Proceedings of the 13th KACC, October 1998

Precision Attitude Determination Design Using Star Tracker

Seung-Wu Rhee and Zeen-Chul Kim

Satellite Bus Dept., Space Division

Korea Aerospace Research Institute (Tel : 82-42-860-2447; Fax : 82-42-860-2603 ; E-mail : srhee@viva.kari.re.kr)

Abstract

Star tracker placement configuration is proposed and the properness of the placement configuration is verified for star tracker's sun avoidance angle requirement. Precision attitude determination system is successfully designed using a gyro-star tracker inertial reference system for a candidate LEO spacecraft. Elaborate kalman filter formulation for a spacecraft is proposed for covariance analysis. The covariance analysis is performed to verify the capability of the proposed attitude determination system. The analysis results show that the attitude determination error and drift rate error are good enough to satisfy the mission of a candidate spacecraft.

1. Introduction

Recently, many research results come out to accomplish a precise attitude estimate using Kalman filter method incorporated with gyro set and star sensor or sun sensor and earth sensor depending on the design philosophy of attitude determination [1,2].

The Attitude Determination and Control System (ADCS) of KOrea Multi-Purpose SATellite (KOMPSAT)[3] whose design had been finished, contains three sets of Kearfott's two-axis rate integrating gyros, two sets of Ithaco's earth sensors providing roll and pitch axis attitude measurements, and Daewoo Heavy Industry's (DHI) two sets of analog fine sun sensors for yaw axis attitude measurement. The required pointing accuracy for the cartographic mission is 0.1deg(2σ) for roll and pitch axes, and 0.15 deg(2σ) for yaw axis.

It is expected that the mission of next satellite will require precision pointing information to satisfy the allowable geolocation error for an earth surface-imaging payload. As is mentioned, the IR-earth sensor is used for attitude measurement in KOMPSAT and the analysis results showed that most part of AD error for each axis is due to the earth radiance effect. Thus, it is speculated that a gyro-earth sensor configuration has a limitation in improving attitude determination capability. For the ADCS of precision satellite, it is thought to be utilized a gyro-star tracker inertial reference system to meet the required attitude knowledge and stability for the successful mission.

It is obvious that many parameters of gyro such as scale factor stability, cross-axis alignment error, systematic drift rate and random drift rate have an effect on the geolocation error if there is no attitude correction with a celestial measurement. Thus, an earth sensor and a fine sun sensor or a star tracker is used for this purpose. Among these sensors, a star tracker is known to be one of the most precise sensors measurable less than one arc-minute. Like any other sensors, a star tracker measurement is also corrupted with star tracker CCD noise, and biased with a centroid error and an internal alignment error. The sensor bias and noise will also degrade the geolocation accuracy, as well as the pointing accuracy. Along with the bias and noise, the star tracker's field of view(FOV) may be obscured by lunar interference, which will also decrease the attitude correction capability. On-board attitude information in flight software is usually obtained through quaternion propagation incorporated with the corrected rate and attitude information. In flight software, depending on how frequently attitude correction using celestial measurement will be executed and how long lunar gap interval we may have, the attitude knowledge capability will be decided.

In this study, a reasonable star tracker placement configuration will be suggested. And the main part will focus on the enhancement of the pointing capability and stability for the proposed system using a gyro-star tracker inertial reference system depending on attitude correction and lunar gap interval. Kalman filter technique will be utilized by incorporating these sensor measurements for attitude and gyro drift rate estimate, and covariance analysis will be performed so as to verify its capability.

2. Star Tracker Configuration

It is assumed that the selected orbit is a sun-synchronous orbit, which has the inclination angle of 98.3 degree and has the sun angle of 17.5 degree in the orbit plane. The altitude of 685Km is selected for the analysis and its orbit period is 98 minute. For this orbit, a star tracker configuration is selected by considering the sun rejection angle of 30 degree as following :

Considering the seasonal variation of sun vector in ECI frame, the sun vector is expressed as

$$\hat{S}_{ECI} = [Cos(\delta) \quad 0 \quad Sin(\delta)]^T \tag{1}$$

where δ denotes the seasonal variation of sun angle,

$-23.5degree \le \delta \le 23.5degree$.

The sun vector in ECI frame can be transformed into the orbital frame, so called, a local vertical local horizontal frame using a ECI-to-Orbit transformation matrix, $R_{ECI2Orbit}$, which is

$$\hat{S}_{Orbit} = R_{ECI2Orbit} \quad \hat{S}_{ECI} \tag{2}$$

where $\hat{S}_{Orbit}$ denotes the sun vector in the orbital frame.

Also, the field of view(FOV) vector of star tracker in orbit frame can be obtained by transforming the FOV vector of star tracker in a satellite body frame after post-multiplying the transformation matrix from the body to the orbit frame, $R_{Body2Orbit}$ as.

$$\hat{ST}_{Orbit} = R_{Body2Orbit} \quad \hat{ST}_{Body} \tag{3}$$

where $\hat{ST}_{Orbit}$ and $\hat{ST}_{Body}$ are FOV vectors of star tracker in the orbit frame and in the satellite body frame. respectively.

Then, using the dot product rule, the sun angle with respect to the FOV vectors of star tracker can be simply expressed as

$$\beta = Cos^{-1}[\hat{S}_{Orbit} \bullet \hat{ST}_{Orbit}] \tag{4}$$

3. Gyro and Star Tracker Model

The gyro model is used as proposed by Farrenkopf[3].

$$\vec{\omega}_g = \vec{\omega} + \vec{b}_g + \vec{\eta}_{g1} \tag{5}$$

$$\dot{\vec{b}}_g = \vec{\eta}_{g2} \tag{6}$$

where $\vec{\omega}_g$ is the measurement output of gyro set, the vector $\vec{b}_g$ denotes the drift rate, $\vec{\eta}_{g1}$ is the zero mean gaussian gyro drift rate noise and $\vec{\eta}_{g2}$ is the zero mean gaussian gyro drift rate random walk.

Then the estimated rate can be expressed as

$$\hat{\vec{\omega}} = \vec{\omega}_g - \hat{\vec{b}}_g \tag{7}$$

$$\dot{\hat{\vec{b}}}_g = 0 \tag{8}$$

Incorporated with gyro measurement data, a star tracker are used for the attitude determination. Generally speaking, a star tracker consists of an optical part, an electric unit and a software. The optical part acquires the photon signal which is detected by CCD array detector and the photon signal is transmited to the electric unit. The software resides in the electric unit, which performs a star indetification and a numerical calculation for the azimuth and elevation angle of the indentifyed star on the focal plane. Some star tracker electronic unit may provide only the azimuth and elevation angle information of the indentifyed star on the focal plane. However, most of star trackers have also the capability of providing attitude information in quaternion form as well as Eular angle. In this study, it is assumed that a star tracker can provide an attitude information in quaternion, and the star tracker model is simply taken as the linear combination of mean measurement, sensor bias and measurement noise

4. Dynamic Equations[4,5] and Kalman Filter [6,7]

Suppose the current true attitude denoted by $\bar{q}$ is defined by first rotating the body by an amount of the current attitude estimation $\hat{q}$ and then rotating the body by a small angle error $\bar{q}_e$. Then the current true attitude expressed in quaternion notation has the form as

$$\bar{q} = \hat{q} \otimes \bar{q}_e \tag{9}$$

where $\otimes$ denotes quaternion multiplication.

After differentiating Eqn(9) and using the quaternion estimate kinematic relation $\dot{\hat{q}} = \frac{1}{2}\hat{q} \otimes \{\hat{\omega}, 0\}$, post-multiplication of the conjugate quaternion $\hat{q}^*$ and simple manipulation produces

$$\dot{\bar{q}}_e = \{-\hat{\vec{\omega}} \times \bar{q}_e, 0\} + \frac{1}{2}\bar{q}_e \otimes \{\vec{\omega} - \hat{\vec{\omega}}, 0\} \tag{10}$$

where $\bar{q}_e$ is the vector part of quaternion.

Using Eqn(5) through Eqn(8), Eqn(10) is modified as

$$\dot{\bar{q}}_e = \{-\hat{\vec{\omega}} \times \bar{q}_e, 0\} + \frac{1}{2}\bar{q}_e \otimes \{-\Delta\vec{b} - \vec{\eta}_{g1}, 0\} \tag{11}$$

$$\Delta\dot{\vec{b}} = \vec{\eta}_{g2} \tag{12}$$

where $\Delta\vec{b} = \vec{b}_g - \hat{\vec{b}}_g$, $\vec{\eta}_{g1}$ denotes the zero mean gaussian gyro drift rate noise and $\vec{\eta}_{g2}$ the zero mean gaussian gyro drift rate random walk same as Eqn(5) and (6).

Assuming small angle approximation $q_e \cong \frac{\varphi_e}{2}$ and neglecting higher than the first order term $\vartheta(2)$, eqn(11) becomes

$$\dot{\vec{\varphi}}_e = -\hat{\vec{\omega}} \times \vec{\varphi}_e - \Delta\vec{b} - \vec{\eta}_{g1} \tag{13}$$

If we express eqn(12) and (13) into a state-space form

$$\begin{Bmatrix} \dot{\vec{\varphi}}_e \\ \Delta\dot{\vec{b}} \end{Bmatrix} = \begin{bmatrix} -\hat{\vec{\omega}}\times & -I_{3x3} \\ 0_{3x3} & 0_{3x3} \end{bmatrix} \begin{Bmatrix} \vec{\varphi}_e \\ \Delta\vec{b} \end{Bmatrix} + \begin{Bmatrix} -\vec{\eta}_{g1} \\ \vec{\eta}_{g2} \end{Bmatrix} \tag{14}$$

where

$$\hat{\vec{\omega}}\times = \begin{bmatrix} 0 & -\hat{\omega}_z & \hat{\omega}_y \\ \hat{\omega}_z & 0 & -\hat{\omega}_x \\ -\hat{\omega}_y & \hat{\omega}_x & 0 \end{bmatrix}$$

Thus, the system matrix is assumed to be 6x6 dimension, the number of states is 6 for this study and $[\vec{\eta}_{g1} \quad \vec{\eta}_{g2}]^T$ is uncorrelated gaussian white noise processes with

$$E[\vec{\eta}_{g1} \quad \vec{\eta}_{g2}]^T = 0,$$

$$E\{[\vec{\eta}_{g1} \quad \vec{\eta}_{g2}]\,[\vec{\eta}_{g1} \quad \vec{\eta}_{g2}]^T\} = Q_F(t)\delta(t-\tau)$$

$$Q_F(t) = \begin{bmatrix} \sigma_{g1}^2 I_{3x3} & 0_{3x3} \\ 0_{3x3} & \sigma_{g2}^2 I_{3x3} \end{bmatrix}$$

where $Q_F(t)$ is the power spectrum density of system noise and σ_{g1}^2, σ_{g2}^2 denote the power spectrum densities for gyro drift rate noise and gyro drift rate random walk, respectively.

For the attitude measurement data, a star tracker are utilized. The measurement ouput of the estimate model is expressed as

$$\bar{z}_F(t) = H_F\bar{x}_F(t) + v(t) \tag{15}$$

where the measurement noise $v(t)$ is a zero mean gaussian noise,

$$E[v] = 0,$$
$$E[vv^T] = R(t)\delta(t-\tau) = \sigma_s^2 I_{3x3}$$

σ_s represents the noise standard deviation of a star tracker.

Usually the higher dimension of true model is considered incorporated with Eqn(14) for covariance analysis depending on how many states are considered in the true model. Suppose the true model has the form of

$$\dot{\bar{x}}_T = A_T\bar{x}_T + \bar{w}_T \tag{16}$$

$$\bar{z}_T(t) = H_T\bar{x}_T(t) + v(t)$$

and the estimate is given by

$$\dot{\hat{\bar{x}}}_F = A_F\hat{\bar{x}}_F + K[\bar{z}_T - \bar{z}_F] \tag{17}$$

The subscripts in Eqn(15) and (16) illustrates the fact that the matrices of true model and measurement output in Eqn(16) are different from Eqn(15).

Assuming there is no variation in the system dynamics over the sampling period, the constant transition matrix can be obtained. Then, the discrete form of the true model with a zero mean random noise sequence is described by

$$\bar{x}(k) = \Phi_T \bar{x}(k-1) + \Gamma \bar{\omega}(k-1) \tag{18}$$

$$\bar{z}(k) = H_T \bar{x}(k) + v(k)$$

Introducing that the best estimate is the conditional mean of given all previous data, the a priori estimate at time k can be found from the a posteriori estimate at time (k-1) by propagation using the deterministic system dynamics as Eqn(18). The discrete form of the a priori estimate model which means that propagating model without a measurement update, simply called an estimate model, is expressed as

$$\hat{\bar{x}}^-(k) = \Phi_F \hat{\bar{x}}^+(k-1) \tag{19}$$

$$\hat{\bar{z}}(k) = H_F \hat{\bar{x}}^-(k)$$

where

$$\Phi_F = \begin{bmatrix} \Phi_{11} & \Phi_{12} \\ 0_{3x3} & I_{3x3} \end{bmatrix} \text{ and}$$

$$\Phi_{11} = \begin{bmatrix} \cos\hat{\omega}_0 \Delta T & 0 & \sin\hat{\omega}_0 \Delta T \\ 0 & 1 & 0 \\ -\sin\hat{\omega}_0 \Delta T & 0 & \cos\hat{\omega}_0 \Delta T \end{bmatrix}$$

$$\Phi_{12} = \begin{bmatrix} \frac{-1}{\hat{\omega}_0}\sin\hat{\omega}_0 \Delta T & 0 & \frac{-1}{\hat{\omega}_0}(1-\cos\hat{\omega}_0 \Delta T) \\ 0 & -\Delta T & 0 \\ \frac{1}{\hat{\omega}_0}(1-\cos\hat{\omega}_0 \Delta T) & 0 & \frac{-1}{\hat{\omega}_0}\sin\hat{\omega}_0 \Delta T \end{bmatrix}$$ if only orbital rate $\hat{\omega}_0$ exists and other rate components are zero and ΔT denotes the attitude update interval.

The a posteriori estimate at time (k) by adding multiplication of the error between measurement and estimated output by Kalman filter to the a priori estimate, is given by

$$\hat{\bar{x}}^+(k) = \hat{\bar{x}}^-(k) + K[\bar{z}(k) - H_F \hat{\bar{x}}^-(k)] \tag{20}$$

Usually the true model of Eqn(18) described a real world, and the estimate model of Eqn(19) is not identical to the true model. Suppose the number of an estimated state is subset of the number of a true model state. Introducing the matrix W=[I 0] to account for the dimensional incompatibility between two models, the a priori estimate error can be expressed by

$$\tilde{x}^-(k) = W^T \hat{\bar{x}}^-(k) - \bar{x}(k) \tag{21}$$

where the matrix W=[I 0], I is an identity matrix and 0 is a null matrix and W has the property of $WW^T = I$.

The a posteriori estimate error can be expressed by

$$\tilde{x}^+(k) = W^T \hat{\bar{x}}^+(k) - \bar{x}(k) \tag{22}$$

Substituting Eqn (18) and (19) into Eqn (21) produces Eqn (23)

$$\tilde{x}^-(k) = W^T \Phi_F W[\tilde{x}^+(k-1) + \bar{x}(k-1)] - \Phi_T \bar{x}(k-1) - \Gamma \bar{\omega}(k-1) \tag{23}$$

Defining $\Delta\Phi = W^T \Phi_F W - \Phi_T$ and rearranging eqn(23) gives Eqn(24)

$$\tilde{x}^-(k) = W^T \Phi_F W \tilde{x}^+(k-1) + \Delta\Phi \bar{x}(k-1) - \Gamma \bar{\omega}(k-1) \tag{24}$$

The a priori error covariance extrapolation equation incoporated with Eqn(18) can transformed into a amtrix form as

$$\begin{bmatrix} P^-(k) & V^{-T}(k) \\ V^-(k) & U^-(k) \end{bmatrix} = \begin{bmatrix} W^T \Phi_F W & \Delta\Phi \\ 0 & \Phi \end{bmatrix} \begin{bmatrix} P^+(k-1) & V^{+T}(k-1) \\ V^+(k-1) & U^+(k-1) \end{bmatrix} \begin{bmatrix} W^T \Phi_F W & \Delta\Phi \\ 0 & \Phi \end{bmatrix}^T + \begin{bmatrix} Q(k-1) & -Q(k-1) \\ -Q(k-1) & Q(k-1) \end{bmatrix} \tag{25}$$

where $Q(k-1) = \int_{t_{k-1}}^{t_k} \Phi(t_k, s)\bar{\omega}\bar{\omega}^T \Phi^T(t_k, s) ds$ Similarly, when a measurement is made for update, the error covariance update equation using Eqn(20) and (22) is derived as

$$\begin{bmatrix} P^+(k) & V^{+T}(k) \\ V^+(k) & U^+(k) \end{bmatrix} = \begin{bmatrix} I - W^T K H_F W & -W^T K \Delta H \\ 0 & I \end{bmatrix} \begin{bmatrix} P^-(k) & V^{-T}(k) \\ V^-(k) & U^-(k) \end{bmatrix} \begin{bmatrix} I - W^T K H_F W & -W^T K \Delta H \\ 0 & I \end{bmatrix}^T + \begin{bmatrix} W^T K R K^T W & 0 \\ 0 & 0 \end{bmatrix} \tag{26}$$

where $\Delta H = H_F W - H_T$ and R is the power spectrum sequence of measurement noise.

Now, the covariance analysis can be performed by iterating Eqn(25) and (26).

5. Attitude Determination System Design

As is already mentioned, the attitude determination system consists of two star trackers and three axes rate integrated gyros for an inertial reference system as shown in Figure 1. Star tracker data in the form of quaternion is processing at 0.5 second step. Gyro data processing for rate information is designed to perform at every 0.25 second step and the residual calculation is set to conduct at every 0.5 second step, which will be fed to Kalman filter as a form of residual average. Attitude and drift rate error will be estimated and corrected at every 10 second step. To provide the current attitude errors to an attitude control system, attitude quaternion propagation will be executed at every 0.25 second period.

Star Tracker Placement Configuration Design and Discussion

According to the star tracker specification, the minimum of sun rejection angle should be greater than 30 degree and FOV vector of star tracker should be greater than 20 degree away from Earth albedo. It is assumed the +/-30 degree roll/ pitch maneuverings are required by a specified mission. Therefore, these requirements as well as the seasonal variations of sun angle should be considered for designing sensor configuration. As a result of design results according to Eqn(4) for two star trackers, the selected FOV vector of star trackers are:

ST#1 $=[0.5\ -0.5\ -0.5]^T$ and ST#1 $=[-0.5\ -0.5\ -0.5]^T$

Figure 2 depicts the star tracker placement configuration according to the selected FOV vector of star trackers. In the figure, Xs/c, Ys/c and Zs/c denote the roll, pitch and yaw axes of a spacecraft, respectively. As shown in the figure, FOV vectors are pointing 45 degree away from the Xs/c-Ys/c plane so as to satisfy the sun rejection angle requirement of 30 degree. Figure 3 shows the sun angle variation starting from the ascending node over one orbit flying time when +30 degree roll maneuver is conducted in winter season. As seen in figure, the star tracker #2 has a minimum sun angle of 38 degree when the spacecraft is flying over the latitude of about 10 degree North. The minimum sun angle of star tracker no.1 and no. 2 are summarized in Table 1 and 2 with respect to roll/pitch maneuvering angle of +/-30 degree, as well as seasonal variation: equinox, summer and winter. The results show that all of the minimum sun angles are greater than 30 degree sun rejection angle requirement.

Covariance Analysis Results and Discussion

The attitude determination capability of KOMPSAT[3] is listed in Table 3. As is mentioned before, the limit of improving attitude determination capability was experienced when an earth sensor was utilized for measurement. It is known that main reason of worse measurement accuracy in an earth sensor is the earth radiance effect. According to the analysis results of KOMPSAT system, the error amount of 0.043 degree out of 0.057 degree was due to the radiance effect in the roll axis and 0.087 degree of 0.093 degree for pitch axis.

For a candidate spacecraft, an attitude determination system is designed. So as to confirm that the attitude determination error and the drift error are within the required error budget of a candidate spacecraft, the covariance analysis is conducted in the absence of any lunar obscuration, as well as with lunar gap of 25 minute period per one orbit as the worst case. In the real world, two star trackers that are placed as in Figure 2 may be not interfered with lunar simultaneously. However, it is assumed for the analysis purpose as the worst case that two sensors are blocked simultaneously for two times of 12.5 minute period per one orbit. As is noted in Figure 1, the error update is performed at every 10 second step. For the analysis purpose, the selected gyro angle random walk is 5.8903e-6 deg/sqrt(sec) and the rate random walk is 2.5305e-8 deg/sqrt(sec^3) in 3 sigma. Star sensor bias of 10 arc-sec(3 sigma) per each axis is assumed and 15 arc-sec(3 sigma) for NEA.

Analysis results in 3 sigma are listed in Table 4 and 5 for the no lunar gap case and the 25 minute lunar gap case. Figure 4 and 5 shows the covariance analysis results for the 25 minute lunar gap case. As is noticed in Figure 4, attitude determination errors are decreasing to about the inherent star tracker bias while minimum number of stars are available for measurement and attitude errors are updating. While the sensor is blocking, attitude determination errors are increasing up to 12 arc-second(3 sigma). Figure 5 shows that roll and yaw axes have almost same gyro drift errors of 0.012 degree/hr because the roll and yaw axes of a spacecraft are coupled dynamically.

6. Conclusions

Two-star tracker placement configuration is studied and proposed for an attitude determination system of a candidate spacecraft after conducting a numerical simulation incorporated with seasonal sun angle variation. A precision attitude determination system is designed successfully to enhance the pointing capability of the LEO candidate spacecraft using a gyro-star tracker inertial reference system. Elaborate kalman filter formulation for a spacecraft is proposed for covariance analysis and covariance analysis is conducted based on the kalman filter formulation for a spacecraft. Covariance analysis results show that attitude determination error can be reduced to 12 arc-second and gyro drift error, 0.0122 degree/hr. using the proposed sensors.

Table 1. Minimum Sun Angle of Star Tracker No.1

Man. Angle(deg)	Equinox(deg)	Summer(deg)	Winter(deg)
Roll +30	40	41	38
Roll -30	72	72	68
Pitch +30	58	60	53
Pitch -30	58	60	53

Table 2. Minimum Sun Angle of Star Tracker No.2

Man. Angle(deg)	Equinox(deg)	Summer(deg)	Winter(deg)
Roll +30	41	42	38
Roll -30	72	73	68
Pitch +30	58	60	55
Pitch -30	45	60	55

Table 3. KOMPSAT AD Analysis Results with Earth Sensor and Fine Sun Sensor

Axis	AD Error(degree)-3sigma	Drift Error(deg/hr)-3sigma
Roll	0.057	0.34
Pitch	0.093	0.06
Yaw	0.094	0.18

Table 4. AD Analysis Results without Lunar Gap

Axis	AD Error(arc-sec)-3sigma	Drift Error(deg/hr)-3sigma
Roll	10.8	0.012
Pitch	10.4	0.0044
Yaw	11.65	0.012

Table 5. AD Analysis Results with 25 Minute Lunar Gap

Axis	AD Error(arc-sec)-3sigma	Drift Error(deg/hr)-3sigma
Roll	12	0.0122
Pitch	11.8	0.005
Yaw	12	0.012

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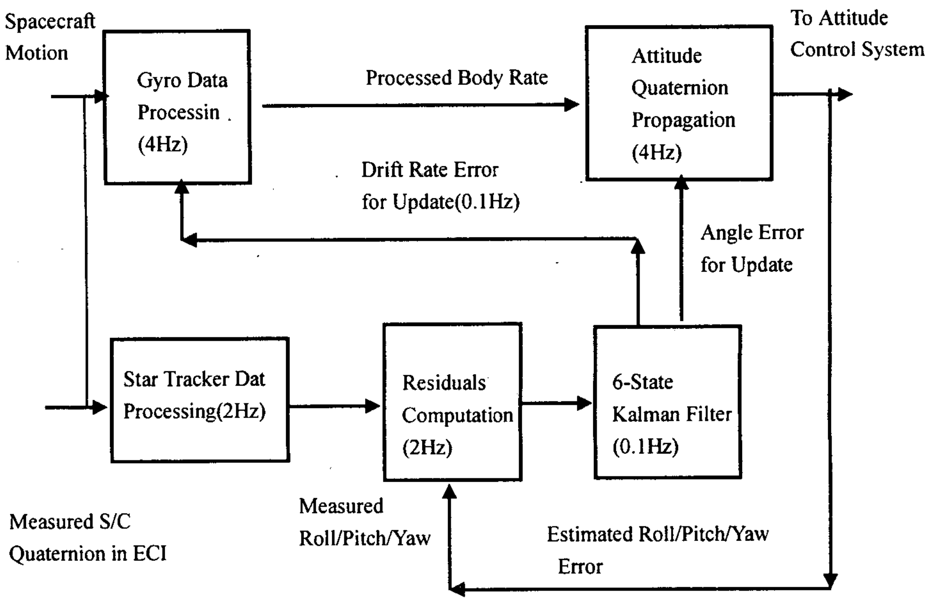

Figure 1. Functional Block Diagram of Attitude Determination System

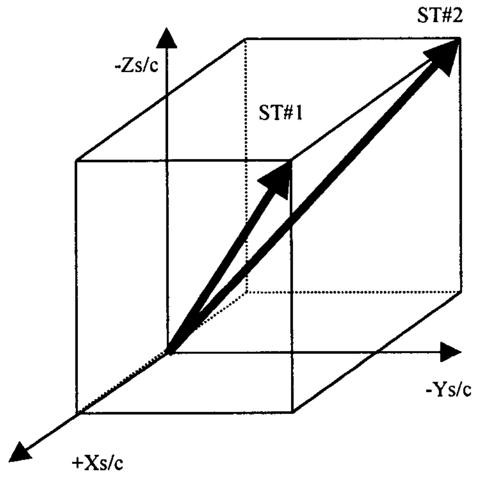

Figure 2. Star Tracker Placement Configuration

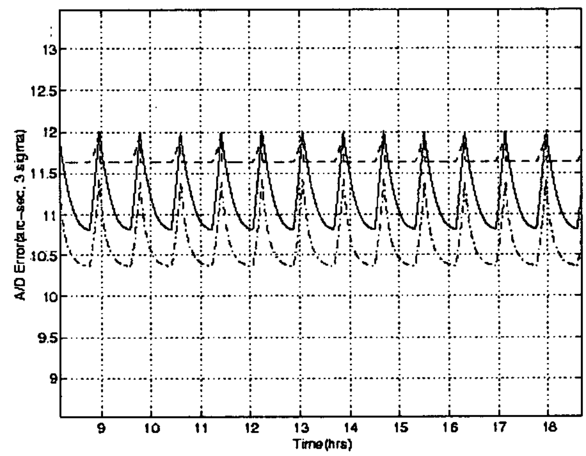

Figure 4. Attitude Determination Errors for Three Axes (solid : roll, dashed dot : pitch, dashed : yaw)

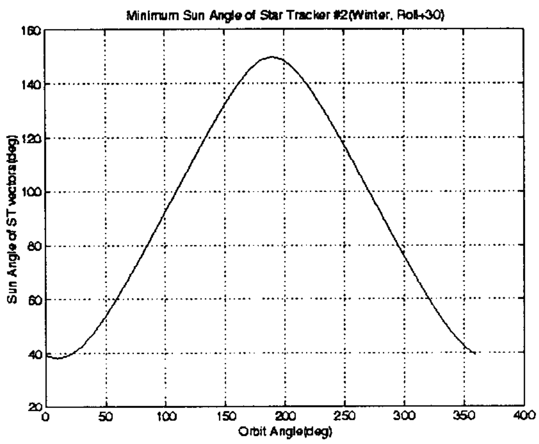

Figure 3. Minimum Sun Angle of Star Tracker #2(Winter. +30 degree Roll Maneuver)

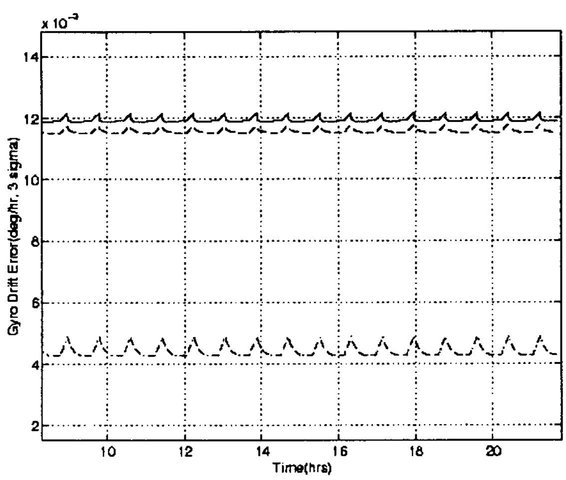

Figure 5. Gyro Drift Errors for Three Axes (solid : roll, dashed dot : pitch dashed : yaw)