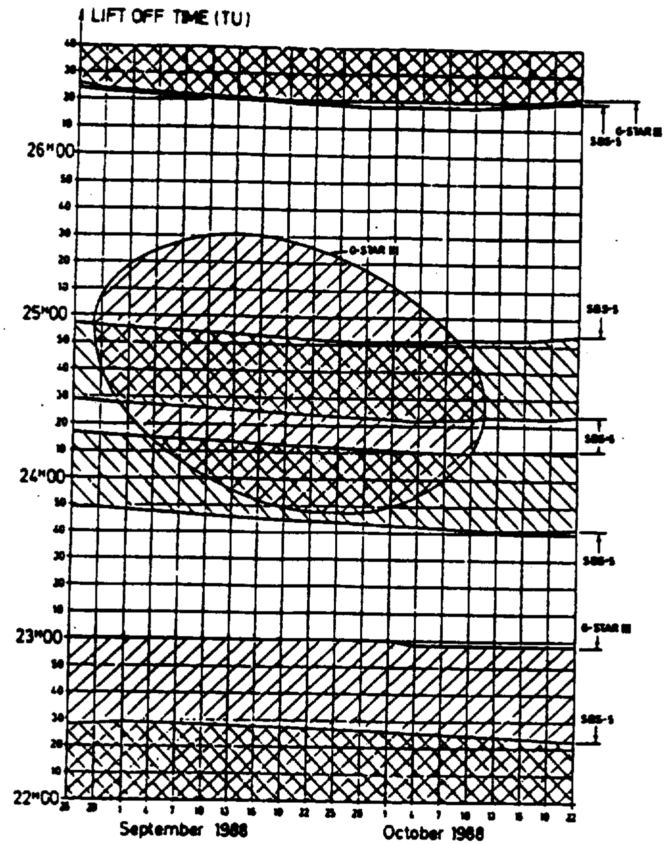

Figure 3. Combined Launch Window Universal Time at Liftoff

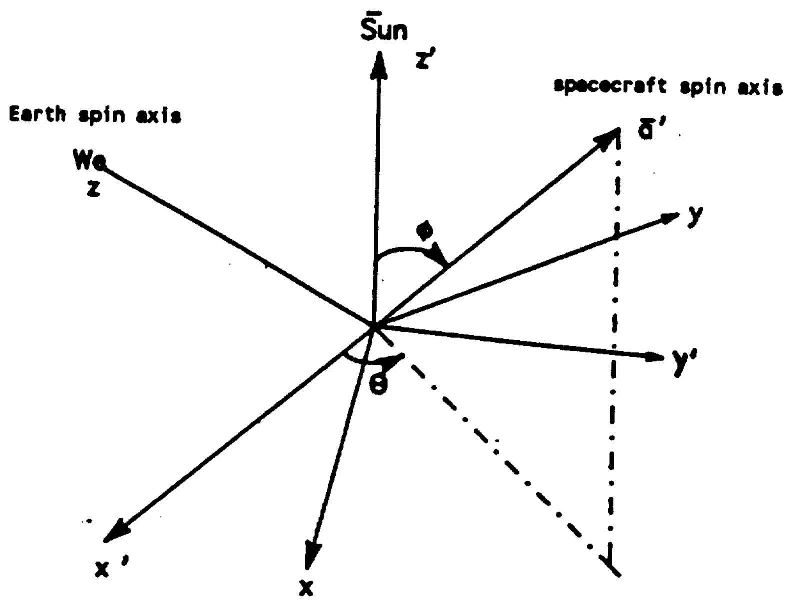

Figure 4. Phi-Theta System

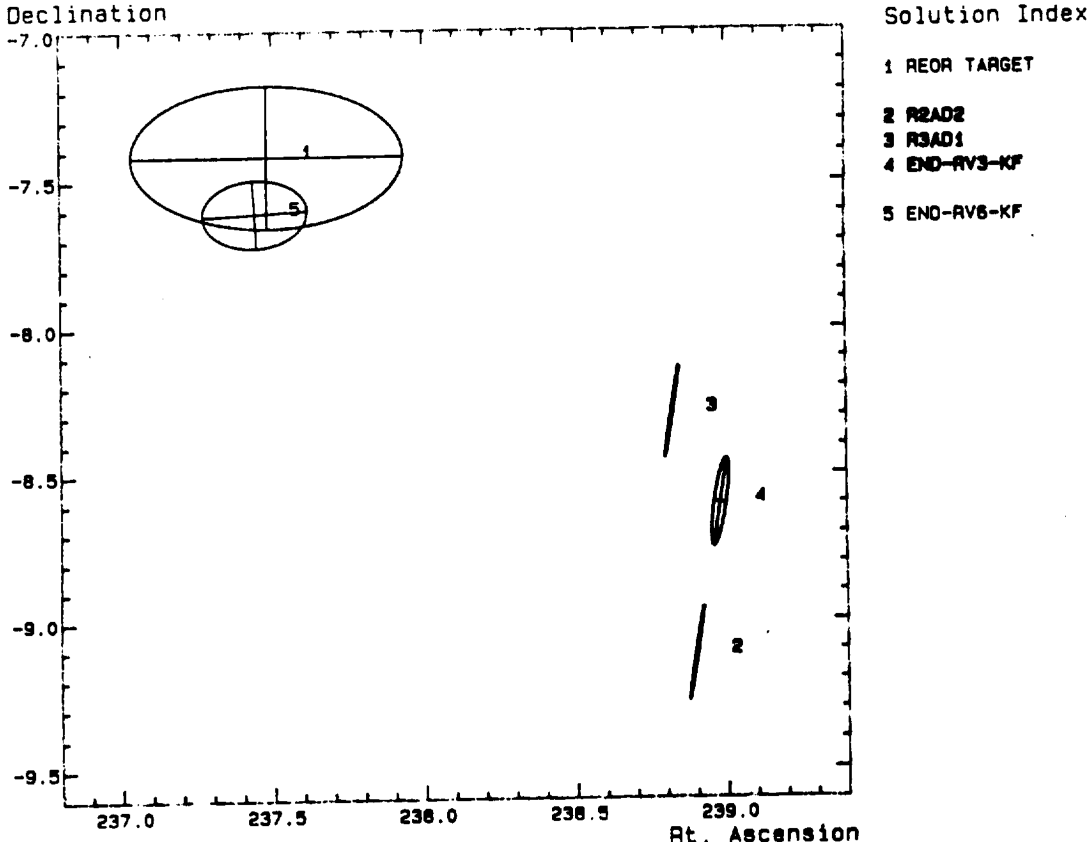

Figure 5. First Attitude Touchup

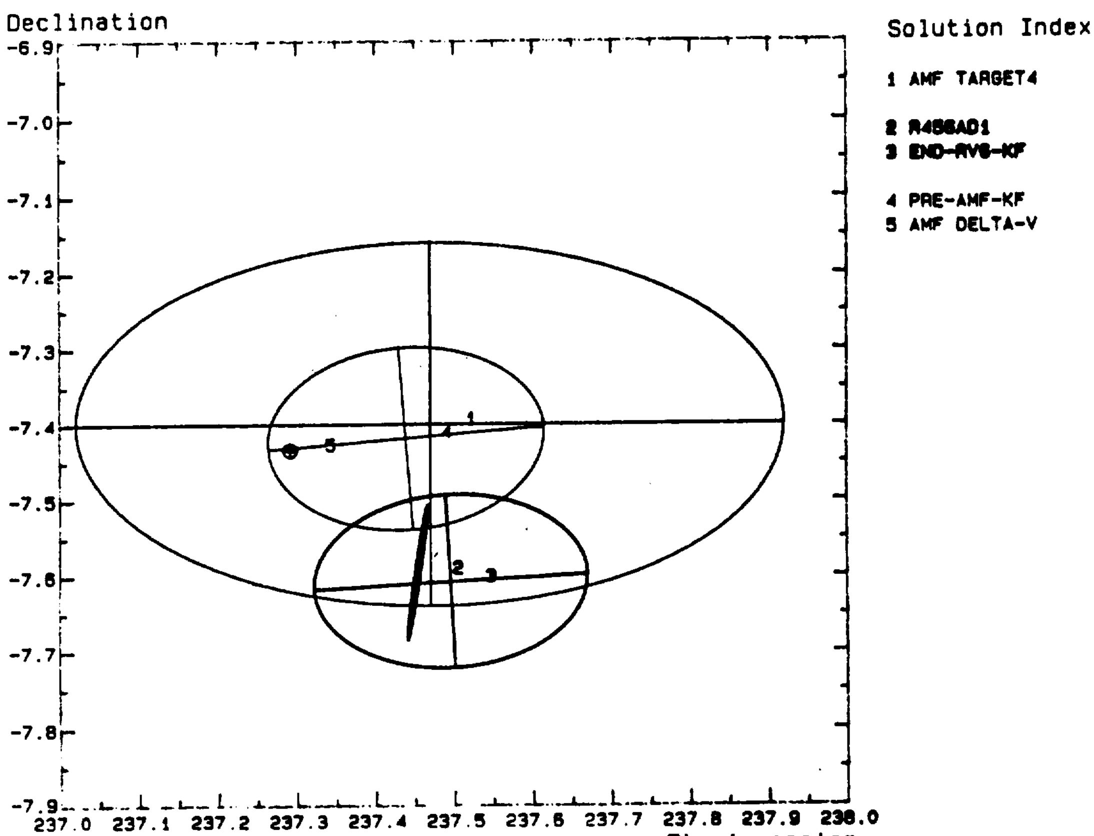

Figure 6. Second Attitude Touchup

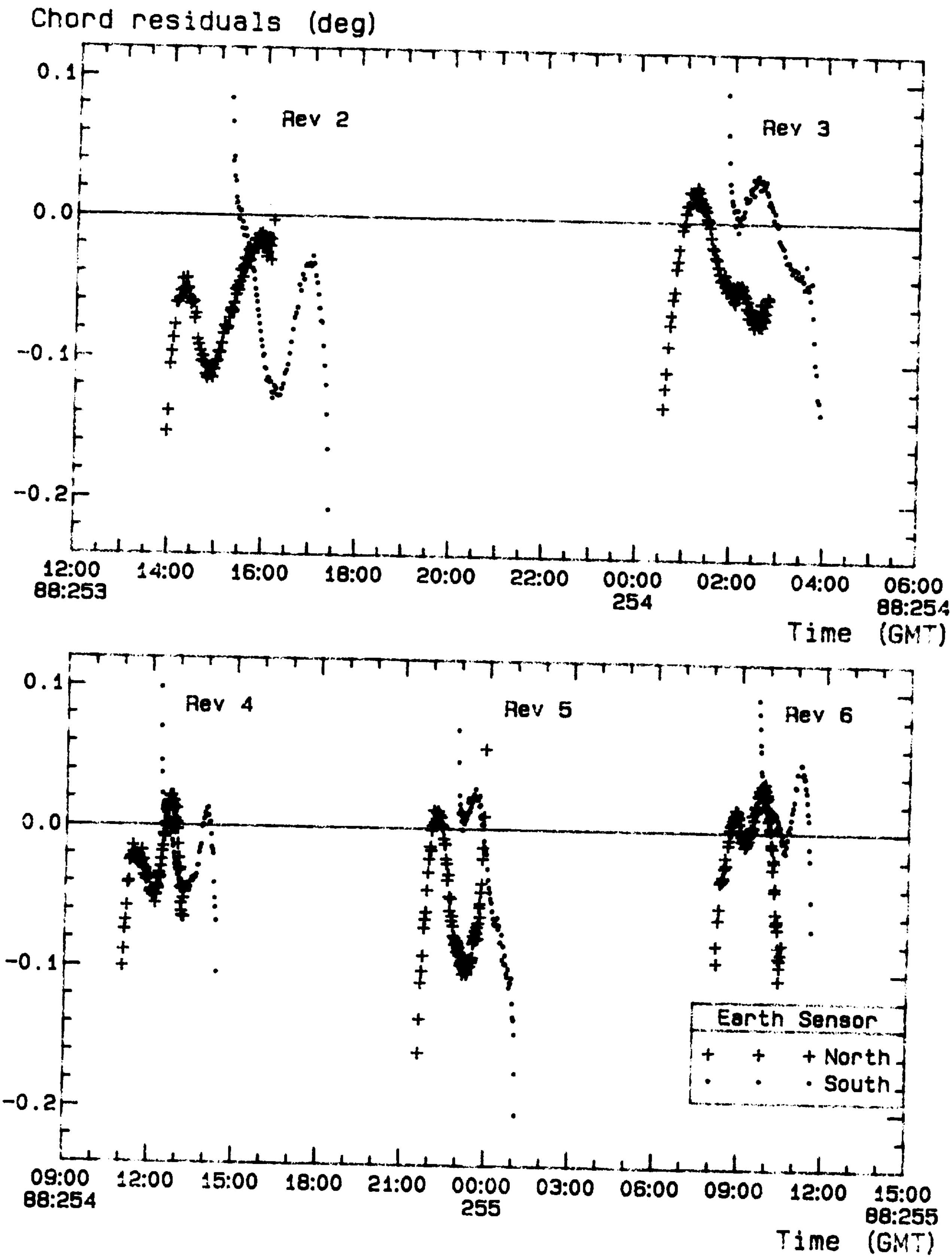

Figure 7. Earth Chord Residuals for Revs 2 Through 6

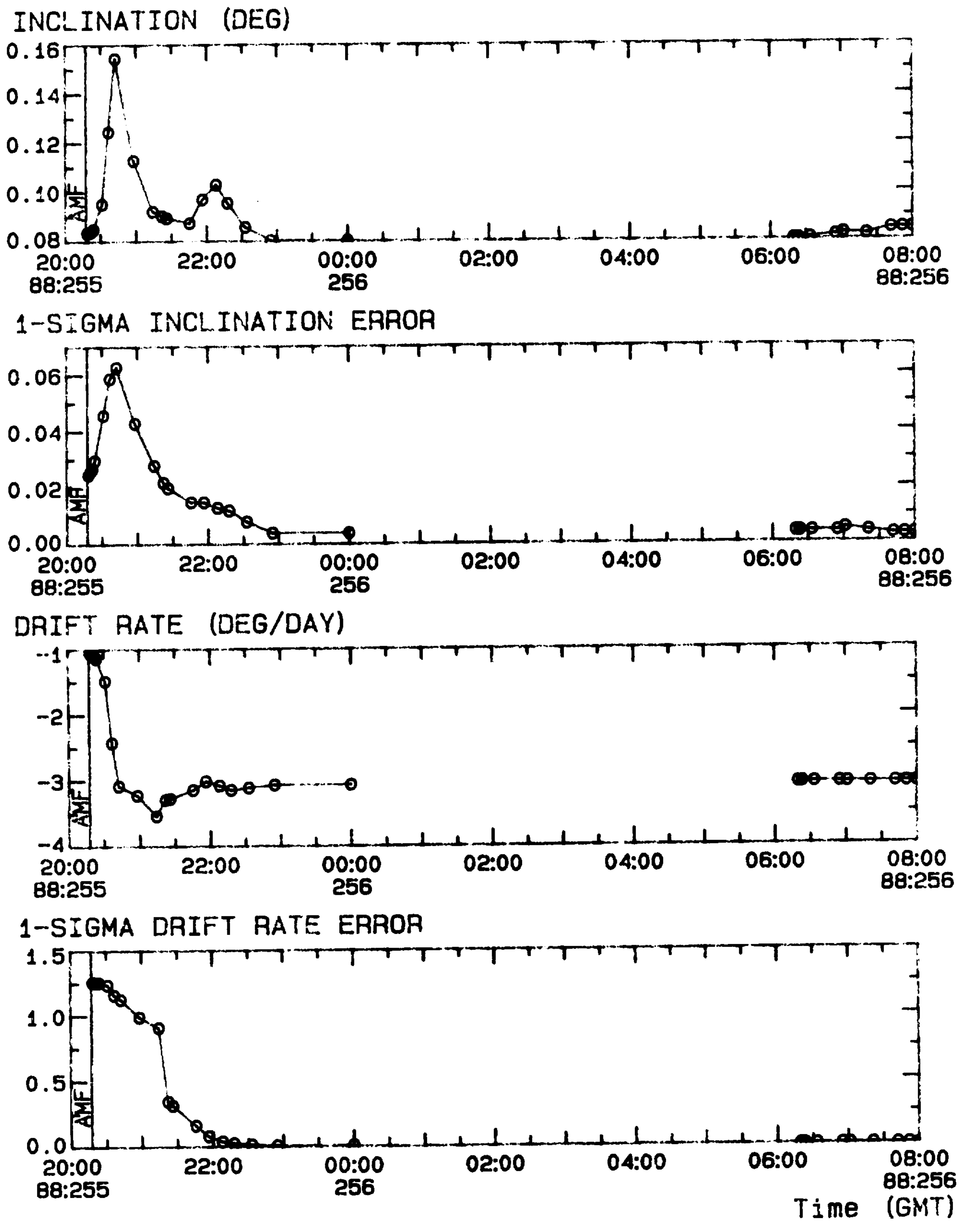

Figure 8. Observed Drift Rate, Inclination, and 1-Sigma Errors Following Apogee Motor Firing

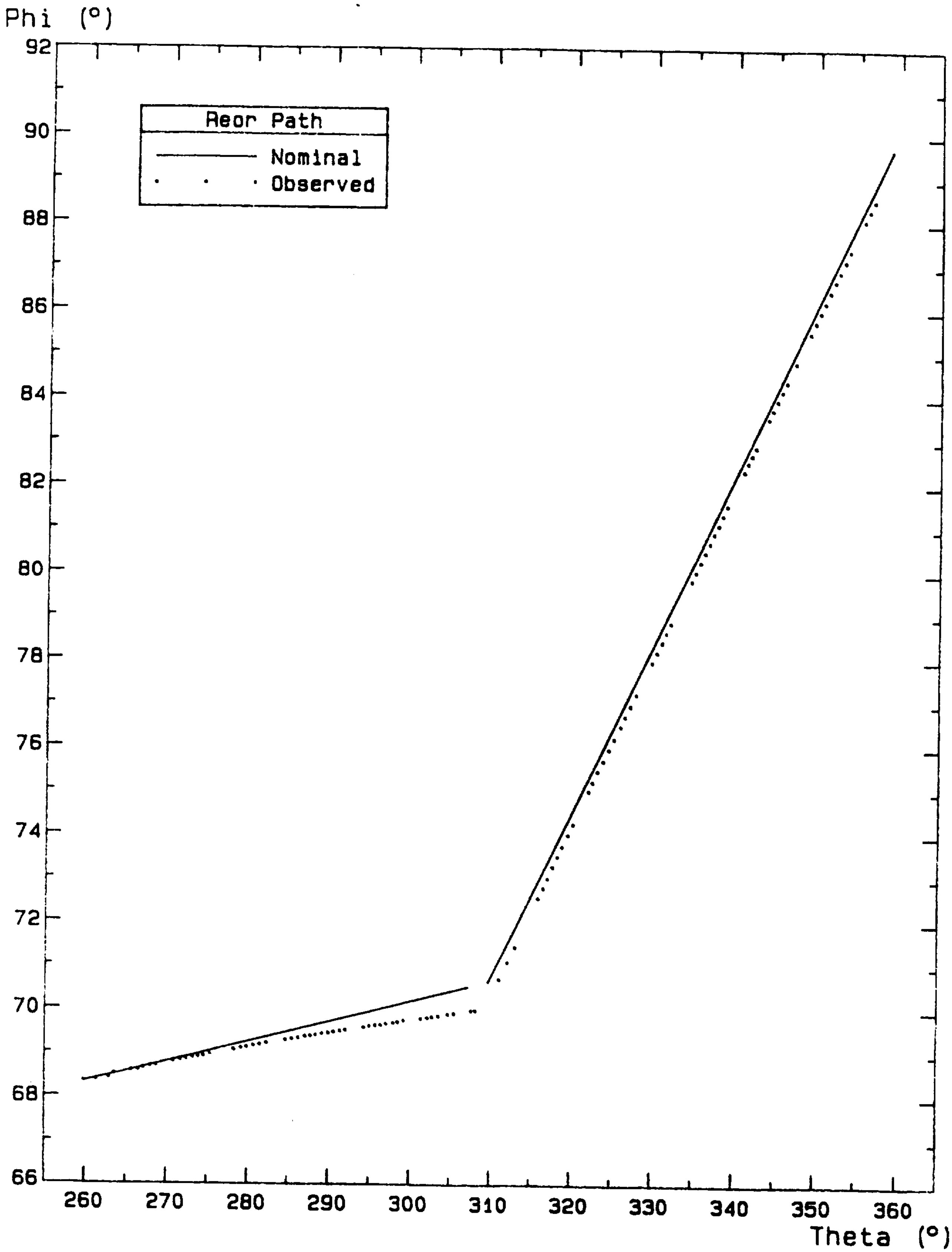

Figure 9. SBS-5 Reorientation to Orbit Normal Attitude