STELLAR TRACKING ATTITUDE REFERENCE SYSTEM
System Application Study

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FOREWORD

The study described in this report was performed by the Guidance and Control Systems Laboratory, Technology Division, Space and Communications Group, Hughes Aircraft Company, El Segundo, California. The work was accomplished under NASA Contract NAS 5-21846, with Mr. H. Paul Scherer, Stabilization and Control Branch, Spacecraft Technology Division, Engineering Directorate, Goddard Space Flight Center, Greenbelt, Maryland as Technical Officer. The principal members of the project team were as follows:

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Preface

This final report documents the detailed results of a system application study, involving the use of the Hughes Stellar Tracking Attitude Reference System (STARS) as the attitude reference for precision pointing (with .001 degree accuracy) of an EOS spacecraft. The STARS accomplishes precision earth pointing by utilizing a single, clustered star tracker assembly mounted on a non-orthogonal, two gimbal mechanism, driven so as to unwind satellite orbital and orbit precession rates.

The fundamental objective of the system application study was to explore, by means of realistic preliminary design, analysis, and simulation, the limitations placed upon STARS performance in a typical low altitude, inclined orbit earth observation mission by the non-ideal characteristics of the system hardware and spacecraft behavior. A basic ground rule of the study was that gyros were not to be used to provide short period stabilization.

The overall conclusion reached in this study is that a relatively straightforward precision pointing system design, utilizing STARS on an EOS type spacecraft, and considering the characteristics of real sensing and control components, can achieve the desired .001 degree pointing capability. It was found that the lower limit on long term pointing precision was determined primarily by the accuracy obtainable from the star trackers. Short term pointing precision (jitter) was easily reduced to below 0.2 arcseconds by proper control system design, and both good jitter stability and excellent transient response characteristics were obtained without the use of gyros. Since all analyses and simulations so far have shown STARS to be capable of the desired pointing precision and stability, the next step in bringing STARS up to the level of a flight qualified system is the development of an engineering model and it is therefore recommended that this be undertaken next.
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1. Introduction

The requirements of various projected scientific, commercial, and defense oriented satellite missions indicate a strong need for a straightforward, conceptually simple, mechanically and electronically uncomplicated, and highly reliable attitude reference and precision pointing system, which has the potential of achieving and the possibility of eventually exceeding 0.001 degree pointing capability. In response to this need, Hughes Aircraft Company has evolved the STARS (Stellar Tracking Attitude Reference System) concept which avoids many of the problems associated with the precision pointing schemes by the use of a compact multistar tracker and a simple, unconventional two-gimbal assembly. The advantages of this approach are as follows:

1) The stellar reference is a single, clustered star tracker assembly on a single gimbal mount, thus providing a uniform thermal environment for its components, minimum distortion, and involving minimum interaction with other spacecraft components.

2) The STARS concept is not dependent upon high precision gyros, with their attendant reliability and cost disadvantages.

3) The star tracker concept provides a high level of redundancy in its basic design, being able to track more than the minimum required number of stars most of the time. In addition, it provides gradual degradation of performance in case of failure of one or more individual trackers.

4) The star tracker design and the particular scanning technique used avoid the usual requirements for critical alignment of the scanning device to the boresight axis.

5) The STARS concept operates with essentially constant rates about both gimbal axes, thereby avoiding many of the more significant sources of angular error normally associated with gimbal systems and their drives.

6) The STARS concept minimizes the amount of computational capability required of the on-board computer.
FIGURE 1-1. STARS - A GYROLESS INERTIAL PLATFORM
7) The STARS concept can operate without the conventionally required on-board star catalog since the locations of only eight preselected stars are required.

This concept was investigated in NASA funded feasibility study (contract NAS 5-21508) which ended in June 1971. The design configuration, developed in the feasibility study, utilized a clustered set of eight star trackers mounted on the inner gimbal of a nonorthogonal two-axis gimbal system. As illustrated diagrammatically in Figure 1-1, the inner gimbal is held essentially parallel to the earth's polar axis, while the outer gimbal is parallel to the satellite pitch axis. Under ideal conditions, the satellite may rotate about its pitch axis while keeping the yaw axis pointed precisely vertical, and the star trackers continue to point at their assigned stars while the orbit precesses about the polar axis. The star trackers are arranged and their signals processed so that two are tracking stars at any one time regardless of earth and satellite body occultation directions. Physical motion of the star trackers in inertial space to permit the tracking function is accomplished by controlling vehicle attitude. Precision pointing of the satellite vertical is achieved by commanding the proper pitch and polar axis gimbal angles, based on ground-determined and regularly updated ephemeris data. An on-board digital computer keeps track of local satellite time, calculates and sets proper gimbal angles, and performs the attitude control signal processing and command functions to enable star tracking as required.

This one year study showed that STARS appears to be a feasible attitude reference system and that no theoretical obstacles exist to a practical implementation. Systems analysis showed that a set of eight stars could indeed be found that would provide a continuous attitude reference for any arbitrary orbit. A straightforward attitude acquisition procedure was developed, and the on-board computational requirements were established. A conceptual design and layout of a suitable gimbal system for the star trackers was developed and the availability of the required components was established. Servo analysis and simulation of the gimbal positioning systems showed that precise positioning at the required position accuracies and rates is theoretically feasible. In addition, assembly and test methods for the STARS gimbal system were explored as an essential part of an overall feasibility study.
A detailed analysis and preliminary design of a star tracker system was performed in the feasibility study. This involved a system analysis of the proposed tracker system including optics, signal processing, and sunshade design. A mechanical design study resulted in a detailed preliminary mechanical layout of a tracker cluster and verification that the design concept was feasible. In addition a detailed error analysis of the total system, based on anticipated thermal and structural characteristics, showed that the STARS mechanism is inherently suitable for high precision attitude reference purposes.

An area excluded from consideration in the feasibility study was the precision pointing capability of a real spacecraft employing STARS for sensing, since such a study is spacecraft configuration dependent. A preliminary, linearized analysis for a typical spacecraft, performed since the completion of the feasibility study, has indicated however that under idealized sensing and torquing conditions, with pure angular rate feedback available, precision vertical pointing to 0.001 degree is practical. The questions that were not answered at that time, and that required further detailed study, concerned the limitations placed on precision pointing by realistic star sensor characteristics, signal processing requirements, control system bandwidth limitations due to sensor noise, and the requirement to generate attitude rate damping information from star sensor signals rather than a built-in gyro. These are the principal questions that were addressed by this system application study.

The study can be divided essentially into three parts: star tracker investigations, attitude control system design, and performance analysis and simulation. Each of these areas is discussed in turn on the following pages.
2. Star Tracker Performance

In order to develop more confidence in the ability of the STARS tracker to achieve the required performance in the postulated environment, the tracker noise model developed during the feasibility study was re-examined. Concurrently, to provide improved insight into photomultiplier noise characteristics obtained experimentally on another program, and to confirm the analytical approach, additional tests were performed on three photomultiplier tubes. The analysis and the experimental results are discussed below.

2.1 Analysis

The star detection process basically consists of setting a threshold level and defining voltages below that level "noise" and voltages above that level as "signal". The determination of the appropriate threshold level setting requires a statistical evaluation of the probability of a noise pulse being greater than this level (false alarm) and a signal + noise pulse being smaller than this level (missed pulse). This analysis is based on the assumption that a false alarm is equivalent to a missed pulse in respect to the perturbation it causes to the system's tracking loop.

The STARS star tracker uses photomultiplier tubes as optical detectors. The model that best describes the pulse amplitude distribution of the photomultiplier tube's noise output has been the principal subject of this analysis. It is this model that permits a determination of where to set the detection threshold and prediction of the false alarm and missed pulse rates for this threshold. In earlier analyses, gaussian statistics were used to describe both the signal and noise characteristics of the PMT output. Experience on the Space Precision Attitude Control System (SPACS) star sensor program found that this assumption yielded extremely optimistic results in comparison to laboratory measured performance.

The noise output of a PMT is primarily a result of PMT leakage current, star background irradiance, and illumination from off-axis bright sources such as sun, earth or moon being reflected and scattered by the sun shade into the field of view. Of these, the reflected and scattered solar illumination was considered to be the dominant performance limiting noise source as discussed in the STARS final report.

In addition to these sources of noise; space radiation, in particular high energy galactic particles, may contribute to the PMT output noise.
A study that was performed to explain the discrepancy between the measured and expected false alarm rates on the SPACS program found that the amplitude of the PMT output noise pulses exhibited an exponential distribution rather than gaussian. This distribution results in much higher false alarm rates as will be shown.

The present analysis includes an examination of the pulse amplitude distribution of three representative photomultiplier tubes operated at the conditions of background illumination and electrical bandwidth anticipated for the STARS sensor.

The solar irradiance that is scattered and reflected from the sun shade will result in an average (d.c.) output $I_B$ at the PMT anode. This current can be related to an equivalent signal irradiance at the entrance aperture of the optics by

$$H_B = \frac{I_B}{RG A_{\text{eff}}}$$

where

$R$ = Photocathode Responsivity
$G$ = Photomultiplier Gain
$A_{\text{eff}}$ = Effective Optical Aperture

The variation of background noise pulse amplitudes is generally described by a gaussian distribution for which the $\sigma$ standard deviation or rms noise is given by

$$i_n(\sigma) = \sqrt{2e \frac{A}{\delta - 1}} A_{\text{eff}} R H_B G^2 \Delta f \text{ (amp)}$$

where

$\delta$ = average gain of an individual dynode
$G$ = multiplier overall gain
$\Delta f$ = noise equivalent bandwidth

The effective signal irradiance equivalent of this noise current is given by

$$\delta = \frac{i_n}{A_{\text{eff}} RG} \quad \text{(watt cm}^{-2})$$

5
The baseline design parameters, applicable to this analysis, for the STARS sensor are shown in Table 2-1.

### TABLE 2-1 BASELINE DESIGN

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Photocathode Responsivity, ( R )</td>
<td>( 0.075 \text{ amp/watt} )</td>
</tr>
<tr>
<td>Effective Collecting Aperture, ( A_{\text{eff}} )</td>
<td>( 6.4 \text{ cm}^2 )</td>
</tr>
<tr>
<td>Noise Equivalent Bandwidth, ( \Delta f )</td>
<td>( 350 \text{ hz} )</td>
</tr>
<tr>
<td>Gain of a single dynode, ( g )</td>
<td>( 2.5 )</td>
</tr>
<tr>
<td>Background Irradiance, ( H_{B} )</td>
<td>( 1.1 \times 10^{-13} \text{ w/cm}^2 )</td>
</tr>
<tr>
<td>Minimum STAR Irradiance, ( H_{\text{eff}} )</td>
<td>( 1.1 \times 10^{-13} \text{ w/cm}^2 )</td>
</tr>
<tr>
<td>Pulse Visibility Factor, ( V_p/V_{ss} )</td>
<td>( 0.75 )</td>
</tr>
</tbody>
</table>

Normalizing irradiance levels and noise currents to the appropriate value of \( \sigma \) provides a simplified means of comparing data taken under different conditions and relating it to the analysis using baseline design parameters.

For the above design parameters, it is found that \( \sigma = 6.52 \times 10^{-15} \text{ watt cm}^{-2} \). The minimum star therefore can be expressed as

\[
H_{\text{eff}} \cdot (V_p/V_{ss}) = 12.65\sigma
\]

For a gaussian distribution of noise pulse amplitudes, the probability of a pulse exceeding the mean value by an amount \( T \) can be described by

\[
P_{-\Delta}(T) = \frac{1}{2} - \frac{1}{\alpha\sqrt{2\pi}} \int_{0}^{T} \exp \left[ -\frac{1}{2} \left( \frac{t}{\sigma} \right)^2 \right] dt
\]

The gaussian function is generally used to describe noise amplitudes greater than and less than the mean value with equal probability. For the purpose of the analysis of false alarms, only positive amplitudes need be considered, resulting in the \( \frac{1}{2} \) factor at the beginning of the above equation.

If the output noise exhibits exponential behavior the distribution of noise amplitudes greater than the mean level can be described by
If the assumption is made that the output noise amplitude is greater than the mean level half the time, then the probability of a false alarm as a function of threshold setting above the mean level can be expressed by

\[ P_{fa}(T) = \frac{1}{2} - \frac{1}{2\sigma} \int_{-\infty}^{T} \exp \left( -\frac{T}{\sigma} \right) dT \]

In the baseline STARS concept, each photomultiplier tube is only active during two 90° windows. This results in a factor of 2 reduction in the number of effective false alarms.

The probability of missing a pulse is described by the probability that the amplitude of the signal plus noise will be less than the threshold level. One must be certain to include the noise because of the finite probability that the direction of the noise pulse at the time of a signal pulse will subtract from the signal thus increasing its probability of falling below threshold. Since both positive and negative deviations of signal and noise from their mean values must be considered in this portion of the analysis, use of a gaussian distribution is most appropriate. Data obtained on the SPACS system supports use of the gaussian model.

The standard deviation or rms value of the signal + noise is given by

\[ \frac{i}{n} = \sqrt{2e \frac{A}{\delta-1} A_{\text{eff}} R (H_{\text{eff}} + H_{B}) G^2 \Delta f} \quad \text{(amp)} \]

and the corresponding signal irradiance by

\[ \sigma' = \frac{i}{A_{\text{eff}} R G} \quad \text{(watt cm}^{-2}) \]

For the baseline case in which \( H_{\text{eff}} = H_{B} \), \( \sigma' = \sqrt{2} \sigma \).

The missed pulse rate is given by

\[ \text{MPR} = P_{\text{MP}} \cdot N \]

where \( N \) is the normally expected rate of signal pulses which for a STARS sensor is four per second.
2.2 Experimental Results

A test was set up in which a light source provided a background irradiance at the PMT photocathode. The d.c. or average and the rms noise currents were measured at the output of a 300 hz (3 db) cutoff filter. A threshold was set at the average and the number of frequency of pulses exceeding this value were measured on a counter. This zero threshold \((T = 0)\) level was found to yield a false alarm rate that could be approximately described by

\[
\text{FAR} \approx 2P_{fa} \Delta f
\]

The earlier SPACS study reported a similar finding; however, certain aspects of that study appear to indicate that noise amplitudes below the mean noise level were neglected, which may imply false alarm rates a factor of two greater than those observed in these tests.

The threshold voltage level was increased and data was collected of the frequency of pulses that exceeded each threshold level. These frequencies or count rates were divided by the zero threshold rate to provide the probability of false alarm as a function of threshold and normalized to a value of .5 for the \(P_{fa}(T = 0)\).

The three tubes used in this test had the same type of photocathode and were manufactured by the same manufacturer. One tube, 7762E, has a slightly larger photocathode area than the other tubes and was manufactured around nine years ago. A second tube, 13873, was manufactured around four years ago and has been used quite extensively for SPACS engineering tests. The third tube, 19391, is of very recent manufacture and has not been used other than for acceptance type tests.

As seen in Figure 2-1, one tube (13873) behaves in an almost perfect exponential manner and the other two tubes are closer to gaussian. PMT 19391 shows a tendency toward exponential behavior for large amplitude pulses.

The nature of the measurement techniques used to obtain the above data necessitated using data taken over extended periods, many days in some cases. This caused some concern over the accuracy of applying the data to short time intervals. A solution to this shortcoming was found to be through the use of a multichannel analyzer that samples noise and counts the distribution of samples
FIGURE 2.1. FALSE ALARM DATA FOR THREE PHOTOMULTIPLIER TUBES
FIGURE 22. PHOTOMULTIPLEXER BACKGROUND CURRENT AMPLITUDE DISTRIBUTION
as a function of their amplitude.

A Northern Scientific, Model NS-900-1A multichannel analyzer was employed for this direct photomultiplier dark and background distribution analysis. In addition a Tektronix 547 oscilloscope with a 1A7A plugin was utilized to produce a +B coincidence gate allowing the analyzer to be operated in a multi-channel A/D conversion mode for analysis of the slowly varying filtered waveforms.

An examination of the data collected in this test (See Figure 2-2) confirms the previous conclusion that the distribution of large amplitude pulses differs markedly between tubes of the same family. Even after substantial filtering (300 hz bandwidth) some of the photomultipliers show an exponential form of tail distribution that is noticeably affecting the background distribution.

The exact cause of this variation in behavior among tubes is not known at this time. It is speculated that the presence of gas in the tubes may significantly affect these distributions, in particular, the large amplitude pulses.
2.3 Results and Conclusions

Figure 2-3 describes the several probability function distributions that have been discussed in this report for both the noise and signal + noise cases. Using these curves, the false alarm and missed pulse rates were computed for the exponential distribution as a worst case condition and for PMT 19391 as a more optimistic case that can probably be realized through photomultiplier tube selection. Tables 2-2 and 2-3 summarize the predicted false alarm and missed pulse rates as a function of threshold setting for the above two cases.

The signal to noise ratio of the STARS sensor as is used in the determination of its sensitivity and stability is given by

$$\text{SNR} = \frac{H_{\text{eff}} (V_p/V_{\text{miss}})}{\sigma}$$

For the baseline design parameters, a SNR of 8.95 is obtained for the minimum required star.

It must be kept in mind that the performance characteristics described are a function of the angle between the pointing direction of the star tracker and the location of bright sources, which in this analysis was $15^\circ$ for the earth and $30^\circ$ for the sun. Performance will improve at greater angles until other sources become the limiting noise factor. In the SPACS program it was determined that galactic particles probably provide this lower limit.

In the course of the experimental portion of this study, the background source was turned off and a count was made of false alarms from all other sources. For a typical threshold setting, the count rate was around two orders of magnitude lower than that obtained with a bright source background.

In summary, this study indicates that as a worst case situation, a false alarm and missed pulsed rate of 75 per hour can be anticipated. A more optimistic, yet realistic condition predicts around 4 false alarms and missed pulses per hour. It should be noted from the probability values listed in Tables 2-2 and 2-3 that the occurrence of two consecutive perturbations has very low probability. It is also very obvious that an order of magnitude improvement is obtainable if desired (at the expense of electronic complexity) if time gating of the star pulses is applied as part of the signal processing.
FIGURE 2.3. PROBABILITY OF FALSE ALARM AND MISSED PULSE AS FUNCTION OF THRESHOLD LEVEL
## TABLE 2-2

**False Alarm & Missed Pulsed Predictions. Exponential Case**

<table>
<thead>
<tr>
<th>Threshold</th>
<th>$P_{fa}$</th>
<th>FAR, hr$^{-1}$</th>
<th>$P_{MP}$</th>
<th>MPR, hr$^{-1}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>7σ</td>
<td>$4.5 \times 10^{-4}$</td>
<td>565</td>
<td>$3.5 \times 10^{-5}$</td>
<td>.505</td>
</tr>
<tr>
<td>8σ</td>
<td>$1.65 \times 10^{-4}$</td>
<td>208</td>
<td>$5.8 \times 10^{-4}$</td>
<td>8.35</td>
</tr>
<tr>
<td>9σ</td>
<td>$6 \times 10^{-5}$</td>
<td>75</td>
<td>$5 \times 10^{-3}$</td>
<td>72</td>
</tr>
<tr>
<td>9.2σ</td>
<td>$5 \times 10^{-5}$</td>
<td>63</td>
<td>$7.2 \times 10^{-3}$</td>
<td>104</td>
</tr>
<tr>
<td>9.5σ</td>
<td>$3.5 \times 10^{-5}$</td>
<td>44</td>
<td>$1.3 \times 10^{-2}$</td>
<td>187</td>
</tr>
</tbody>
</table>

## TABLE 2-3

**False Alarm & Missed Pulse Rates, PMT 19391**

<table>
<thead>
<tr>
<th>Threshold</th>
<th>$P_{fa}$</th>
<th>FAR, hr$^{-1}$</th>
<th>$P_{MP}$</th>
<th>MPR, hr$^{-1}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>6σ</td>
<td>$2.4 \times 10^{-5}$</td>
<td>30</td>
<td>$1.0 \times 10^{-6}$</td>
<td>.026</td>
</tr>
<tr>
<td>7σ</td>
<td>$6.6 \times 10^{-6}$</td>
<td>8.2</td>
<td>$3.5 \times 10^{-5}$</td>
<td>.505</td>
</tr>
<tr>
<td>7.5σ</td>
<td>$3.5 \times 10^{-6}$</td>
<td>4.4</td>
<td>$1.5 \times 10^{-4}$</td>
<td>2.16</td>
</tr>
<tr>
<td>7.7σ</td>
<td>$3 \times 10^{-6}$</td>
<td>3.8</td>
<td>$2.2 \times 10^{-4}$</td>
<td>3.2</td>
</tr>
<tr>
<td>7.8σ</td>
<td>$2.75 \times 10^{-6}$</td>
<td>3.5</td>
<td>$3.0 \times 10^{-4}$</td>
<td>4.3</td>
</tr>
<tr>
<td>8σ</td>
<td>$2.2 \times 10^{-6}$</td>
<td>2.8</td>
<td>$5.8 \times 10^{-4}$</td>
<td>8.35</td>
</tr>
</tbody>
</table>
3. Attitude Control System Design

3.1 Spacecraft Dynamics

The spacecraft dynamics used in the simulation included nonlinear
dynamics to obtain a more accurate representation of the spacecraft dynamics
for the STARS application. The high degree of accuracy desired in the
dynamics model complements the high degree of accuracy expected from STARS.
Nonlinear terms as well as product of inertias are included in the analysis
to formulate a more representative dynamics model of the spacecraft.

The inertia matrix for the spacecraft is shown below. The coordinates,
x, y, z are inertially referenced general orthogonal coordinates.

\[
I = \begin{bmatrix}
I_{xx} & -I_{xy} & -I_{xz} \\
-I_{xy} & I_{yy} & -I_{yz} \\
-I_{xz} & -I_{zy} & I_{zz}
\end{bmatrix}
\]

where

\[
I_{ij} = -\int i \cdot j \, dm
\]

\[
I_{ii} = \int (j^2 + k^2) \, dm
\]

i, j, k = coordinates, origin c.g.

\[
m = \text{mass}
\]

The angular velocity and acceleration matrices for the vehicle are

\[
\ddot{\omega} = \begin{bmatrix}
\ddot{w}_x \\
\ddot{w}_y \\
\ddot{w}_z
\end{bmatrix}, \quad \dot{\ddot{\omega}} = \begin{bmatrix}
\dddot{w}_x \\
\dddot{w}_y \\
\dddot{w}_z
\end{bmatrix}
\]
FIGURE 3-1. BLOCK DIAGRAM FOR VEHICLE DYNAMICS
where $\omega_q = \frac{d\theta}{dt}$ \(\theta = \) angle rotation about \(q\)-axis, \(q = x, y, z\)

The moment torque equation is

$$\mathbf{\tau}_{cg} = \mathbf{\tau} + \mathbf{\omega} \times \mathbf{\tau}$$

where

$$\mathbf{\tau} = I \mathbf{\omega}$$

The moment equation for the EOS vehicle can be calculated to be (See Appendix 5-1)

$$\mathbf{M}_r = I_R \mathbf{\omega}_r + I_{RY} \mathbf{\omega}_y + I_{RY} \mathbf{\omega}_p \mathbf{\omega}_r + I_Y \mathbf{\omega}_p \mathbf{\omega}_y - I_P \mathbf{\omega}_p \mathbf{\omega}_y$$

$$\mathbf{M}_p = I_P \mathbf{\omega}_p - I_{RY} \mathbf{\omega}_r^2 - I_Y \mathbf{\omega}_r \mathbf{\omega}_y + I_R \mathbf{\omega}_r \mathbf{\omega}_y + I_{RY} \mathbf{\omega}_y^2$$

$$\mathbf{M}_y = I_Y \mathbf{\omega}_y + I_{RY} \mathbf{\omega}_r \mathbf{\omega}_y + I_P \mathbf{\omega}_p \mathbf{\omega}_y - I_R \mathbf{\omega}_r \mathbf{\omega}_p - I_{RY} \mathbf{\omega}_p \mathbf{\omega}_y$$

Figure 3-1 shows the block diagram for the spacecraft dynamics based on the set of equations above. The torque exerted on the spacecraft produces roll, yaw, and pitch angles \(\phi, \psi, \text{ and } \theta\), respectively. The torque input is supplied by the control system that receives attitude error information from STARS. The spacecraft attitude angles, \(\phi, \psi, \text{ and } \theta\) are then reflected in the errors sensed by the star sensor/telescope.

3.2 Disturbance Torques

Sources of disturbance torques on an earth satellite are environmental (external) or internal. Those sources, due to interactions of the spacecraft with the environment, are gravitational gradient, solar-radiation pressure, aerodynamic pressure, meteorite impact and magnetic interaction between the earth's field and spacecraft electric currents. Internal sources are the
expulsion of matter (e.g., thrusters), activation of spacecraft elements or appendages and interaxis coupling.

Disturbance torques can be either cyclic or secular as referenced from inertial space. Momentum storage devices (e.g., reaction wheels) can be utilized to accommodate angular momentum due to cyclic disturbance without requiring unloading and exceeding minimal attitude error. Secular torques require unloading of stored momentum periodically because of continuous momentum buildup.

**SOLAR PRESSURE TORQUE**

The resultant solar pressure torque is a function of the effective area of the spacecraft exposed to the sun, the reflectivities of the exterior surfaces, and the distance between the center of mass and center of pressure.

The basic equations for solar pressure forces are:

\[
\mathbf{F}_i = -\frac{1}{2} \rho_0 \left[ \int_{A_i} (1 + \nu_i \mu_i) \left( \hat{v} \cdot \hat{n} \right) (\hat{v} \cdot \hat{n}) \hat{n} \, dA_i \right] \quad \text{(Normal Forces)}
\]

\[
+ \int_{A_i} (1 - \nu_i \mu_i) \left( \hat{v} \cdot \hat{n} \right) (\hat{n} \times \hat{v} \times \hat{n}) \, dA_i \quad \text{(Shear Forces)}
\]

\[
+ \int_{A_i} \frac{2}{3} \nu_i (1 - \mu_i) \left( \hat{v} \cdot \hat{n} \right) \hat{n} \, dA_i \quad \text{(Diffuse Reflection Forces)}
\]
where the terms are given as:

\[ \nu_i = \text{reflectance coefficient over the } i\text{th surface} \]
\[ \mu_i = \text{diffusivity coefficient over the surface } (\mu_i = 1 \text{ for specular reflection}) \]
\[ \hat{n} = \text{unit vector directed outward normal to } dA_i \]
\[ \rho_o = \text{solar radiation pressure constant for the normal incidence } (\hat{\nu}, \hat{n}) = 1 \text{ and complete absorptivity } (\nu = 0) = 9.4 \times 10^{-8} \text{ pound/foot}^2 \]
\[ A_i = \text{the area of the } i\text{th surface} \]
\[ \hat{\nu} = \text{unit vector directed to sunline} \]

The torque produced by the solar radiation pressure is

\[ \mathbf{T}_s = \sum_{i} \int_{A_i} \mathbf{t}_i \times \mathbf{F}_i \, dA_i \]

where \( t_i \) is a vector from the vehicle center of mass to the surface \( (dA_i) \) center of pressure.

Normal force effects were considered adequate for the generalized EOS spacecraft configuration used in the study. The solar panel is sun oriented and contributes heavily to the normal force, while the remaining surfaces tend to produce cancellation of shear force and/or have low effective reflectances.

The maximum solar pressure torque has been calculated to be \( 1.0 \times 10^{-5} \) foot-pound for the 9:00 A.M. orbit.

For the computer simulation, the sun line is assumed to be along the vernal equinox and the spacecraft is assumed to be symmetric about the pitch axis such that the center of pressure is located along the pitch axis. Torque about each axis is approximated by a sinusoid based on the orbit rate. The approximations used are:
$T_{Roll} = 1.0 \times 10^{-5} \cos \omega_0 t \text{ foot-pound}$

$T_{Pitch} = 0.0 \text{ foot-pound}$

$T_{Yaw} = 1.0 \times 10^{-5} \sin \omega_0 t \text{ foot-pound}$
The solar torque presented above should provide a representative disturbance torque for observing STARS performance.

AERODYNAMIC PRESSURE TORQUE

The aerodynamic pressure effect on the spacecraft at an altitude of 1000 km produces a torque on the order of $2 \times 10^{-6}$ foot-pound, primarily in the pitch direction for an angle of attack of eight (8) degrees. The general equation for aerodynamic torque is

$$T_A = P_A A_A (X_{cg} - X_{cp}) \sin \alpha$$

where

- $T_A$ = torque due to aerodynamic pressure, foot-pound
- $P_A$ = atmospheric pressure, pound/foot$^2$
- $A_A$ = surface area exposed to aerodynamic pressure, foot$^2$
- $\alpha$ = vehicle angle of attack, radians

Compared to solar pressure torque, aerodynamic pressure torque is small.

GRAVITY GRADIENT TORQUE

A spacecraft with non-equal principal moments of inertia tends to align itself in the gravity field so that its principal axis of minimum inertia lies along the gravitational vertical. The torque components due to gravity gradient are

$$T_g \text{ roll} = -\frac{3}{2} w_o \left( I_p - I_Y \right) \sin 2 \theta$$

$$T_g \text{ pitch} = -\frac{3}{2} w_o \left( I_p - I_Y \right) \sin 2 \theta$$

$$T_g \text{ yaw} = 0$$
where \( \omega_0 \equiv 10^{-3} \) radian/second orbital rate

\( I_R, I_P, I_Y \) are the vehicle moments of inertia about roll, pitch and yaw, foot-pound-second\(^2\); \( \theta \) and \( \psi \) are the roll and pitch error angles, radians.

The gravity gradient effect produces a torque about the pitch axis equal to \( 2.6 \times 10^{-8} \) foot-pound, which is small compared to the solar pressure torque.

**MAGNETIC PRESSURE TORQUE**

The magnetic field of the earth exerts magnetic pressure on the spacecraft through an interaction with electric currents on the spacecraft. The torque applied to the spacecraft is given by

\[
\mathbf{T} = 6.86 \times 10^{-6} \mathbf{M} \times \mathbf{B} \text{ foot-pound}
\]

where

\[ \mathbf{M} = \text{magnetic moment, amp-turn-foot}^2 \]

\[ \mathbf{B} = \text{magnetic flux density due to earth's magnetic field, gauss} \]

The magnitude of the earth's magnetic field can be expressed as

\[
B = \frac{1.43 \times 10^{10}}{r^3} (\sin^2 \psi + 4 \cos^2 \psi)^{1/2}
\]

where

\[ r = \text{distance from dipole center to field point, radius of orbit (nautical miles)} \]

\[ \psi = \text{angle \( r \) makes with the earth's north-south dipole centerline (radians) or \( \psi = \omega_0 t \) where \( \omega_0 \) is the orbit rate.} \]

The distance \( r \) is the sum of the earth's radius (3444 nm) and \( h_1 \) the altitude of the orbit. For an altitude of 540 nm, the magnetic flux density of the earth is \( 4.6 \times 10^{-3} \) gauss. Assuming 600 ampere-turns-foot\(^2\) for the spacecraft's magnetic moment, the peak value of magnetic pressure torque is \( 1.9 \times 10^{-5} \) foot-pound. The torque varies with position along the orbit.
3.3 **Compensation**

The vehicle control subsystem that receives attitude reference data from the STARS sensor and gimbal subsystems must provide the proper compensation for the overall STARS attitude control system in order for the vehicle's response to be stable and have the desired pointing characteristics. The compensation that provides the control law for this system has two major tasks; it must hold the vehicle pointing error due to disturbance torques to within ±0.20 arcseconds, and it must also attenuate the STARS sensor noise. The disturbance torques acting on the vehicle occur at orbit frequency, while the STARS sensor noise occurs at frequencies a couple of magnitudes higher than orbit frequency.
A simplified block diagram of the overall STARS attitude control system is shown below.

The requirement that the vehicle pointing error due to maximum disturbance torques is less than .20 arcseconds; places a magnitude constraint on the transfer function, $\theta / T_n$

$$\left| \frac{\theta (j \omega)}{T_n (j \omega)} \right| = \frac{10^{-6}}{10^{-5}} = \frac{1}{10} \ \text{radian foot-pound}$$

From the figure

$$\frac{\theta (j \omega)}{T_n (j \omega)} = \frac{G_2 (j \omega)}{1 + G_1 (j \omega) G_2 (j \omega)}$$

$$\left| \frac{G_2 (j \omega)}{1 + G_1 (j \omega) G_2 (j \omega)} \right| = \frac{1}{10}$$

$$\left| G_1 (j \omega) \right| = 10 - \left( 1/\left| G_2 (j \omega) \right| \right)$$
This expression yields a gain of 10 for \( G_1 \) at orbit frequency. To reduce pointing error due to STARS sensor noise, the gain of \( \Theta/T_n \) should be made as small as possible at the higher frequencies. With these gain constraints and other considerations in mind, the following compensation was chosen:

\[
G_1(s) = \frac{K (\tau_1 s + 1)^2}{s (\tau_2 s + 1)}
\]

\( K = .01 \) foot-pounds/radian

\( \tau_1 = 67 \) seconds

\( \tau_2 = 6.7 \) seconds

The gain \( K \) is chosen, as described above, to achieve the required pointing accuracy in the presence of disturbance torques. The double lead is required to obtain sufficient gain and phase margins for system stability. An integrator was added to alleviate a steady-state error, a result of the annual precession of the orbit plane about the center of the earth, which was encountered during simulation. The lag is used to produce a roll off at higher frequencies to filter out STARS sensor noise.

3.4 Control System Configuration

The control system torques the spacecraft to produce attitude changes. Torquing can be produced by momentum exchange between the vehicle and a momentum storage device, such as the reaction wheel. Three reaction wheels can be used to handle three attitude angles of the spacecraft.
FIGURE 32. CONTROL SYSTEM CONFIGURATION
Other means of applying torque are available, but for the sample example of torquing separately in each of the three axis and of making use of the cyclic nature of torque disturbance from the environment, the reaction wheel torquing system is a good candidate. Jet thrusters are also used, and they transfer momentum from the wheels during the unloading of the wheel momentum.

The control that produces signals to command the reaction wheel includes a shaping network or compensation to give good control characteristics to the control system. The compensation is designed to attenuate sensor noise and to enable the control system to maintain ±.20 arcsecond vehicle pointing accuracy under the effects of disturbance torques.

The simulation uses this relatively simple control law. The results show that this control law is effective in providing good response characteristics in controlling the nonlinear spacecraft dynamics. Figure 3-2 shows the control system configuration. Included in the figure is the logic box for unloading the wheels. Various torques that the vehicle experiences are also shown.

The use of a relatively simple control law was desirable to show that STARS can operate with a simple control system rather than only with a limited and/or complex type of control law. The use of a simple control law also allows for the use of more specific or precise control, if needed. A more complex control law can be designed to handle more difficult control problems if such problems should arise.
4. **Performance Analysis and Simulation**

4.1 **Precision Gimbal System**

Figure 4-1 shows the configuration of the precision gimbal pointing control system. This configuration is identical for both the pitch and polar axes.

The command to the system ($\theta_c$) is a binary number that represents the desired gimbal position which is calculated on board and updated at predetermined intervals using ephemeris data. The actual position of the gimbal is monitored by use of an Inductosyn. The output of the Inductosyn, a binary signal, is feedback and a binary subtraction is made between this signal and the command. The resulting error signal is passed first through a digital to analog converter and then through a shaping network. The output of the shaping network is a voltage that drives an Iland brush dc motor which positions the gimbal shaft. The positioning of the gimbal shaft by the motor is corrupted by bearing noise. This bearing noise which is due to mechanical tolerances and some uncertainties is described in Table 4-1. The other noise source in the gimbal system is on the Inductosyn's measurement of the actual gimbal position. This is described in Table 4-2. Both noise sources are assumed to be white with zero mean.

4.1.1 **Pitch Gimbal Simulation**

The pitch gimbal servo is used to unwind the orbital rate of the spacecraft about the earth; this must be performed in order to maintain the pitch axis of the telescope fixed in inertial space. The command angle to the pitch servo is the orbital rate of the spacecraft about the earth. This command is approximately 240 arcseconds per second and is updated every .015 seconds. Due to this relatively high angular rate and to the high load inertia of the pitch axis, a stick-slip friction model can be avoided and
PARAMETERS:

$K_D = 6.25 \times 10^{-3}$ Volts/bit

$K_I = 0.404$ bit/arcsec

$K_M = 0.376$ lb/ampere

$R = 13 \Omega$

$T = 0.01$ sec

$T_1 = 1.0$ sec

$J = 1.5$ slug$\cdot$ft$^2$ - PITCH AXIS

$0.05$ slug$\cdot$ft$^2$ POLAR AXIS

$K = 200$ VOLTS/VOLT - PITCH AXIS

$0.80$ VOLTS/VOLT - POLAR AXIS

$\theta_N$ = GAUSSIAN DISTRIBUTION, MEAN = 0, VARIANCE = 1.137 arcsec

$\theta_i_N$ = GAUSSIAN DISTRIBUTION, MEAN = 0, VARIANCE = 1.720 arcsec

**FIGURE 4-1. PRECISION GIMBAL SYSTEM**
## TABLE 4-1
**BEARING NOISE**

<table>
<thead>
<tr>
<th>Item</th>
<th>Error Class</th>
<th>Error Quantity</th>
<th>Definition</th>
<th>Assigned Value Maximum, micrometers</th>
<th>Angular Error Contribution in Assembly, microradians</th>
<th>Remarks</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>E</td>
<td>L (IRB-IRC) 1e</td>
<td>Eccentricity, inner race groove to inner race bore (TIR)</td>
<td>60</td>
<td>Not applicable</td>
<td>Each inner race marked with high spot. High spots of bearing pair aligned, thus causing residual error due to a) magnitude difference between race eccentricities and b) deviation in true high spot location and alignment.</td>
</tr>
<tr>
<td>2</td>
<td>E</td>
<td>L (IRB-IRC) 1e</td>
<td>Residual error of 1</td>
<td>15</td>
<td>15 * 1.5</td>
<td>Uncertainty in (E) L (IRB-IRC) 1e measurement.</td>
</tr>
<tr>
<td>3</td>
<td>U</td>
<td>L (IRB-IRC) 1e</td>
<td>Uncertainty of 1</td>
<td>15</td>
<td>15 * 1.5</td>
<td>Axial wobble high spot rarely coincides with eccentricity high spot so that correction cannot be made simultaneously for wobble. Assumed worst case for opposite bearing pairs diametrically opposed, however, mean was assumed within each bearing pair.</td>
</tr>
<tr>
<td>4</td>
<td>U</td>
<td>L (IRRF-IRCA) AW</td>
<td>Orthogonal component (radial direction). Axial wobble of inner race groove to inner race reference face</td>
<td>7.5</td>
<td>15 * 1.5</td>
<td>Assuming: larger balls (1/2 of the complement) diametrically opposite smaller balls. Worst possible case - both bearing pairs.</td>
</tr>
<tr>
<td>5</td>
<td>U</td>
<td>L (B-B) 1V</td>
<td>Variance, Ball-to-ball diameters (for bearing ball complement)</td>
<td>5</td>
<td>10 * 1.00</td>
<td>Assuming: some particles at times enter ball to race contact and not necessary simultaneously in both bearings and pairs in diametrically opposed locations.</td>
</tr>
<tr>
<td>6</td>
<td>U</td>
<td>L PC</td>
<td>Size of particulate contamination</td>
<td>45</td>
<td>45 * 4.5</td>
<td>Interference to exist over entire operational thermal range.</td>
</tr>
<tr>
<td>7</td>
<td>E</td>
<td>L (HI-HJ)</td>
<td>Fit between inner race bore and shaft</td>
<td>Interference</td>
<td>0</td>
<td>Interference to exist over entire operational thermal range.</td>
</tr>
<tr>
<td>8</td>
<td>E</td>
<td>L (HI-HJ)</td>
<td>Fit between outer race and housing</td>
<td>Interference</td>
<td>0</td>
<td>Combined with item 1, races are angular; positioned to magnitude-effective total eccentricity, i.e., shaft rotation axis defined by race grooves.</td>
</tr>
<tr>
<td>9</td>
<td>E</td>
<td>L(V, R1, Re, S1)</td>
<td>Shaft eccentricity between bearing locations 1 and 2 (TIR)</td>
<td>60</td>
<td>Not applicable</td>
<td></td>
</tr>
<tr>
<td>10</td>
<td>E</td>
<td>L [1-4] e, SH</td>
<td>Residual error of shaft axis definition (1 and 9 combination)</td>
<td>15</td>
<td>15 * 1.5</td>
<td>Thermal differential expansion induced change in initial interference (fit) may not be part equal, everywhere due to nonhomogeneity of materials and structures.</td>
</tr>
<tr>
<td>11</td>
<td>U</td>
<td>L [1-4] e, SH</td>
<td>Uncertainty of shaft axis definition (1 and 9 combination)</td>
<td>15</td>
<td>15 * 1.5</td>
<td>Even if both inner and outer races experience identical change in temperature (no gradient) and if both bearings experienced identical change, larger outer race will change at a higher rate than inner race. Contact angle shifts. Assume: bearing pair 1 experiences thermal shift larger or smaller than that experienced by bearing pair 2 (this causes angle uncertainty (13) due to difference in race curvatures and ball sizes between bearing pairs (1) and (2)).</td>
</tr>
<tr>
<td>12</td>
<td>U</td>
<td>L [1-4] e, S1, TH</td>
<td>Uncertainty in 1 and 9 combination due to nonhomogeneous interference fit change due to thermal changes</td>
<td>2.5</td>
<td>5 * 0.50</td>
<td>Original outer race and housing errors (as assembled) are lumped in an axial angle error - a constant. However, if changes occur, the outer angle changes in an unpredictable manner.</td>
</tr>
<tr>
<td>13</td>
<td>U</td>
<td>L [1-4] e, TH</td>
<td>Ball contact position change due to outer race to inner race thermal condition change</td>
<td>2.5</td>
<td>5 * 0.50</td>
<td>Original outer race and housing errors (as assembled) are lumped in an axial angle error - a constant. However, if changes occur, the outer angle changes in an unpredictable manner.</td>
</tr>
<tr>
<td>14</td>
<td>U</td>
<td>L (OR-HSC) e, TH</td>
<td>Housing to outer race induced eccentricity change due to nonhomogeneous interference fit change due to thermal variations</td>
<td>2.5</td>
<td>5 * 0.50</td>
<td>Original outer race and housing errors (as assembled) are lumped in an axial angle error - a constant. However, if changes occur, the outer angle changes in an unpredictable manner.</td>
</tr>
<tr>
<td>15</td>
<td>U</td>
<td>f (LE)</td>
<td>Uncertainty due to launch environment caused Brinell spots</td>
<td>30</td>
<td>10 * 3.0</td>
<td>75% Brinell spot depth of indentation in one bearing, taking total load in radial direction, 100 g push acting line x 1 second.</td>
</tr>
</tbody>
</table>

---


*Bulk temperature variation limited to \( \pm 10^\circ F \), radial gradient limited to \( 5^\circ F \) maximum.*
<table>
<thead>
<tr>
<th>Item</th>
<th>Error Quantity, microradians</th>
<th>Quantity Squared, microradians</th>
<th>Extension</th>
</tr>
</thead>
<tbody>
<tr>
<td>2</td>
<td>1.5</td>
<td>1.5</td>
<td>Σ = 5 (errors)</td>
</tr>
<tr>
<td>10</td>
<td>3.0</td>
<td>9.0</td>
<td>Rss = 3.24 microradians</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>~0.7 arcsec</td>
</tr>
<tr>
<td>3</td>
<td>1.5</td>
<td>2.25</td>
<td>Σ = 37.8 (uncertainties)</td>
</tr>
<tr>
<td>4</td>
<td>1.5</td>
<td>2.25</td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>1.00</td>
<td>1.00</td>
<td>Rms = \sqrt{37.8 / Q-1}</td>
</tr>
<tr>
<td>6</td>
<td>4.5</td>
<td>20.3</td>
<td>Σ = 2.17 microradians</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>~0.45 arcsec</td>
</tr>
<tr>
<td>11</td>
<td>1.5</td>
<td>2.25</td>
<td></td>
</tr>
<tr>
<td>12</td>
<td>0.50</td>
<td>0.25</td>
<td></td>
</tr>
<tr>
<td>13</td>
<td>0.50</td>
<td>0.25</td>
<td></td>
</tr>
<tr>
<td>14</td>
<td>0.50</td>
<td>0.25</td>
<td></td>
</tr>
<tr>
<td>15</td>
<td>3.0</td>
<td>9.00</td>
<td></td>
</tr>
</tbody>
</table>

Therefore: Total error and uncertainty contributions (maximum) = 0.70 + 0.45 = 1.15 arcsec
TABLE 4-2

INDUCTOSYN ERRORS

<table>
<thead>
<tr>
<th>A. Errors systematic with poles (512 per revolution)</th>
<th>0.45 arcseconds</th>
</tr>
</thead>
<tbody>
<tr>
<td>B. Errors systems per revolution</td>
<td></td>
</tr>
<tr>
<td>1. Centering indication on Inductosyn discs</td>
<td>0.68</td>
</tr>
<tr>
<td>2. Effect of bearing bias</td>
<td>0.50</td>
</tr>
<tr>
<td>3. Rotor to housing alignment</td>
<td>1.00</td>
</tr>
<tr>
<td>4. Stator to shaft alignment</td>
<td>1.00</td>
</tr>
<tr>
<td>5. Wobble</td>
<td>0.20</td>
</tr>
<tr>
<td>6. Bearing random effect</td>
<td>0.10</td>
</tr>
</tbody>
</table>

RSS TOTAL 1.72 arcseconds
a conventional coulomb friction model used.

A separate computer simulation of the pitch gimbal servo was performed using the MIMIC language. The computer listing of this pitch simulation can be found in Appendix 5.2. Because the command angle to the pitch gimbal is updated every .015 seconds, a very small integration step is required, thus only short simulation runs (approximately 20 seconds) were economically feasible. The pitch gimbal system response is shown in Figures 4-2, 4-3, and 4-4. In all three figures, the independent axis is scaled to seconds of time and the dependent axis is scaled to arc-seconds.

Figure 4-2 shows both the command angle (dashed line) and the actual angle (solid line) as a function of time. Both curves lie on top of one another, making the gimbal error barely visible on this scale. Figure 4-3 shows the difference between the command angle and the actual angle on a much smaller scale. The gimbal error is seen to stay almost completely within a plus or minus two arcsecond error band. The RMS value of the gimbal pointing error is shown by Figure 4-4 to be 1.2 arcseconds. This pointing error is obviously well within the 1.80 arcsecond error budget allotted to each gimbal system.

4.1.2 Polar Gimbal Simulation

The polar axis servo is used to unwind the annual precession of the orbit plane, which must be performed in order to maintain the polar axis of the telescope fixed in inertial space. The command to the polar axis servo corresponds to a one revolution or 360 degrees per year rate. This command is updated only every 90 seconds. Unlike the pitch gimbal, the slow rotation and small angle incremental motion of the polar gimbal cause
an elastic behavior between the moving surfaces of the gimbal. As a consequence, the polar gimbal simulation utilizes an elastic friction model, based on experimental data obtained on another Hughes program. Figure 4-5 shows the elastic friction characteristics observed on the Hughes OSO Program. The STARS polar gimbal operating region is shown in the lower left hand portion of the figure. For this region, the friction torque is directly proportional to the angular displacement. The elastic friction model that was used in the simulation is shown in the right half portion of the same figure. The elastic friction torque in the simulation model is equal to $K \times \Delta \theta$, when $\Delta \theta$ is not equal to zero, and zero when $\Delta \theta$ is equal to zero. Where $\Delta \theta$ is the incremental change in the command at each update interval.

A separate computer simulation of the polar gimbal servo was performed using the MIMIC language. The computer listing can be found in Appendix 5.2. Figures 4-6, 4-7, and 4-8 show the polar gimbal system response. In all three figures, the independent axis is scaled to seconds of time and the dependent axis is scaled to arcseconds.

Figure 4-6 shows both the command angle (dashed line) and the actual angle (solid line) as a function of time. The gimbal angle is seen to follow the staircase type motion of the command angle; however, its amplitude is corrupted by noise. Figure 4-7 shows the error between the commanded angle and the gimbal's actual position. Most of the error is seen to fall into a plus or minus 2 arcsecond region about zero. Figure 4-8 shows the RMS value of the pointing error to be 1.80 arcseconds, which is exactly the allotted amount of error for the polar gimbal system.
The pitch and polar gimbal simulations described above demonstrate that the precision gimbal of Figure 4-1 can meet the high accuracy requirements of STARS.
FIGURE 4-2. PITCH GIMBAL RESPONSE
FIGURE 4-3. PITCH GIMBAL ERROR
FIGURE 4.4. RMS PITCH GIMBAL ERROR
ANGULAR DISPLACEMENT, ARCSECONDS

FIGURE 4-5. ELASTIC FRICTION CHARACTERISTICS
FIGURE 4.6. POLAR GIMBAL RESPONSE
FIGURE 4-7. POLAR GIMBAL ERROR
FIGURE 4-8. RMS POLAR GIMBAL ERROR
4.2 Control System

Figure 4-9 shows the control system configuration block diagram. The command to the control system is the desired spacecraft attitude. The spacecraft pointing error, which is obtained by comparing the reference attitude to the actual spacecraft attitude measured by STARS, is passed through a shaping network that commands a torquer to torque the spacecraft in order to produce the required attitude correction. This system contains two major disturbance sources; the spacecraft disturbance torques and the sensor noise on the STARS measurement of spacecraft attitude. The shaping network is designed to attenuate the STARS sensor noise and compensate for the disturbance torques.

The control system of Figure 4-9 was simulated under the assumptions that the torquer is ideal (no dynamics) and that STARS gives a perfect measurement of the spacecraft attitude. The purpose of this simulation was to investigate the system's ability to control the spacecraft dynamics and compensate for the disturbance torques.

Figures 4-10 thru 4-12 show the control system response when no disturbances are acting on the system. Figure 4-10 shows the spacecraft attitude over one orbit to be perfect. Figures 4-11 and 4-12 show the spacecraft's body rates. From the definition of the coordinate systems (see Figure 4-13), the body rates for perfect pointing should be:

\[ w_{yaw} = \dot{\beta} \sin \gamma \sin \alpha = 0.04 \sin \dot{\alpha} \text{ arcsecond/second} \]

\[ w_{roll} = \dot{\beta} \sin \gamma \cos \alpha = 0.04 \cos \dot{\alpha} \text{ arcsecond/second} \]

\[ w_{pitch} = \dot{\beta} \cos \gamma + \alpha = 206 \text{ arcsecond/second} \]

where \( \dot{\alpha} = \text{orbit rate} \)
From examination of the three figures, it can be stated that under ideal conditions and no disturbances, the control system can point the spacecraft perfectly.

Figures 4-14 thru 4-16 show the control system response when disturbance torques are acting on the system. Figure 4-14 shows the spacecraft attitude time history for one orbit period. The large peak in the roll response is an initial transient due to the manner in which the simulation was initialized. The pointing error caused by the disturbance torque is well within the prescribed accuracy of STARS. Figures 4-15 and 4-16 show the spacecraft body rates. The body rates are seen to be unaffected by the disturbance torques.

It can be concluded from the above simulations that the control system design of Figure 4-9 can control the spacecraft dynamics and also compensate for any disturbance torques acting on the spacecraft.

![Diagram](image-url)
FIGURE 4-10. SPACECRAFT ATTITUDE, ZERO DISTURBANCE TORQUES
FIGURE 4-11. ROLL-YAW BODY RATES, ZERO DISTURBANCE TORQUES
FIGURE 4-12. PITCH BODY RATE, ZERO DISTURBANCE TORQUES
FIGURE 4-13. STARS GEOMETRY
FIGURE 4-14. SPACECRAFT ATTITUDE, DISTURBANCE TORQUES PRESENT
Figure 4-15. Roll-Yaw Body Rates, Disturbance Torques Present
FIGURE 4-16. PITCH BODY RATE, DISTURBANCE TORQUES PRESENT
4.3 Star Tracker

A block diagram of the tracker model illustrating the relationships of its component functions is shown in Figure 4-17.

The time interval (\(\Delta t\)) between a position reference pulse coming from a pickoff on the optical wedge and the pulse produced by the star image crossing the recticle slit within the telescope is stored in a register for use by the tracker system. The register is updated every 250 milliseconds. A sinusoidal relation exists between the time interval and the corresponding error angle, given the scan circle radius and its period. Thus, the star line-of-sight error angle is derived directly from the pulse time interval measurement.

The realistic tracker model includes inputs for the two major disturbances to the system; missed (or false) pulses and pulse jitter. Both photon noise from the star background and dark current noise of the photomultiplier tube will cause missed (or false) pulses. This subject was treated in Section 2 of this report. With respect to this type of disturbance, the following assumptions were made in conjunction with the simulation model:

1) Missed or false pulses will perturb the system in the same manner.

2) An exponential noise distribution is assumed, yielding the worst case condition of 72 missed and 75 false pulses per hour.

3) By the use of electronic gating, detection of missed and false pulses will be possible.

4) If a missed (or false) pulse occurs on one of the star signals, the system will ignore that signal and use the last stored value of error corresponding to this signal.

Resolution of a pulse in time (pulse jitter) and hence the angular error associated with detection of the signal depends directly on pulse (or slit) width and inversely on signal-to-noise ratio. Using the baseline star tracker parameter
FIGURE 4-17. STAR TRACKER CONFIGURATION
values given in the STARS Feasibility Study, the $1\sigma$ value of track loop angle noise (pulse jitter) is 1.15 arcseconds.

Figures 4-18 thru 4-21 show the STARS system response when the star tracker simulation model is added to the control system simulation of Section 4.2. In this case, no disturbance torques are acting on the spacecraft and the torquer is still ideal. Figure 4-18 shows the spacecraft attitude time history for half an orbit period. A peak to peak error of approximately .20 arcsecond is observed in all three axes. This is obviously well within the accuracy requirements of the STARS system. Figure 4-19 shows the star tracker's estimate of the spacecraft pointing error. If the estimate were perfect Figure 4-18 and 4-19 would be identical. However, by a comparison, the measurement is seen to be very noisy, yet the spacecraft error is very small. The important point demonstrated here is that the error in the star tracker's estimate is filtered by the large inertia of the vehicle. Figures 4-20 and 4-21 show the body rates. These rates contain a small magnitude, high frequency component in addition to the required perfect pointing rates.

In conclusion, this simulation shows that the present star tracker design will provide the required high accuracy performance.
FIGURE 4-18. SPACECRAFT ATTITUDE WITH TYPICAL STAR TRACKER NOISE BUT NO DISTURBANCE TORQUES
FIGURE 4-19. STAR TRACKER ESTIMATE OF POINTING ERROR, ZERO DISTURBANCE TORQUES
FIGURE 4-20. ROLL-YAW BODY RATES WITH TYPICAL STAR TRACKER NOISE BUT NO DISTURBANCE TORQUES
FIGURE 4.21. PITCH BODY RATE WITH TYPICAL STAR TRACKER NOISE BUT NO DISTURBANCE TORQUES
4.4 STARS System Model

A signal flow diagram of the complete STARS system, illustrating the functional interfaces between the computer, telescope assembly, and the spacecraft attitude control system is shown in Figure 4-22. The functional description of the individual components of the system have been discussed previously.

The total system was simulated on a GE635 digital computer using the MIMIC simulation language. The computer simulation flow diagram is given in Figure 4-23. The complete STARS system simulation is a combination of the simulation models of the precision gimbals, the star tracker, and the control system discussed previously, with some additional modelling added for the on-board computer requirements. All major disturbances were included. The computer listing can be found in Appendix 5.2.

Figures 4-24 through 4-27 show the complete STARS system response for one orbit period. The spacecraft attitude is shown in Figure 4-24. The response consists of an error component at orbit frequency due to the disturbance torques and a high frequency error component due to the star tracker's estimate of spacecraft attitude. The maximum peak to peak error in spacecraft pointing is observed to be well less than half of an arcsecond. The star tracker's estimate of the spacecraft attitude is shown in Figure 4-25. By comparing this figure to 4-25, the filtering effect of the large mass of the spacecraft is observed once again. Figures 4-26 and 4-27 show the spacecraft body rates. These rates contain a small magnitude, high frequency component, due to STARS sensor noise, in addition to the required perfect pointing rates. This simulation shows convincingly that the STARS system has the ability of high precision spacecraft pointing.
I, Memory, Timing, and Control

Gimbal, Angle, Command, Generator

Star Coordinates

Telescope Biases

Orbit Parameters

Star Select Logic

Sequencing Logic

Command Generator

Orbit Parameters

Master Clock

Star Select Logic

Sequencing Logic

Figure 4-22. Stars Signal Flow Diagram
FIGURE 4.23. COMPUTER FLOW DIAGRAM
FIGURE 4-24. SPACECRAFT ATTITUDE, DISTURBANCE TORQUES AND STAR TRACKER NOISE INCLUDED
FIGURE 4.25. STAR TRACKER ESTIMATE OF POINTING ERROR, DISTURBANCE TORQUES INCLUDED
FIGURE 4-26. ROLL-YAW BODY RATES, DISTURBANCE TORQUES AND STAR TRACKER NOISE INCLUDED
FIGURE 4-27. PITCH BODY RATE, DISTURBANCE TORQUES AND STAR TRACKER NOISE INCLUDED
Three additional runs were made using the complete STARS system simulation to investigate the effect of gimbal axis misalignment, the effect of a non-ideal torquer, and the initial acquisition ability of STARS.

1) Gimbal axis misalignment

The effect of an error in the built-in angle between the pitch and polar axis of the telescope assembly was considered. A three arcsecond error in this angle was assumed to be representative of this type of an error. The total STARS system simulation described above was run with this three arcsecond gimbal misalignment. The system response to this error was not visible, thus yielding a conclusion that this type of error will not affect the STARS system performance.

2) Non-ideal Torquer

A one second deadband region was added to the torquer model shown in Figure 4-9. The system response to this error is very small. Figure 4-28 shows the spacecraft attitude for this simulation run. When comparing this figure with Figure 4-24, an increase in the peak to peak magnitude of the pointing error of .1 arcsecond is observed. This type of error has a greater effect on spacecraft pointing than the gimbal misalignment error, however even with this error, the STARS control system is well within its accuracy requirements.

3) Initial Acquisition

The acquisition problem of the STARS system begins when the star's image is first seen in the telescope. The image will first appear when the error between the telescope boresight axis and the line-of-sight to the star is less than .5 degrees (1800 arcseconds).
This condition was simulated by giving the spacecraft an appropriate initial attitude so that the error in the telescope frame was approximately .5 degrees. The STARS system response for this simulation is shown in Figures 4-29 through 4-31. Figure 4-29 shows that the spacecraft pointing error is brought to zero in approximately 300 seconds. Figures 4-30 and 4-31 show the spacecraft's body rates, which are also brought to their respective perfect pointing values in approximately 300 seconds.
FIGURE 4-28. SPACECRAFT ATTITUDE USING CONTROL TORQUER WITH DEADBAND
Figure 4-29. Spacecraft attitude, initial acquisition.
FIGURE 4-30. ROLL-YAW BODY RATES, INITIAL ACQUISITION
FIGURE 4-31. PITCH BODY RATE, INITIAL ACQUISITION
4.5 Conclusions

A number of conclusions may be reached, based on the analyses, experiments, and simulations performed in this study. These cover star tracker characteristics and performance, the capabilities of the precision gimbal system, control loop design, acquisition performance, pointing precision and pointing stability.

Star tracker analyses, based on experimental data from several photomultiplier tubes, indicate that without using elaborate electronic processing, the false alarm and missed pulse rate to be anticipated lies between 75 per hour and 4 per hour. Even though these rates appear quite acceptable, an additional order of magnitude improvement is available if needed by applying time gating of the star pulses as part of the signal processing.

A detailed simulation of STARS gimbal characteristics shows that the gimbals may be positioned with a precision of better than two arcseconds, using shaft angle transducers with a resolution of 19 bits. Since the gimbal positioning precision appears to be limited chiefly by the resolution of the feedback transducer, the prospects of higher precision in this area (if desired) are very good.

Simulation of the spacecraft control system shows that a relatively uncomplicated control system design can reduce the effect of various external disturbance torques on pointing precision to negligible values while simultaneously keeping pointing jitter due to sensor noise at acceptable levels. It was found that a pointing error of 0.5 arcsecond due to disturbance torques and less than 0.2 arcsecond jitter due to sensor noise is achievable without placing significant demands on system gains or bandwidth. At the same time the transient response of the overall system appears quite adequate, with acquisition from the edge of the field of view occurring in less than 300 seconds.

The overall conclusion that may be drawn from the results of the STARS
System Application Study is that a relatively straightforward precision pointing system design utilizing STARS on an EOS type spacecraft, and considering the characteristics of real sensing and control hardware, can achieve the desired 0.001 degree earth pointing capability. The STARS approach should therefore be a strong contender in any application where a large payload must be pointed earthward with high precision from any arbitrary orbit.
5. APPENDIX
5.1 \textbf{Spacecraft Equations of Motion}

The spacecraft angular momentum is

\[ \overrightarrow{H} = I \overrightarrow{w} \]

or, in expanded form

\[
\begin{bmatrix}
\dot{H}_R \\
\dot{H}_P \\
\dot{H}_Y \\
\end{bmatrix}
= 
\begin{bmatrix}
I_R & I_{RP} & I_{RY} \\
I_{PR} & I_P & I_{PY} \\
I_{YR} & I_{YP} & I_Y \\
\end{bmatrix}
\begin{bmatrix}
w_R \\
w_P \\
w_Y \\
\end{bmatrix}
\]

For the EOS configuration specified by GSFC, the only non-zero crossproduct of inertia is \( I_{RY} (= I_{YR}) \). Therefore

\[
\begin{bmatrix}
\dot{H}_R \\
\dot{H}_P \\
\dot{H}_Y \\
\end{bmatrix}
= 
\begin{bmatrix}
I_R \dot{w}_R + I_{RY} \dot{w}_Y \\
I_{PR} \dot{w}_P \\
I_{YR} \dot{w}_Y + I_{RY} \dot{w}_R \\
\end{bmatrix}
\]

and

\[
\begin{bmatrix}
\ddot{H}_R \\
\ddot{H}_P \\
\ddot{H}_Y \\
\end{bmatrix}
= 
\begin{bmatrix}
I_R \ddot{w}_R + I_{RY} \ddot{w}_Y \\
I_{PR} \ddot{w}_P \\
I_{YR} \ddot{w}_Y + I_{RY} \ddot{w}_R \\
\end{bmatrix}
\]

The moment equation is

\[ \overrightarrow{M} = \overrightarrow{H} + \overrightarrow{\omega} \times \overrightarrow{H} \]

now,

\[
\overrightarrow{\omega} \times \overrightarrow{H} = 
\begin{bmatrix}
1 & j & k \\
w_R & w_P & w_Y \\
H_R & H_P & H_Y \\
\end{bmatrix}
\begin{bmatrix}
H_R w_P - H_P w_Y \\
H_Y w_R + H_R w_Y \\
H_P w_R - H_R w_P \\
\end{bmatrix}
\]
Thus the expanded moment equations are

\[ M_R = I_R \dot{\omega}_R + I_{RY} \dot{\omega}_Y + I_{RY} \omega_R \omega_Y + I_Y \omega_p \omega_Y - I_P \omega_p \omega_Y \]

\[ M_P = I_P \dot{\omega}_p - I_{RY} \omega_R^2 - I_Y \omega_R \omega_Y + I_R \omega_R \omega_Y + I_{RY} \omega_Y^2 \]

\[ M_Y = I_Y \dot{\omega}_Y + I_{RY} \dot{\omega}_R + I_P \omega_p \omega_R - I_R \omega_R \omega_p - I_{RY} \omega_p \omega_Y \]
5.2 COMPUTER PROGRAM LISTINGS
$ IDENT 366*54790*10870*78294*ROY
$ LIMITS 30*30K*10K*1
$ PROGRAM RLHS*NDUMP
$ LIMITS 30*30000*10000
$ PERM H*MIMIC-SYSTEM
$ TAPE 09*A25

CON(K1*KM1*KII*KD1)
CON(TAU1+TAU11*R1*J1)
CON(THETD0+THETAO)
DBM(TRUE)

DTMIN 1/(2**13)
DTMAX DTMIN
DT = 0!
F(T>20)

S=1 FS(T*FALSE*TRUE*FALSE)
S=1 KRDA 3600*180/3*14159
S=1 KS2 (KD1*KI1*KM1)/R1
S=1 OMEG (13.737*360)/24
S=3 FSW(REM(T,0.0015)*FALSE*TRUE*FALSE)
S=3 ALPHA OMEG*T
S=3 THECOM ALPHA
S=3 THETAC ING(KI1*THECOM)
INDTER RNG(O1.75*1)
BEARER RNG(O1.137*2)
EII0 THETAC-THETB
EII K52*EII0
P21 INT(EII1*0)
P11 INT((10*TAU1+TAU11)*EII1-E01+P21*G)
EO1 (10*TAU1*TAU11*EII1+P11)/TAU1
O11 (EO1+F11)/J1
F1 FS(THETAD+.026+.0-.026)
THETAD INT(G11*THETD0)
THETA INT(THETD0+THETAO)
THETAA KRDA*THETA+BEARER
THETB ING(KI1*(THETAA+INDTER))
THEDEL THECOM-THETAA
R (1/T)*INT(THEDEL*0)
RMS SQR((1/T)*INT(THEDEL*THEDEL*0)-R^R)
PLOT(T*RMS)
SCL(20*0.5*-5)

END

200 .376 .404 6.25E-3
404 1
.01 13 1.5
0 0
$ ENDJOB

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PITCH GIMBAL SIMULATION

83
$ IDENT 366*54790*10870*78294*ROY
$ LIMITS 10*30K**10K**1
$ PROGRAM RLHS*NDUMP
$ LIMITS 10*30000**10000
$ PERM H*IMIC-SYSTEM
$ TAPE 09*A2S

CON(KF2)
CON(PHIDO**PHI0)
PAR(K2*KM2*K12*KD2)
PAR(TAU2*TAU12*R2*J2)
PAR(CCUNT**SUM)
DBM(TRUE)

DTMIN 1/(2**5)
DTMAX DTMIN
DT 1
FIN(T*1000)
F1Sw(T**FALSE**TRUE**FALSE)

Sw 1
K3 (KD2*K2*KM2)/R2

Sw 3
FSW(REM(T,90)**FALSE**TRUE**FALSE)

Sw 1
KRDA (180*3600/3*14159

Sw 1
OMEGA (360/((365*26*24)

Sw 3
BETA 0;EGA*T

Sw 3
PHICOM BETA
BEARER RNG(0*1.137*5)
INDTER RNG(0*1.6175*7)

******** ELASTIC STICKTION MODEL

Sw 3
CO PHI
Sw 4
FSW(ABS(PHIDO)-.000001**FALSE**TRUE**FALSE)
Sw 5
COM(SW4)

Sw 3
FSU(F2)

Sw 3
C1 PHI-C0
Sw 3
F2 -KF2*C1

********
EI20 PHIC-PHB
EI2 KS*EI20
P22 INT(EI2*0)
P12 INT((10*TAU2*TAU12*EI2-E02+P22*0)
E02 (10*TAU2*TAU12*EI2+P12)/TAU2
O12 (E02+F2)/J2
PHID INT(C12**PHIDO)
PHI INT(PHIDO**PHI0)
PHA KRDA*PHI**BEARER
PHB INT(K12*(PHI**INDTER))
PHIDEL PHICOM**PHA
R1 (1/T)*INT(PHIDEL*0)
RMS1 SGF((1/T)*INT(PHIDEL**PHIDEL*0)-R1**R1)
PRINT FSW(REM(T,1)**FALSE**TRUE**FALSE)
PRINT OUT(T*PHA**PHICOM**PHIDEL**RMS1)
PRINT PLOT(T*PHA**PHICOM)
PLOT(T*PHIDEL)
PLOT(T*RMS1)
PLOT(T*RMS1)
END

182.63 0 0 POLAR GIMBAL SIMULATION
| 1.5 | 0.376 | 0.404 | 6.25E-3 |
| 0.01 | 1 | 13 | 0.05 |
| 1 | 0 |  |

$ $ ENDJOB
STARS SYSTEM MODEL

IDENT 366*54730+10870*78294*ROY
LIMITS 65*30K**10K**1
PROGRAM RLM5MDOUPK
LIMITS 65*30000**10000
PERM M=5#NUTIC-SYSTEM
TAPE 09#A35

A STELLAR TRACKING ATTITUDE REFERENCE SYSTEM ---- STARS ----

BEGIN DATA INPUT

ORBIT CONSTANTS
RC DISTANCE FROM VEHICLE TO EARTH MILES
VO MAGNITUDE OF EARTHS VELOCITY ABOUT SUN MILES/SEC
(10 NEGLECT STELLAR ABERRATION SET RC=VO=0)
GAMMA ORBIT INCLINATION ANGLE DEGREES

SIDERAL CONSTANTS
K1 PITCH DRIVER GAIN VOLT/VOLT
K2 PITCH VOLT GAIN FT-LB/AMS
K3 PITCH INDUCED GAIN BIT/ARCSEC
K4 PITCH V/A CONVERSION GAIN VOLT/BIT
K5 VERTICAL FRICTION PITCH AX, SECONDS
K6 PITCH TIME Constants 1
K7 PITCH TIME Constants 2
K8 PITCH WINDING RESISTANCE OHM
K9 PITCH INERTIA SLUG-FT#FT

ANGULAR CONSTANTS
K10 POLAR DRIVER GAIN VOLT/VOLT
K11 POLAR VOLT GAIN FT-LB/AMS
K12 POLAR INDUCED GAIN BIT/ARCSEC
K13 POLAR V/A CONVERSION VOLT/BIT
K14 ELASTIC FRICTION COEFF (POLAR AXIS) FT-LB/ARCSEC
K15 PERS TIME Constants 1
K16 PERS TIME Constants 2
K17 PERS WINDING RESISTANCE OHM
K18 PERS INERTIA SLUG-FT#FT

GIA ERROR COMPENSATION CONSTANTS
WGTM PITCH GLOBAL GIA ERROR DEGREES
WGPOL POLAR GLOBAL GIA ERROR DEGREES
WGLCAI GIA INCLINATION GLOBAL GIA ERROR DEGREES
DAJ ROLL TELESCOPICIAS FOR J-STAR DEGREES
DXK ROLL TELESCOPICIAS FOR K-STAR DEGREES
SBY YAW TELESCOPICIAS FOR J-STAR DEGREES
SYK YAW TELESCOPICIAS FOR K-STAR DEGREES
BZK PITCH TELESCOPICIAS FOR J-STAR DEGREES
BZK PITCH TELESCOPICIAS FOR K-STAR DEGREES

CON (DELTHI+ELPM+GELCAI)
CON (DXK+SBY+SZK+SYK+BZK)

TIME TO ANGLE CONVERSION CONSTANTS
KX SCAN CIRCLE RADIUS DEGREES
P SCANNING PERIOD SEC
MPH RW INDEXED FALSE PULSE RATE PER HOUR

CONTROL SYSTEM CONSTANTS
KX CONTROL SYSTEM GAIN -- YAW FT-LB/RAD
KZM CONTROL SYSTEM GAIN -- ROLL FT-LB/RAD
KZM CONTROL SYSTEM GAIN -- PITCH FT-LB/RAD

86 REPRODUCIBILITY OF THE
ORIGINAL PAGE IS POOR
TAU1Y LEAD TIME CONSTANT -- YAW SECONDS
TAU1R LEAD TIME CONSTANT -- ROLL SECONDS
TAU1P LEAD TIME CONSTANT -- PITCH SECONDS
TAU2Y LAG TIME CONSTANT -- YAW SECONDS
TAU2R LAG TIME CONSTANT -- ROLL SECONDS
TAU2P LAG TIME CONSTANT -- PITCH SECONDS
CON(K2Y*TAU1Y+TAU2Y+K2R*TAU1R+TAU2R)
CON(K2P*TAU1P+TAU2P)

SPACECRAFT DYNAMICS CONSTANTS
IR SPACECRAFT ROLL INERTIA SLUG-FT#FT
IP SPACECRAFT PITCH INERTIA SLUG-FT#FT
IY SPACECRAFT YAW INERTIA SLUG-FT#FT
IRY SPACECRAFT ROLL-YAW CROSS INERTIA SLUG-FT#FT
CON(IR*IP+IY*IRY)

INITIAL CONDITIONS
ALPHA0 ALPHA IC DEGREES
BETA0 BETA IC DEGREES
THETA0 PITCH GIMBAL INITIAL POSITION ARCSEC
THETAD0 PITCH GIMBAL INITIAL RATE ARCSEC/SEC
PHI0 POLAR GIMBAL INITIAL POSITION ARCSEC
PHI10 POLAR GIMBAL INITIAL RATE ARCSEC/SEC
CON(ALPHA0+BETA0)
CON(THETAD0+THETA0+PHI0+PHI10)

YAWDIC YAWDOT IC ARCSEC/SEC
ROLDIC ROLDOT IC ARCSEC/SEC
PITDIC PITDOT IC ARCSEC/SEC
YAW0 YAW IC ARCSEC
ROL0 ROL IC ARCSEC
PIT0 PIT IC ARCSEC

PAR(YAWDIC+ROLDIC+PITDIC+YAW0+ROL0+PIT0)
T0 SIMULATION INITIAL TIME SECONDS
TFIN SIMULATION FINISH TIME SECONDS
N STEP SIZE = 1/12**N SECONDS
PRINT PRINTING INTERVAL SECONDS
PAR(T0+TFIN+N+PRINT)

STAR SELECTION
DELTAJ DECLINATION OF J-STAR DEGREES
DELTAK DECLINATION OF K-STAR DEGREES
LAMDAJ RIGHT ASCENSION OF J-STAR DEGREES
LAMDAK RIGHT ASCENSION OF K-STAR DEGREES
PAR(DELTAJ+LAMDAJ+DELTAK+LAMDAK)

ETAK RIGHT ASCENSION OF K-TELESCOPE BORESIGHT DEGREES
ZETAK DECLINATION OF K-TELESCOPE BORESIGHT DEGREES
ETAJ RIGHT ASCENSION OF J-TELESCOPE BORESIGHT DEGREES
ZETAJ DECLINATION OF J-TELESCOPE BORESIGHT DEGREES
PAR(ZETAJ+ETAJ+ZETAK+ETAK)

PRELIMINARY CALCULATIONS
DBM(TRUE)

Sw1 FS(T*FALSE+TRUE*FALSE)
Sw1 PIE 3.14159
Sw1 RTD 180/PIE
Sw1 C 186000
Sw1 KRDA 3600*RTD
Sw1 KA KR*KRDA
Sw1 KP (KD1*K1*KM1)/R1
Sw1 KS (KD2*K2*KV2)/R2
Sw1 OMEG (13.737*360)/(24*3600)
Sw1 OMEGA 360/(365*26*24*3600)
ALPHA = 34EG*3600
BETA = OMEG*3600
ALPHAB = OMEGA/3
SINOME = SIN(OMEGA*TI/RTD)
COSME = COS(OMEGA*TI/RTD)
COS = COS(360*RTD)
PSI = GAMMA
CPSI = COS(PSI/RTD)
SPSI = SIN(PSI/RTD)
CANGAM = COS(GAMMA/RTD)
SGAMMA = SIN(GAMMA/RTD)
CMDTJ = COS(DeltaJ/RTD)
SDeltaJ = SIN(DeltaJ/RTD)
CMDTK = COS(DeltaK/RTD)
SDELTK = SIN(DeltaK/RTD)
CLANDJ = COS(LANDJ/RTD)
SLANDJ = SIN(LANDJ/RTD)
CLANDK = COS(LANDK/RTD)
SLANDK = SIN(LANDK/RTD)
CZETAK = COS(ZETAK/RTD)
SZETAK = SIN(ZETAK/RTD)
CZETAJ = COS(ZETAJ/RTD)
SZETAJ = SIN(ZETAJ/RTD)
CI1 = IY/KRDA
CI2 = IR/KRDA
I3 = IP/KRDA
I2 = IRY/KRDA

RUN CONTROL
DIMIN = 1/(2**N)
DTHAX = DIN IN
DT = 1
TI = TD + FIN(TI + TFIN)

MEMORY, TIMING, AND CONTROL
FRG = (REX(TI > 25) * FALSE) * TRUE * FALSE
COSALPHA = COS(ALPHA/RTD)
SINALPHA = SIN(ALPHA/RTD)
ALPHA = OMEGA*TI + ALPHAO
BETA = OMEGA*TI + BETAO
COSBETA = COS(BETA/RTD)
SINBETA = SIN(BETA/RTD)

GIMBAL ANGLE COMMAND GENERATOR
THECON = (ALPHA + DELTHETA)*3600
THETA = SIN(THETA/RTD)
COS = COS(THETA/RTD)
PHI = SIN(DELPHI/RTD)
COSPHI = COS(DELPHI/RTD)

PITCH GIMBAL EQUATIONS
INDTEP = RNG(0, 15, 75, 1)
BEAREP = RNG(0, 137, 2)
THETAC = ING(K11, THECON)
E110 = THETAC - THETAO
E11 \* KP\*E110
P21 \* INT(E11+0)
P11 \* INT((10\*TAU1+TAU11)*E11-E01+P21+0)
E01 \* (10\*TAU1*TAU11*E11+P11)/TAU1
Q11 \* (E01+F1)/J1
F1 \* FSW\*THETAD+KF1+0\*KFSW
THETAD \* INT(O11+THETDO)
THETA \* INT(THETA+THETAD)
THEATA \* KRDA\*THETA+BEAREP
THETB \* ING(K11*THETATA+INTEP)

** POLAR GIMBAL EQUATIONS **

** ELASTIC FRICITION MODEL **

<table>
<thead>
<tr>
<th>Sw3</th>
<th>C0</th>
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<tr>
<td>Sw5</td>
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<td>C1</td>
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<tr>
<td>Sw6</td>
<td>F2</td>
<td>-KF2*C1</td>
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** OPTICAL STAR SENSORS **

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<tr>
<th>Sw2</th>
<th>A612</th>
<th>CTHETA-CHI-STHETA<em>CPSI</em>SPHI</th>
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<tr>
<td>Sw2</td>
<td>A613</td>
<td>SPHI*SPS</td>
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<tr>
<td>Sw2</td>
<td>A621</td>
<td>CTHETA<em>SPHI-CTHETAITAPEX</em>CPSI</td>
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<tr>
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<td>CTHETAITAPEX<em>CTHETA</em>CPSI*SPHI</td>
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<tr>
<td>Sw2</td>
<td>A623</td>
<td>-SPSI*CPHI</td>
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<tr>
<td>Sw2</td>
<td>A631</td>
<td>STHETA*CPSI</td>
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<tr>
<td>Sw2</td>
<td>A632</td>
<td>CTHETA*CPSI</td>
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<tr>
<td>Sw2</td>
<td>A633</td>
<td>CPSI</td>
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** INTERVAL TIMING **

| Sw2 | TXJ  | -P*ASN(RMHXJ/KA)/(2*PIE) |
MUXX RHUX+(X1XK+NUXX+AXK)*3600
MUYY RHYY+(X1YJ+NUYY+AYJ)*3600
MUYK RHYY+(X1YJ+NUYK+AYK)*3600

COORDINATE TRANSFORMATION AND ERROR ESTIMATION

R11 CALPHA#CBETA-SALPHA#S#E#A#CGAMMA
R12 -SALPHA#CBETA-CALPHA#S#BETA#CGAMMA
R13 S#BETA#CGAMMA
R21 CALPHA#CBETA+SALPHA#C#BETA#CGAMMA
R22 -SALPHA#S#BETA+CALPHA#C#BETA#CGAMMA
R23 -SGAMMA#CBETA
R31 SALPHA#SGAMMA
R32 CALPHA#SGAMMA
R33 CGAMMA
R11J SDELTJ#CLAMDJ#R11+SDELTJ#SLAMDJ#R21-CDELTJ#R31
R12J SDELTJ#CLAMDJ#R12+SDELTJ#SLAMDJ#R22-CDELTJ#R32
R13J SDELTJ#CLAMDJ#R13+SDELTJ#SLAMDJ#R23-CDELTJ#R33
R21J -SLAMDJ#R11+CLAMDJ#R21
R22J -SLAMDJ#R12+CLAMDJ#R22
R23J -SLAMDJ#R13+CLAMDJ#R23
R11K SDELTJ#CLAMDJ#R11+SDELTJ#SLAMDJ#R21-CDELTJ#R31
R12K SDELTJ#CLAMDJ#R12+SDELTJ#SLAMDJ#R22-CDELTJ#R32
R13K SDELTJ#CLAMDJ#R13+SDELTJ#SLAMDJ#R23-CDELTJ#R33
R21K -SLAMDJ#R11+CLAMDJ#R21
R22K -SLAMDJ#R12+CLAMDJ#R22
R23K -SLAMDJ#R13+CLAMDJ#R23
A21 A12
A22 R12J*R12J+R22J*R22J+R12K&R12K+R22K&R22K
A31 A13
A32 A23
A33 R13J*R13J+R23J*R23J+R13K&R13K+R23K&R23K
DETA A11*(A22*A33-A32*A32)+A12*(A23*A31-A21*A31)
DETA DETAI+A13*(A21*A32-A22*A31)
A11IN (A22*A33-A23*A32)/DETA
A12IN (A13*A32-A12*A33)/DETA
A13IN (A13*A32-A12*A33)/DETA
A21IN (A23*A31-A21*A33)/DETA
A22IN (A11*A33-A13*A31)/DETA
A23IN (A13*A21-A11*A23)/DETA
A31IN (A21*A32-A22*A31)/DETA
A32IN (A12*A31-A13*A32)/DETA
A33IN (A11*A22-A12*A21)/DETA
A511 A11IN*R11J+A12IN*R12J+A13IN*R13J
A512 A11IN*R11J+A12IN*R12J+A13IN*R13J
A513 A11IN*R11J+A12IN*R12J+A13IN*R13J
A514 A11IN*R12K+A12IN*R12K+A13IN*R13K
A521 A21IN*R11J+A22IN*R12J+A23IN*R13J
A522 A21IN*R12J+A22IN*R12J+A23IN*R13J
A523 A21IN*R11K+A22IN*R12K+A23IN*R13K
A524 A21IN*R12J+A22IN*R12J+A23IN*R13J
A531 A31IN*R11J+A32IN*R12J+A33IN*R13J
A532 A31IN*R13K+A32IN*R12J+A33IN*R13J
A533 A31IN*R11K+A32IN*R12K+A33IN*R13K
A534 A31IN*R21K+A32IN*R22K+A33IN*R23K
EAX1 A511#MUXX+A512#MUYY+A513#MUXX+A514#MUYK
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END JOB