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SPIN-AXIS ATTITUDE DETERMINATION PROGRAM FOR THE GEOSYNCHRONOUS TRANSFER ORBIT SPACECRAFT

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ABSTRACT

Three types of spin-axis attitude determination program for the geosynchronous transfer orbit spacecraft are developed. Deterministic closed-form algorithm, batch least-square algorithm and stabilized Kalman filter algorithm are used for implemetation of three programs. EUROSTAR bus model from British Aerospace is used for attitude sensor modelling. Attitude determinations using three programs are performed for the simulated sensor data according to INMARSAT 2-F1 prelaunch mission analysis.

1. INTRODUCTION

Most of geostationary communications satellites are spin stabilized during their transfer orbit phase. At that time spacecraft attitude is usually expressed as single-axis attitude in the inertial reference frame, i.e. spin-axis right ascension and declination. The Sun sensor and horizon sensor mounted on the spacecraft provide the sensor data for attitude determination on the ground. The attitude sensor design is different according to the manufacturers. Therefore the attitude determination program has to be coded with respect to sensor design.

The real-time dynamic spacecraft simulator from British Aerospace has been operated by our section and the spacecraft ground control and analysis software has been developed for the simulator as a target spacecraft. The spacecraft simulator software is based on the BAe's EUROSTAR bus. EUROSTAR bus is the

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baseline model for INMARSAT-2 and TELECOM-2. According to EUROSTAR bus model, ESS(Earth-Sun Sensor) is used as attitude sensor during transfer orbit. Computer software which determines the spacecraft attitude during the transfer orbit phase using ESS data in the telemetry stream is developed. The software consists of three types of program based on the attitude determination methods. Firstly, DETSPAD(DETerministic SPin-axis Attitude Determination) is based on the cone-intersect method which determines the spacecraft attitude analytically using each sensor data set. Secondly, BATSPAD(BATch SPin-axis Attitude Determination) is based on the weighted least square method which iteratively estimates the spacecraft attitude using whole batch of the sensor data set. Finally, KALSPAD(KALman SPin-axis Attitude Determination) is based on Kalman filter algorithm which sequentially processes each of sensor data set. Each program is executed separately and then the results are compared. This software is directly applied to the spacecraft based on EUROSTAR bus model and easily applied to the other spacecraft after modifying the modules for sensor modelling.

In this paper, the attitude sensor data using developed attitude sensor simulation program are obtained, and then the various ranges of random noise are added to the simulated sensor data, and finally spacecraft attitude is determined by three types of program using the simulated sensor data including random noise. Inmarsat 2-F1 prelaunch mission analysis data are used for attitude sensor simulations.

Section 2 depicts the ESS sensor modeling and processing for obtaining sun aspect angle and earth nadir angle from sensor data set. Section 3 describes the algorithm for each program DETSPAD, BATSPAD, and KALSPAD. Section 4 presents results of attitude determination for each program. Section 5 concludes the paper.

2. ATTITUDE SENSOR MODELLING AND SENSOR DATA PROCESSING

The spacecraft based on EUROSTAR bus from British Aerospace uses ESS(Earth-Sun Sensor) as an attitude sensor during spin-stabilized transfer orbit phase. The ESS contains a vertical Sun slit, an inclined Sun slit and two pencil beam Earth pippers(BAe, 1991). A vertical sun slit produces the reference event(Sun Reference Pulse : SRP) and an inclined sun slit produces an information about the sun direction(Sun Elevation Pulse : SEP). The sun slits have 160° fields of view, and the field of the 28° inclined slit crosses the vertical slit in the plane of the box base. The Earth pippers have 1°.5 beamwidths and its infrared beams detects the space-earth and earth-space transitions. The Earth pippers are inclined at 6° and 10° above the ESS box base in the same plane as the vertical sun slit. The vertical sun slit cuts the XY plane nominally 14° from -X towards -Y, and ESS box base is canted at an elevation of 7°.5 from the XY plane towards +Z. The geometrical configuration of ESS is depicted in Figure 1.

The following six attitude sensor data are telemetered by ESS, namely the spin rate and the five phases relative to the SRP.

- spin rate
- Sun phase(Sun Elevation Pulse)
- Earth sensor 1 space-Earth pulse
- Earth sensor 1 Earth-space pulse
- Earth sensor 2 space-Earth pulse
- Earth sensor 2 Earth-space pulse

Sun phase is the output from V-slit sun sensor and the sun aspect angle is derived from the spherical triangle which is composed of spin-axis direction, sun direction and crossing point of V-slit sun sensor(Chen and Lerner 1984) as the following.

$$\tan(\beta + \psi_{ESS}) = \tan \frac{\theta_0}{\sin \omega \Delta t} \tag{1}$$

where β is sun aspect angle, ψ_{ESS} ESS box canted angle, θ_0 is the angle between vertical sun slit and inclined sun slit, ω is the spin rate and Δt is the time interval between SRP and SEP.

The Earth widths scaned by two Earth pippers are calculated from the differences between space-Earth/Earth-space pulses. The modelling of Earth horizon sensor is

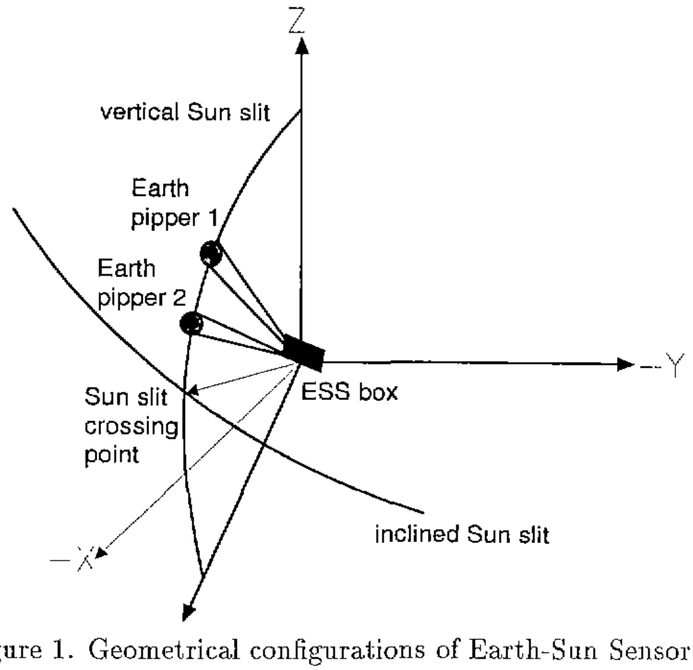

Figure 1. Geometrical configurations of Earth-Sun Sensor(ESS)

derived from the spherical triangle composed of spacecraft spin-axis direction, Earth nadir direction and Earth sensor pointing direction(Hotovy 1984).

$$
\begin{aligned}
\cos\rho &= \cos(\gamma_1 + \psi_{ESS})\cos\eta + \sin(\gamma_1 + \psi_{ESS})\sin\eta\cos\left(\frac{\Omega_1}{2}\right) \\
\cos\rho &= \cos(\gamma_1 + \psi_{ESS})\cos\eta + \sin(\gamma_1 + \psi_{ESS})\sin\eta\cos\left(\frac{\Omega_2}{2}\right) \\
\eta &= \arctan\left\{\frac{\cos(\gamma_2 + \psi_{ESS}) - \cos(\gamma_1 + \psi_{ESS})}{\cos(\gamma_1 + \psi_{ESS})\cos(\frac{\Omega_1}{2}) - \sin(\gamma_2 + \psi_{ESS})\cos(\frac{\Omega_2}{2})}\right\}
\end{aligned}
\tag{2}
$$

where ρ is the apparent angular radius of the Earth at the position of the spacecraft, γ_1 and γ_2 are inclined angles of the Earth pippers, Ω is the Earth width sensed by Earth horizon sensor, and η is the Earth nadir angle.

The attitude sensor data processing calculates Sun aspect angle and Earth nadir angle for each sensing time using the Sun phase, Earth scan width 1 and Earth scan width 2. The spacecraft spin-axis attitude is determined using the data set of Sun aspect angle data and Earth nadir angle data.

3. THEORY OF SPIN-AXIS ATTITUDE DETERMINATION PROGRAM

3-1. DETSPAD(DETerministic SPin-axis Attitude Determination)

DETSPAD program analytically determines the spacecraft attitude (spin-axis right ascension and declination) from Sun aspect angle (β) and Earth nadir angle (η) using two-cone intersect method. Two-cone intersect method finds the position (spacecraft spin-axis) at which Sun cone and Earth cone are intersected. And the solution consists of real and imaginary parts. Analytically, this kind of geometrical problem is composed of the following simultaneous equations.

$$
\begin{aligned}
\hat{A}\cdot\hat{S} &= \cos\beta \\
\hat{A}\cdot\hat{E} &= \cos\eta \\
\hat{A}\cdot\hat{A} &= 1
\end{aligned}
\tag{3}
$$

where $\hat{S}$ is the unit vector of Sun position, $\hat{E}$ is the unit vector of Earth position and $\hat{A}$ is the unit vector of attitude which has to be solved. Also is the Sun aspect angle and is the Earth nadir angle.

Equation (3) is solved in equation (4) using the Grubin (1977)'s method.

$$
\begin{aligned}
x &= \frac{(\cos\beta - \hat{E}\cdot\hat{S}\cos\eta)}{(1-(\hat{E}\cdot\hat{S})^2)} \\
y &= \frac{(\cos\eta - \hat{E}\cdot\hat{S}\cos\beta)}{(1-(\hat{E}\cdot\hat{S})^2)} \\
z &= \pm\left\{\frac{(1 - x\cos\beta - y\cos\eta)}{(1-(\hat{E}\cdot\hat{S})^2)}\right\}^{\frac{1}{2}} \\
\hat{A} &= x\hat{S} + y\hat{E} + z\hat{C}
\end{aligned} \tag{4}
$$

where $\hat{C} = \hat{S} \times \hat{E}$.

The vector components of rectangular coordiante system in equation(3) is transformed to spin-axis right ascension(α) and declination(δ) of spherical coordinate system. The real attitude and imaginary attitude from z component are determined by processing many sensor data using the phenomenon that the time variations of real solutions are much smaller than those of the imaginary solutions.

3-2. BATSPAD(BATch SPin-axis Attitude Determination)

BATSPAD determines the spacecraft spin-axis right ascension and declination using the least-square method up to 300 sun aspect angle and earth nadir data sets. The state vectors are spin-axis right ascension and declination, and these are described in cartesian coordinate system in equation (5).

$$
\begin{aligned}
\vec{x} &= [\alpha, \beta]^T \\
\vec{A} &= A_x\vec{X} + A_y\vec{Y} + A_z\vec{Z} \\
A_x &= \cos\alpha \ \cos\delta \\
A_y &= \sin\alpha \ \cos\delta \\
A_z &= \sin\delta
\end{aligned} \tag{5}
$$

Equation (6) describes the observation vector which consists of n pair of sun aspect angle and earth nadir angle from preprocessed sensor telemetry data.

$$
\bar{y}_{obs} = [\beta_{o1}, \beta_{o2}, \cdot, \cdot, \cdot, \beta_{on}, \eta_{o1}, \eta_{o2}, \cdot, \cdot, \cdot, \eta_{on}]^T \tag{6}
$$

Sun aspect angle and Earth nadir angle are calculated from a priori spacecraft spin axis attitude (A) for differential correction, and the calculated observation vector is described in equation (7).

$$
\begin{aligned}
\beta_{ci} &= \cos^{-1}(\vec{S}_i \cdot \vec{A}_i) \\
&= \cos^{-1}(S_{xi} \cdot A_{xi} + S_{yi} \cdot A_{yi} + S_{zi} \cdot A_{zi}) \\
\eta_{ci} &= \cos^{-1}(\vec{E}_i \cdot \vec{A}_i) \\
&= \cos^{-1}(E_{xi} \cdot A_{xi} + E_{yi} \cdot A_{yi} + E_{zi} \cdot A_{zi}) \\
\bar{y}_{cal} &= [\beta_{c1}, \beta_{c2}, \cdot, \cdot, \cdot, \beta_{cn}, \eta_{c1}, \eta_{c2}, \cdot, \cdot, \cdot, \eta_{cn}]^T
\end{aligned} \tag{7}
$$

The differential correction matrix is defined in equation (8), which is partial differential derivatives of the observation with respect to the state vector. Each component is described in equation (9) (Fallon and Rigteink 1984).

$$
F_k = \begin{bmatrix} \frac{\partial \beta_1}{\partial \alpha} & \frac{\partial \beta_1}{\partial \delta} \\ \cdots & \cdots \\ \frac{\partial \beta_n}{\partial \alpha} & \frac{\partial \beta_n}{\partial \delta} \\ \frac{\partial \eta_1}{\partial \alpha} & \frac{\partial \eta_1}{\partial \delta} \\ \cdots & \cdots \\ \frac{\partial \eta_n}{\partial \alpha} & \frac{\partial \eta_n}{\partial \delta} \end{bmatrix}_k \tag{8}
$$

$$
\begin{aligned}
\frac{\partial \beta_i}{\partial \alpha} &= \frac{S_{xi} \sin \alpha \cos \delta - S_{yi} \cos \alpha \cos \delta}{\sqrt{1 - (S_{xi} \cos \alpha \cos \delta + S_{yi} \sin \alpha \cos \delta + S_{zi} \sin \delta)^2}} \\
\frac{\partial \beta_i}{\partial \delta} &= \frac{S_{xi} \cos \alpha \sin \delta - S_{yi} \sin \alpha \sin \delta}{\sqrt{1 - (S_{xi} \cos \alpha \cos \delta + S_{yi} \sin \alpha \cos \delta + S_{zi} \sin \delta)^2}} \\
\frac{\partial \eta_i}{\partial \alpha} &= \frac{E_{xi} \sin \alpha \cos \delta - E_{yi} \cos \alpha \cos \delta}{\sqrt{1 - (E_{xi} \cos \alpha \cos \delta + E_{yi} \sin \alpha \cos \delta + E_{zi} \sin \delta)^2}} \\
\frac{\partial \eta_i}{\partial \delta} &= \frac{E_{xi} \cos \alpha \sin \delta - E_{yi} \sin \alpha \sin \delta}{\sqrt{1 - (E_{xi} \cos \alpha \cos \delta + E_{yi} \sin \alpha \cos \delta + E_{zi} \sin \delta)^2}}
\end{aligned} \tag{9}
$$

where S is the position vector of the Sun and E is the position vector of the Earth with respect to the spacecraft-centered coordinate system.

Equation (10) gives the solved state vector in the (k+1)th iteration.

$$
\bar{X}_{k+1} = \begin{bmatrix} \alpha_k \\ \delta_k \end{bmatrix} + \left[\begin{bmatrix} \sigma_\alpha^{-2} & 0 \\ 0 & \sigma_\delta^{-2} \end{bmatrix} + \sigma_\theta^{-2} F_k^T F_k \right]^{-1} [\sigma_\theta^{-2} F_k^T (\bar{y}_{obs} - \bar{y}_{cal}] \tag{10}
$$

where σ_α^{-2} and σ_δ^{-2} are estimated accuracy of the state vector and σ_θ^{-2} is the estimated accuracy of the observation.

3-3. KALSPAD(KALman SPin-axis Attitude Determination)

KALSPAD determines the spacecraft attitude using Kalman filter which sequentially processes the Sun aspect angle and Earth nadir angle data sets. Bierman (1973)'s stabilized Kalman covariance update method is used. The same state vector and observation vector are used for this program as BATSPAD.

Equations (11) describes the core equation for the stabilized Kalman filter algorithm(Capellari *et al.* 1976).

$$\begin{aligned}
\hat{x}(t_{i+1} \mid t_i) &= \Phi(t_{i+1} \mid t_i)\hat{x}(t_i \mid t_i); predictedstate \\
P(t_{i+1} \mid t_i) &= \Phi(t_{i+1}, t_i)P(t_i \mid t_i)\Phi(t_{i+1}, t_i)^T + Q_{i+1} \\
&\quad ; predictederrorcovariance \\
K(t_{i+1}) &= P(t_{i+1} \mid t_i)F_{i+1}^T[F_{i+1}P(t_{i+1} \mid t_i)F_{i+1}^T + w_{i+1}^{-1}]^{-1} \\
&\quad ; Kalmangain \qquad (11) \\
\hat{x}(t_{i+1} \mid t_{i+1}) &= \hat{x}(t_{i+1} \mid t_i) + K(t_{i+1})[\Delta y(t_{i+1})]; updatedstate \\
P(t_{i+1} \mid t_{i+1}) &= [I - K(t_{i+1})F_{i+1}]P(t_{i+1} \mid t_i)[I - K(t_{i+1})F_{i+1}]^T \\
&\quad + K(t_{i+1})w_{i+1}^{-1}K^T(t_{i+1}); updatederrorcovariance
\end{aligned}$$

where $\Phi, Q_{i+1}, F_{i+1}, \Delta y(t_{i+1})$, and w_{i+1}^{-1} denote state transition matrix, covariance of state noise, partial derivative of observation wrt state vector, measurement residual, variance of the observation, respectively.

3-4. Orbit Propagation and Sun Ephemeris Generation

Spacecraft orbit determination and propagation should precede attitude determination. The Earth position with respect to the spacecraft can be calculated based on the previously determined and propagated orbit at the time of attitude collection. Also, the Sun position should be calculated at each time of observation for preprocessing the sun sensor data. Currently implemented orbit propagator algorithm is Markely and Jeletic(1991)'s. This orbit propagator is simple analytic one incorporating a mean-to-osculating transformation and J2 terms. The algorithm used for the Sun ephemeris generation is Flandern and Pulkkinen(1979)'s.

4. ATTITUDE DETERMINATION SIMULATIONS

INMARSAT 2-F1 and F2 spacecraft were manufactured based on EUROSTAR bus. Accordingly, ESS(Earth-Sun Sensor) was used as an attitude sensor during transfer orbit phase. INMARSAT 2-F1 spacecraft was launched by DELTA II 6925 on Mar. 8, 1991. INMARSAT 2-F1 spacecraft was injected to transfer orbit by Thiokol Star 48B solid rocket which was 3rd stage of DELTA II launcher.

Table 1. INMARSAT 2-F1 Transfer Orbit Elements(guidance tape 2) Time from Lift-off ; 1572.433 sec.

a(km)	e	i(deg)	Ω (deg)	ω (deg)	ν (deg)
24852.562	0.7359075	24.02646	170.92144+GSTi	179.87003	14.4829

Table 2. INMARSAT 2-F1 Attitude Maneuver during transfer orbit(tape 2).

+ Z spinning	Injection 1st slew	intermediate	2nd slew 1st AEF
spin-axis R.A.	$81°.6$+GSTi	$172°.9$+GSTi	$260°.5$+GSTi
spin-axis Dec.	$-8°.6$	$+85°.0$	$-17°.3$
spin rate(rpm)	$23.7 \rightarrow 16 - 11$	$\rightarrow 12 \rightarrow$	$11 \rightarrow 13$

In this section, INMARSAT 2-F1 prelaunch mission analysis data are used for simulation of developed attitude determination program. The 2nd tape of 4 transfer orbit injection states is described in Table 1(BAe, 1990).

After 20 minutes from transfer orbit injection, the spacecraft is separated from 3rd stage of launcher and then recovered from flat spin, rotating about Z-axis. When the spacecraft rotates about +Z direction, the slew angle required from injection attitude to 1st AEF attitude is about 210° and this angle is achieved by two separate spin precession maneuvers. The spacecraft attitude variations during that time are summarized in Table 2. Here, GSTi is Greenwich Sidereal Time at injection time. Intermidiate attitude is selected satisfying the sun angle constraints, maintaining good coverage, and raising the perigee by 1st slew maneuver.

The mission control center on the ground accomplishes the following activities for the attitude maneuvers in Table 2.

- tracking data collection
- orbit determination
- sensor data collection
- attitude determination
- spin rate control
- attitude maneuver

Injection time for attitude sensor data simulation is at 1:30 UT, Oct. 30, 1990 and at this time, Greenwich sidereal time is 60.610° in angle.

Table 3 shows the mission time-line related to attitude maneuver and determination from transfer orbit injection to apogee engine firing(BAe, 1990). From

Table 3. Mission Timeline for Attitude Maneuver and Determination.

TIME	EVENTS	TIME	EVENTS
5:20	Apogee 1	23:10	Start Collection of Attitude
6:05	Begin Spin Down to 16 rpm		Data(r ¿ 25000 km)
6:45	End Spin Down	25:20	End Collection of Attitude Data
6:50	Deploy Stability Booms		Attitude Determination
6:55	Begin Spin Down to 11 rpm	25:50	Start Attitude Trim Maneuver
7:20	End Spin Down	26:05	End Attitude Trim Maneuver
7:20	Start 1st Part of Slew	26:52	Apogee 3
8:50	End 1st Part of Slew		
12:15	Start Collection of Attitude	33:55	Start Collection of Attitude
	Data(r > 25000 km)		Data(r > 25000 km)
13:00	End Collection of Attitude	36:05	End Collection of Attitude
	Data		Data
	Attitude Determination		Attitude Determination
13:45	Adjust Spin Rate for Slew	36:35	Verify Attitude for AEF 1
	Maneuver	37:38	Apogee 4
14:00	End Spin Adjust	37:42	Start AEF 1
14:20	Start 2nd Part of Slew	38:06	End AEF 1
15:50	End 2nd Part of Slew	39:00	Adjust Spin Rate to 13 rpm
16:06	Apogee 2	39:15	End Spin Adjust
18:15	Adjust Spin Rate to 13 rpm		
18:30	End Spin Ajust		

the injection time to the AEF 1 time at 4th apogee, the Earth width sensed by ESS is simulated in Figure 2 using the orbit/attitude parameter in Table 1 and Table 2. In Figure 2, the orbital elements is fixed to that of Table 1 and the attitude is different for each stage of transfer orbit according to Table 2. Figure 2 presents each events in mission time-line of Table 3. ESS output above 25,000 km from Earth center is used for attitude determination. After 3rd apogee, i.e., attitude is established for AEF, attitude sensor data can be collected during two and half hours, from 3.5 hrs before apogee to 1 hr before apogee. After collecting the sensor data, final attitude determination has to be accomplished for Apogee Engine Firing.

Each program determines the spacecraft attitude using simulated sensor data

Table 4. The parameter variation due to the iterative calculation (initial spin-axis $\alpha = 321°.0, \delta = -17°.0$).

Iter.	$\Delta\alpha$	$\Delta\delta$	α	δ	$\sigma\alpha$	$\sigma\delta$
1	.109E+00	-.246E+00	321.1085	-17.2457	.21E-03	.64E-03
2	.869E–03	-.170E–02	321.1094	-17.2474	.21E-03	.64E-03
3	.499E–05	-.154E–04	321.1094	-17.2474	.21E-03	.64E-03
4	.426E–07	-.132E–06	321.1094	-17.2474	.21E-03	.64E-03
5	.365E–09	-.113E–08	321.1094	-17.2474	.21E-03	.64E-03
6	.310E–11	-.970E–11	321.1094	-17.2474	.21E-03	.64E-03

prior to 3rd apogee in Figure 2. At this time, sensor data for attitude determination are the data set which presents one Sun sensor output and two Earth sensor outputs, and the Earth widths are above 5 degree.

Sensor data satisfying this condition was simulated from 0:41:00 to 2:48:30 Oct 31, 1990 and the number of data pair was 255 in 30 second interval. Random numbers of ranging $\pm 0.5°, \pm 0.1°, \pm 0.05°$ were added to the 255 attitude sensor data set and attitude determination program used it as attitude sensor data including random noise. The processed Sun angle and Earth nadir angle from this data set are depicted in Figure 3.

DETSPAD program finds the position of intersecting two cones analytically, each representing Sun aspect angle and Earth nadir angle. DETSPAD can be used when there is no a priori attitude information for batch processing. Figure 4, Figure 5, and Figure 6 represents the attitude determined by DETSPAD program using 255 sensor data set including random noises. Figure 4 represents variation of spin axis right ascension due to the sensor data and Figure 5 represents that of spin-axis declination. The exact solution of spin-axis right ascension equals 321°.110 and declination equals $-17°.3$ for all three simulations. Figure 6 describes spin-axis declinations vs. right ascensions. In Figure 5, spin-axis declination is poorly determined near the crossing point of real and imaginary solutions. That is caused by the geometrical configurations of Sun, Earth and spacecraft attitude.

BATSPAD program basically finds a good solution for the sensor data including random noise. The results of BATSPAD program for 255 sensor data 4sets having $\pm 0^{circ}.5$ random noise is in Table 4. In this case, iterative calculation is terminated based on the variation of the spin-axis right ascension and declination.

KALSPAD program processes the attitude sensor data sequentially. Figure 7, Figure 8 and Figure 9 represent the spin-axis right ascensions and declinations for each data sequence. In Figure 7 and Figure 8, the outer envelop is the solution for the sensor data set which has $\pm 0°.5$ random noise, inner envelop is for $\pm 0°.05$

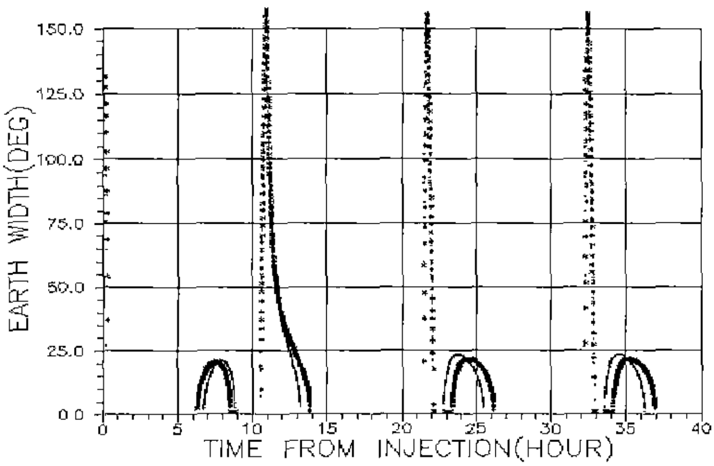

Figure 2. Earth width sensed by ESS during transfer orbit.

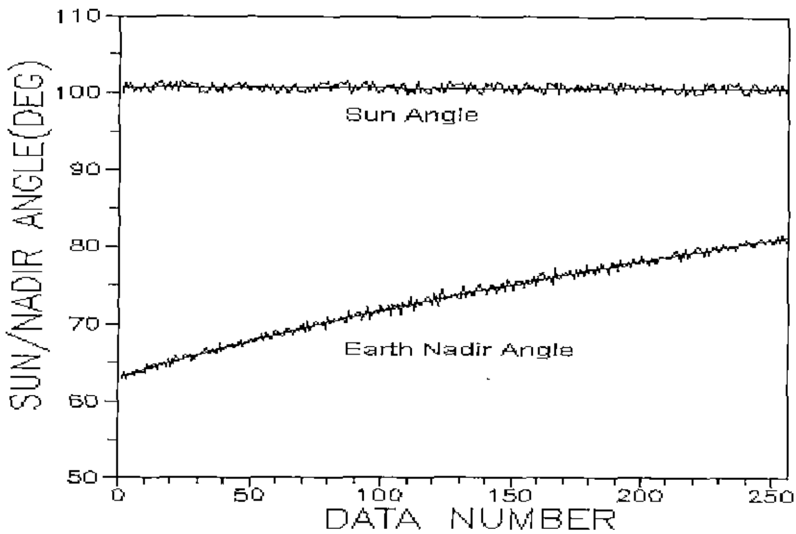

Figure 3. Processed Sun aspect angle and Earth nadir angle (0°.5 random error).

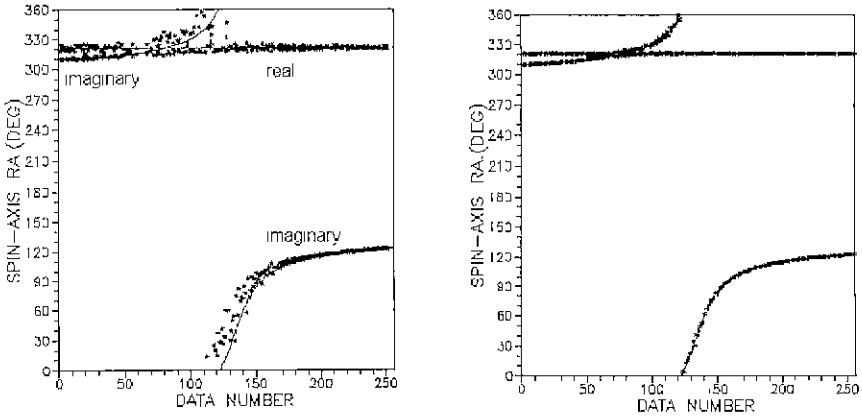

Figure 4. Spin-axis right ascension for sensor data (left; 0°.5 random error, right; 0°.05 random error).

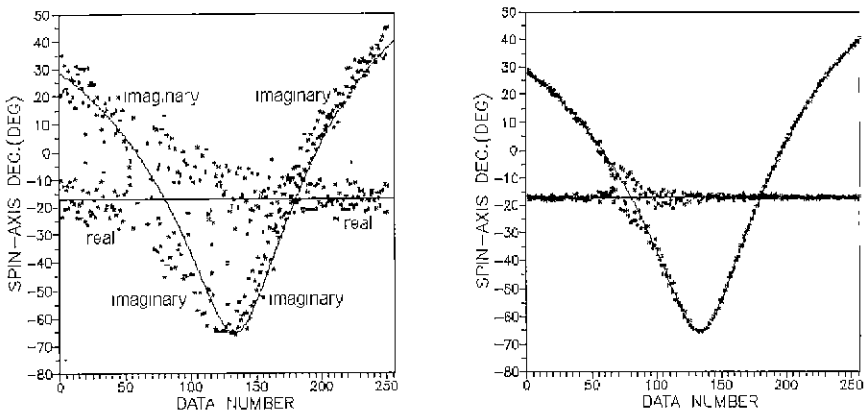

Figure 5. Spin-axis declination for sensor data (left; 0°.5 random error, right; 0°.05 random error).

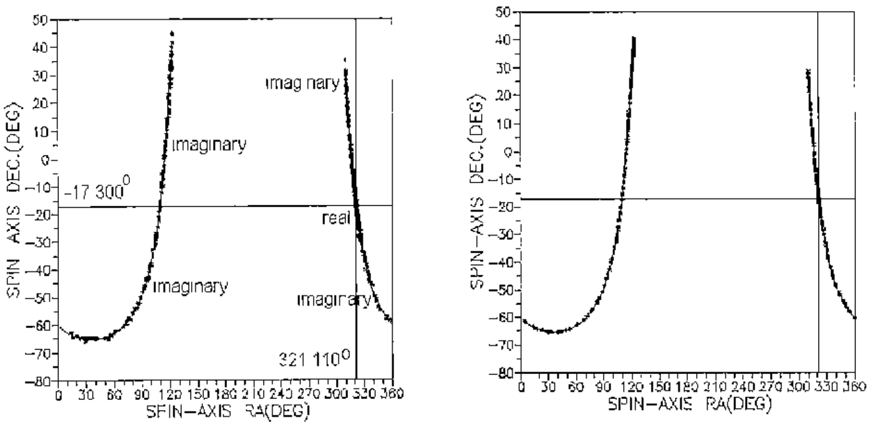

Figure 6. Spin-axis right ascension vs. declination (left; 0°.5 random error, right; 0°.05 random error).

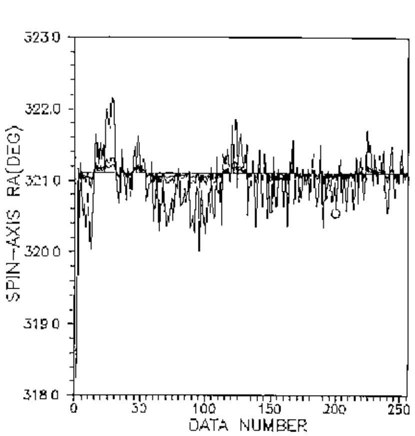

Figure 7. Spin-axis right ascension (KALSPAD).

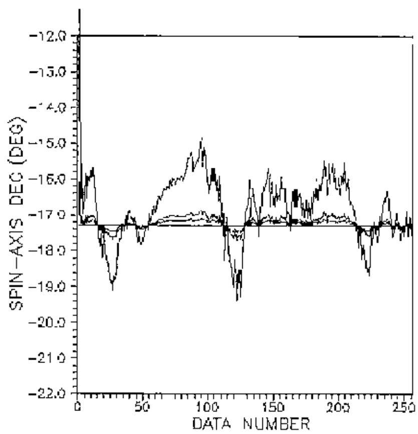

Figure 8. Spin-axis declination (KALSPAD).

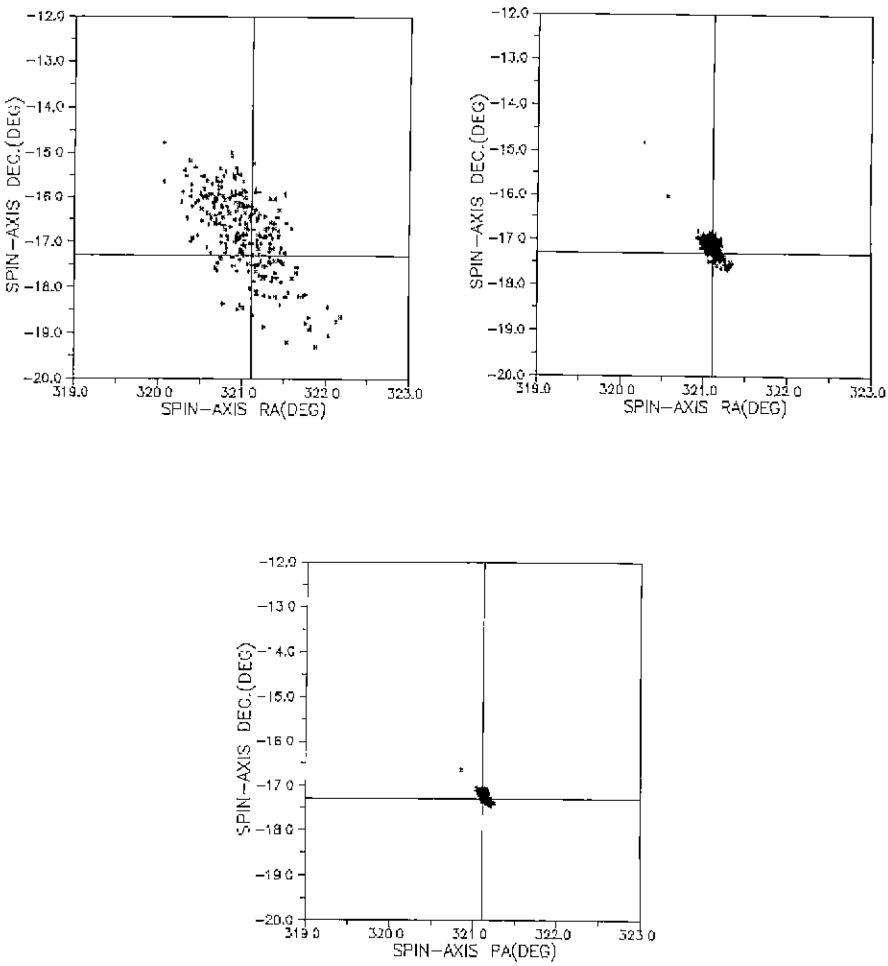

Figure 9. Spin-axis right ascension vs. declination(KALSPAD) (left top; 0°.5 random error, right top; 0°.1 random error, bottom; 0°.05 random error).

random noise and middle envelop is for $\pm 0^\circ.1$ random noise. In Figure 8, spin-axis declination is also poorly determined in the area of the real/imaginary crossing point, same as in Figure 5.

5. CONCLUSION

The results of attitude determination obtained from DETSPAD (DETerministic SPin-axis Attitude Determination) can be used in BATSPAD (BATch SPin-axis Attitude Determiniation) and KALSPAD (KALman filter SPin-axis Attitude Determination) as an inital value. Also, the results of the DETSPAD shows the geometrical configurations of the solutions. BATSPAD, however, is less affected by random errors. So, we have to rely on the BATSPAD for the final attitude solutions. KALSPAD can be used for real-time monitoring of the spacecraft attitude. These three attitude determination programs can be applied to the KOREASAT spacecraft transfer orbit mission after changing the spacecraft sensor model.

At this time, three attitude determination programs are all stand-alone programs, that is, it includes spacecraft orbit propagator and Sun/Moon ephemeris generator. Simple orbit propagator/ephemeris generator are implemented for the simplicity of the program. But for precise attitude determinations, the attitude determination algorithm itself would be included in the Flight Dynamics Software (FDS) which contains precise orbit propagator and ephemeris generator. Also, covariance analysis and the more state vectors including sensor biases have to be added in the attitude determination program. In the near future, the attitude determination program for the insufficient sensor data pair, when one of the attitude sensors is failed, has to be developed for the complete mission operations.

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