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Optimal Attitude Maneuver Execution for the Advanced Composition Explorer (ACE) Mission

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ABSTRACT

The Advanced Composition Explorer (ACE) spacecraft will require frequent attitude reorientations in order to maintain the spacecraft high gain antenna (HGA) within 3 degrees of earth-pointing. These attitude maneuvers will be accomplished by employing a series of ground-commanded thruster pulses, computed by ground operations personnel, to achieve the desired change in the spacecraft angular momentum vector. With each maneuver, attitude nutation will be excited. Large nutation angles are undesirable from a science standpoint. It is important that the thruster firings be phased properly in order to minimize the nutation angle at the end of the maneuver so that science collection time is maximized.

The analysis presented derives a simple approximation for the nutation contribution resulting from a series of short thruster burns. Analytic equations are derived which give the induced nutation angle as a function of the number of small thruster burns used to execute the attitude maneuver and the phasing of the burns. The results show that by properly subdividing the attitude burns, the induced nutation can be kept low. The analytic equations are also verified through attitude dynamics simulation and simulation results are presented. Finally, techniques for quantifying the post-maneuver nutation are discussed.

I. INTRODUCTION

The Advanced Composition Explorer (ACE) spacecraft will be launched in August 1997. The spacecraft will be placed into a spin-stabilized attitude. The spacecraft will carry a pair of Adcole two-axis digital Sun sensors and a Ball Aerospace CT-631 series charged-coupled device (CCD) star tracker. Telemetry data from these sensors will be downlinked to allow spacecraft attitude determination at the NASA Goddard Space Flight Center (GSFC). Both spin rate and spin axis attitude will be open-loop controlled by ground commanded hydrazine thruster firings.

Following launch, GSFC personnel will design and execute a series of trajectory maneuvers to transfer ACE from a low earth orbit to a Lissajous orbit about the Sun-earth L1 libration point. The following constraints are levied on the spacecraft attitude by the mission design:

1) The spin axis (the spacecraft +Z axis) must be maintained within 20° of the spacecraft-Sun line for power, thermal, and science instrument safety reasons

2) The spacecraft high-gain antenna boresight, which is along the spacecraft -Z axis, must be maintained with 3° of nadir to allow sufficient link margin for radio frequency (RF) communications with the Deep Space Network (DSN) ground stations

3) The spacecraft spin rate must be maintained to 5.0 ± 0.1 RPM.

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The ACE attitude and orbit control system (AOCS) consists of a set of 10 1-lb_f hydrazine thrusters placed around the spacecraft structure. The thrusters are mounted in pairs; 4 of the thrusters are oriented to provide axial forces and 6 provide radial forces. The thrusters can be commanded to fire either individually or in groups. The ACE thruster layout is shown in Figure 1. Groups I and II are the upper deck (+Z) thrusters; groups III and IV are the lower deck (-Z) thrusters. Axial thrusters are denoted as 'A', while radial thrusters are denoted as 'R'.

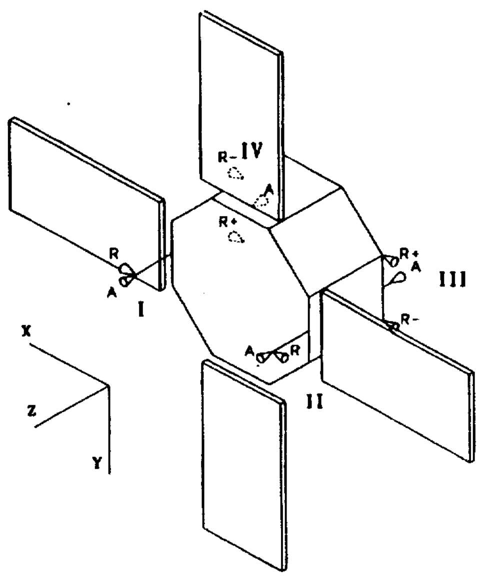

Figure 1. ACE Thruster Locations

Once the Lissajous orbit has been achieved, orbit stationkeeping maneuvers will be performed about once every eight weeks. More frequently, attitude maneuvers will need to be performed to maintain the HGA within 3° of nadir. Since the L1 point moves relative to inertial space at the sidereal rate (0.9829° per *solar* Earth day), attitude reorientations will need to be performed at least every 6.1 days to maintain the HGA within the ±3 degree deadband.

When spacecraft maneuvers are performed, a certain amount of nutation will be induced. ACE carries a passive onboard nutation damper to dissipate the excess spacecraft rotational kinetic energy introduced during a maneuver and thereby decrease nutation over time as an exponentially decaying function. The 1/e time constant for the ACE nutation damper is 9.5 hours, so nutation will be damped very slowly.

Nutation is undesirable for ACE since it can cause errors in science data collection. The ACE Command and Data Handling (C&DH) subsystem uses the Sun sensor to measure the period of the last spin. It then divides the spin period into 16384 ticks, with an equal time allotted per tick, and places science instrument data into resulting sectors. In the presence of nutation, the observed spin period, P, will vary as

$$\Delta P \cong \frac{2\Theta}{\Omega_s \cdot \tan\beta} \tag{1}$$

where

Θ = nutation angle
Ω_s = z-axis angular velocity (spin rate)
β = angle between the Z-axis and the spacecraft-Sun line

Any variation in the spin period greater than approximately 7 msec will cause science data to be placed into the wrong sector and will necessitate reprocessing of science data on the ground. To avoid reprocessing of data, the attitude should be controlled to minimize ΔP. This can be accomplished in one of three ways. The first two ways would be to increase spin rate or Sun angle (Ω_s or β); however, these parameters are constrained by the mission constraints listed above. The more effective way is to minimize the nutation angle, Θ, during each maneuver. By keeping the induced nutation angle small throughout the duration of the maneuver, the impact to science data collection will be minimized. The analysis below investigates how this may be accomplished during the ACE attitude maneuvers and identifies methods to use in planning and executing the maneuvers.

II. ANALYSIS

The goal of this analysis is to explore strategies for performing spacecraft attitude maneuvers while keeping the induced spin-axis nutation angle, Θ, small. First, the total thruster on time to achieve a 6 degree maneuver is computed. For properly centered burn arcs of finite length, the angular momentum change in the desired maneuver direction is given as:

$$\delta H = \frac{2F \cdot R_m \cdot \sin\alpha}{\Omega_s} \tag{2}$$

where

F = thruster force, 4.448 N
R_m = thruster moment arm
α = burn arc half-angle
Ω_s = nominal spin rate, π/6 radian/sec

For each thruster, the moment arm normal to the spin axis was computed from information provided by the spacecraft manufacturer. The results are given in Table 1. Note that the moment arm varies significantly, based on thruster location; the moment arm for an upper deck axial thruster is nearly twice that of a lower deck radial thruster.

Table 1. ACE Thruster Moment Arms

Thruster Location	Thruster Orientation	Thruster Moment Arm
Upper Deck	Axial	0.78 m
	Radial	0.60 m
Lower Deck	Axial	0.69 m
	Radial	0.40 m

An attitude maneuver could be performed by firing 1 lower deck radial thrusters several times until the desired attitude reorientation is achieved. Assuming no initial nutation, the spin axis and the angular momentum vector, **H**, are coincident prior to the maneuver. When the thruster is fired, the change in angular momentum, δH, is directed perpendicular to the spin axis, and thus **H**. The angular change in the spin axis direction, Γ, is given as:

$$\Gamma \cong \frac{\delta H}{H} = \frac{\delta H}{I_3 \cdot \Omega_s},$$

or,

$$\Gamma = \frac{2 \cdot F \cdot R_m \cdot \sin\alpha}{I_3 \cdot \Omega_s^{\,2}} \tag{3}$$

where

I_3 = the principal moment of inertia

(The calculated values of I_3 are 373.44 kg-m² at beginning-of-life (BOL) and 334.19 kg-m² at end-of-life (EOL). An average value of 353.82 kg-m² will be used in this analysis.)

For 1 thruster burning for 1 second, α = 15°, and

$$\Gamma = \frac{2\,(4.448\text{ N})\,(0.40\text{ m})\,(\sin 15°)}{(353.82\text{ kg-m}^2)\,(\pi/6\text{ rad/sec})^2} = 0.0095\text{ rad}$$

The total thruster on time required to achieve the desired 6° attitude maneuver is given by

$$t = \frac{(6\text{ deg})\,(\pi/180)}{0.0095\text{ rad}} = 11.02\text{ seconds}$$

However, if the 1 second pulses were broken down into shorter pulses, the maneuver could be executed more efficiently. The following analysis also shows that shorter pulses will induce less nutation.

A simple approximation for the nutation contribution resulting from a series of short burns is derived by making the following assumptions:

- the spacecraft is a rigid body
- the spacecraft is axially symmetric ($I_1 = I_2 = I_T$)
- the nutation angle, Θ, remains small, so $\cos\Theta \approx 1$
- small burns are treated impulsively (the width of the burn arc is neglected)
- spacecraft torques are small; torque-free motion is assumed
- there is no initial nutation prior to the maneuver.

The basic attitude dynamics of a nutating body states that the spacecraft principal Z axis, **Z**, revolves around the angular momentum vector, **H**, at the inertial nutation rate, $\Omega_I = I_3/I_T \cdot \Omega_s$. Then, **Z** revolves $2\pi \cdot I_3/I_T$ radians about **H** during the course of one spin period. The angle ϕ can be defined as the angular rotation of the spacecraft between thruster burns and is represented as:

$\phi = \pi^* I_3/I_T$ for 1 small burn per ½ revolution,
$\phi = 2\pi^* I_3/I_T$ for 1 small burn per 1 revolution,
$\phi = 3\pi^* I_3/I_T$ for 1 small burn per 1½ revolution,
etc.

The ACE attitude maneuvers will be modeled as a number of small burns that are performed after each successive spacecraft rotation by the angle ϕ. By assuming small nutation angles, the offset between **Z** and **H** for each small burn may be added vectorially to get the approximate position of **Z** relative to **H** at the end of the maneuver. The components of **Z** in the plane perpendicular to **H** may be expressed conveniently as the real and imaginary parts of a complex sum as follows:

$$\delta\Theta = \frac{\Gamma}{n} \cdot \left[1 + e^{j\phi} + e^{j2\phi} + \ldots + e^{j(n-2)\phi} + e^{j(n-1)\phi}\right] \quad (4)$$

where

$\delta\Theta$ = final position of **Z** relative to **H**
Γ = total attitude motion required in the attitude maneuver
n = number of small burns used to perform the maneuver

By multiplying each term in equation 4 by the quantity $e^{i\phi}$, we get

$$e^{i\phi} \cdot \delta\Theta = \frac{\Gamma}{n} \cdot \left[e^{j\phi} + e^{j2\phi} + \ldots + e^{j(n-1)\phi} + e^{j(n)\phi}\right] \quad (5)$$

Equations 4 and 5 can then be differenced to yield

$$\delta\Theta = \frac{\Gamma}{n} \cdot \frac{\left(1 - e^{in\phi}\right)}{\left(1 - e^{i\phi}\right)} \quad (6)$$

The nutation angle induced by the maneuver is just the magnitude of this vector in the complex plane. This magnitude can be derived from equation 6 by using the equality

$$e^{in\phi} = \cos(n\phi) + i \cdot \sin(n\phi) \quad (7)$$

Thus,

$$|\delta\Theta| = \frac{\Gamma}{n} \cdot \frac{\left|\left[1 - \cos(n\phi)\right] - i\left[\sin(n\phi)\right]\right|}{\left|\left(1 - \cos\phi\right) - i\left(\sin\phi\right)\right|}$$

This equation reduces to

$$|\delta\Theta| = \frac{\Gamma}{n} \cdot \sqrt{\frac{1 - \cos(n\phi)}{1 - \cos\phi}} \quad (8)$$

Equation 8 thus gives an approximation for the nutation angle, $\delta\Theta$, induced by an attitude maneuver of size Γ that is subdivided into a number of small burns, n. It is immediately apparent from this equation that the induced nutation angle can be reduced by sufficiently increasing the value of n, i.e., by dividing the maneuver into as many small burns as possible. For ACE, the smallest allowable burn time is dictated by the thruster command resolution and is equal to 32 msec. It is also clear from equation 8 that the induced nutation angle is directly proportional to the size of the maneuver, Γ. Thus, nutation could be reduced by performing attitude maneuvers more frequently than every 6 days and by maintaining a tighter deadband on the HGA-nadir angle than the allowable 3 degrees.

Based on this knowledge, we can estimate how much nutation would be induced for a maneuver using a ±1 degree deadband and the maximum number of small burns (minimum pulsewidth.) This gives a burn arc half-angle of

$$\alpha = \frac{1}{2} \cdot (0.032\,\text{sec}) \cdot (30°/\text{sec}) = 0.48°$$

From equation 3, we compute that each small burn moves the spin axis 0.0176°; thus it will take approximately 114 small burns of a lower deck radial thruster to achieve the 2° maneuver.

The function given in equation 8 are shown in Figures 2 and 3 for the two lowest values of ϕ, which represent thrusting once per 1/2 revolution and once per revolution, respectively. (*Higher values of ϕ were also analyzed. However, the results were essentially the same as those for the two lowest values of ϕ. Since the time required to execute a maneuver increases with ϕ, the results from using higher values of ϕ are not presented.*) Figures 2 and 3 show that the functions exhibit a sinusoidal behavior under an envelope proportional to 1/n. The envelope represents the maximum expected induced nutation angle for an attitude maneuver subdivided into an integral number of small burns. The envelope is minimized for the case of 1 small burn executed per revolution (Figure 3.)

An analytic expression can be derived for the envelope by using Taylor Series expansion of the cosine terms and assuming that the ratio I_3/I_T is close to 1.50. (*This number gives values of ϕ that are integral multiples of $\pi/2$ and simplifies the cosine terms in equation 8.*) This assumption is valid for the deployed ACE spacecraft configuration from BOL to EOL. The Taylor Series expansion reduces to:

$$\delta\Theta_{\max} = \frac{\Gamma}{n}\cdot\sqrt{2[1+(2\phi_1 - 3\pi)]} \qquad (9)$$

for $\phi_1 = \pi^* I_3/I_T$ (1 small burn per 1/2 revolution, as seen in Figure 2),

or,

$$\delta\Theta_{\max} = \frac{\Gamma}{n}\cdot\sqrt{1-\frac{(\phi_2 - 3\pi)^2}{12}} \qquad (10)$$

for $\phi_2 = 2\pi^* I_3/I_T$ (1 small burn per 1 revolution, as seen in Figure 3)

The size of the envelope at the far right end of the horizontal axis in each figure represents how much nutation should be expected at the end of a maneuver when using a single lower deck radial thruster. The induced nutation angle is approximately 0.0237 ° when using ϕ_1 and 0.0177 ° when using ϕ_2. This suggests that the final nutation angle for ϕ_2 will be only 75% of that for ϕ_1. Thus, even though using ϕ_2 will take twice as long as using ϕ_1 to execute the maneuver (22.6 minutes vs. 11.6 minutes), more than 2-¾ hours [*ln(0.0237)·9.5 hours)*] of nutation damping time can be avoided. When using the axial thrusters, the maneuver will be complete after about 58 small burns. The induced nutation will be nearly twice as high (approximately 0.0461 ° for ϕ_1 and 0.0345 ° for ϕ_2) as with the radial thrusters. This increase is caused because the larger moment arm imparts larger, less impulsive, torques on the spacecraft with each thruster burn. Therefore, it is desirable to use the radial thrusters instead of the axial thrusters to perform attitude maneuvers.

Note that the 1/n envelope represents a conservative estimate of what the induced nutation will be after n thruster firings. In theory, the induced nutation should actually be less, as dictated by equation 8. The true nutation angle should actually fall along the sinusoidal curve as shown in Figure 3. In practice, though, it will be difficult to predict the true shape of the sinusoid, since that would require an accurate prediction of the phase angle, ϕ, and consequently the ratio I_3/I_T. Predictions of the inertia properties, based on onboard fuel estimates, will be maintained for ACE, but may be in error by several percentage points.

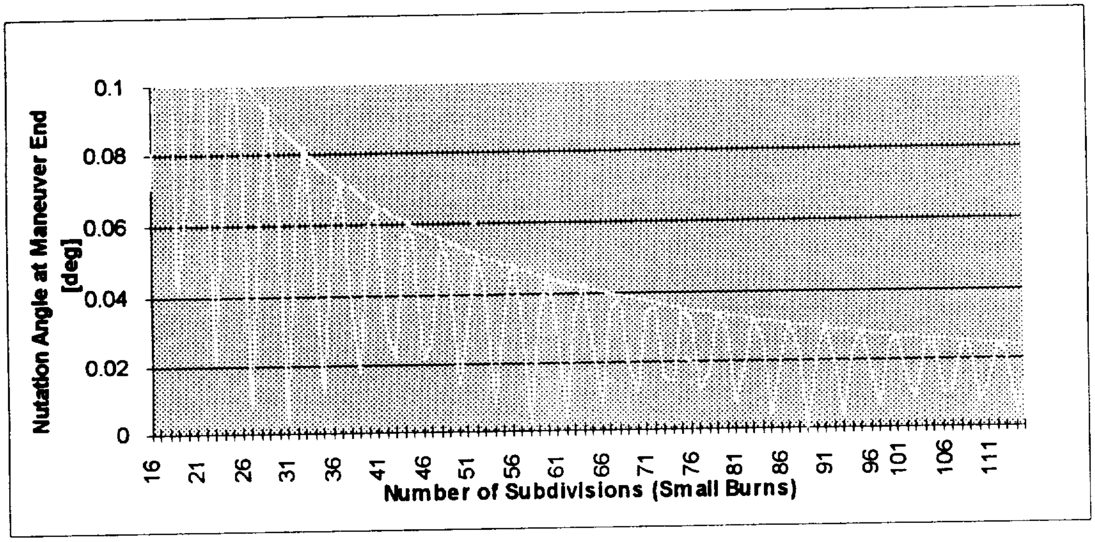

Figure 2. ACE Nutation: 1 Burn per 1/2 Revolution

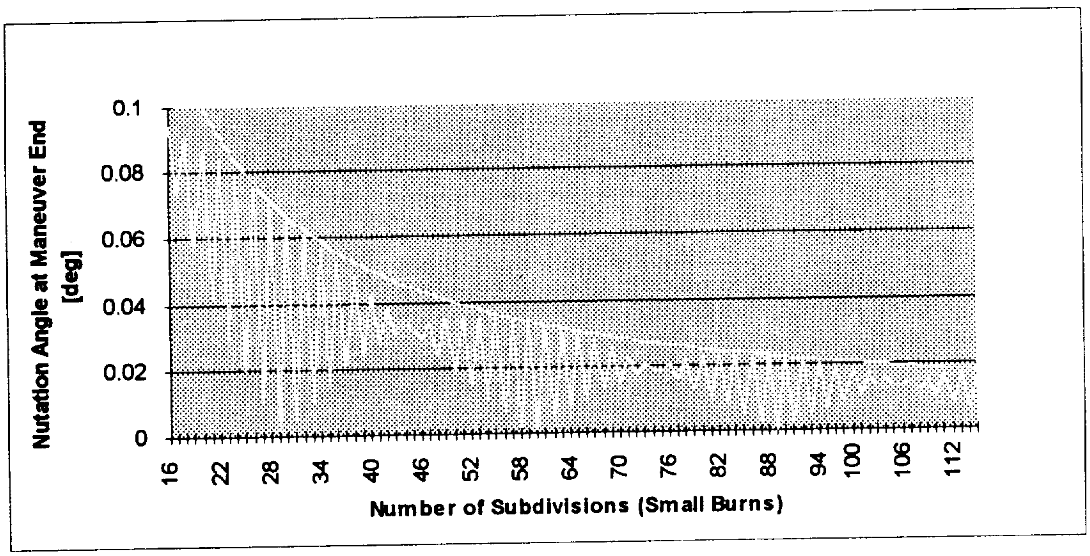

Figure 3. ACE Nutation: 1 Burn per 1 Revolution

III. VERIFICATION VIA SIMULATION

A dynamics simulator developed from PC-MATLAB was used to verify the accuracy of the analytic equation for the induced nutation angle given in Equation 8. The dynamics simulator models the ACE thruster burns and integrates the equations of motion to predict the effect on the spacecraft attitude.

The dynamics simulator was run and compared with the results shown in Figure 3. The following scenario was used in the simulation:

- a 2° attitude maneuver was modeled,
- firing of the lower deck thruster, IIIR, was simulated,
- thruster pulsing was once per spacecraft revolution,
- the minimum on-time of .032 sec was used,
- the simulation was run for 1400 sec (approximately 116 thruster firings.)

The simulation results are shown in Figure 5. The Figure shows the actual nutation angle as it changes during the course of the 2° attitude maneuver. The nutation angle changes after each thruster burn, and varies from about 0.001° to 0.032°. At the end of the attitude maneuver (after 1368 seconds), the residual nutation is approximately 0.024°. This agrees within about 25% to the value of 0.0177° derived in the previous section.

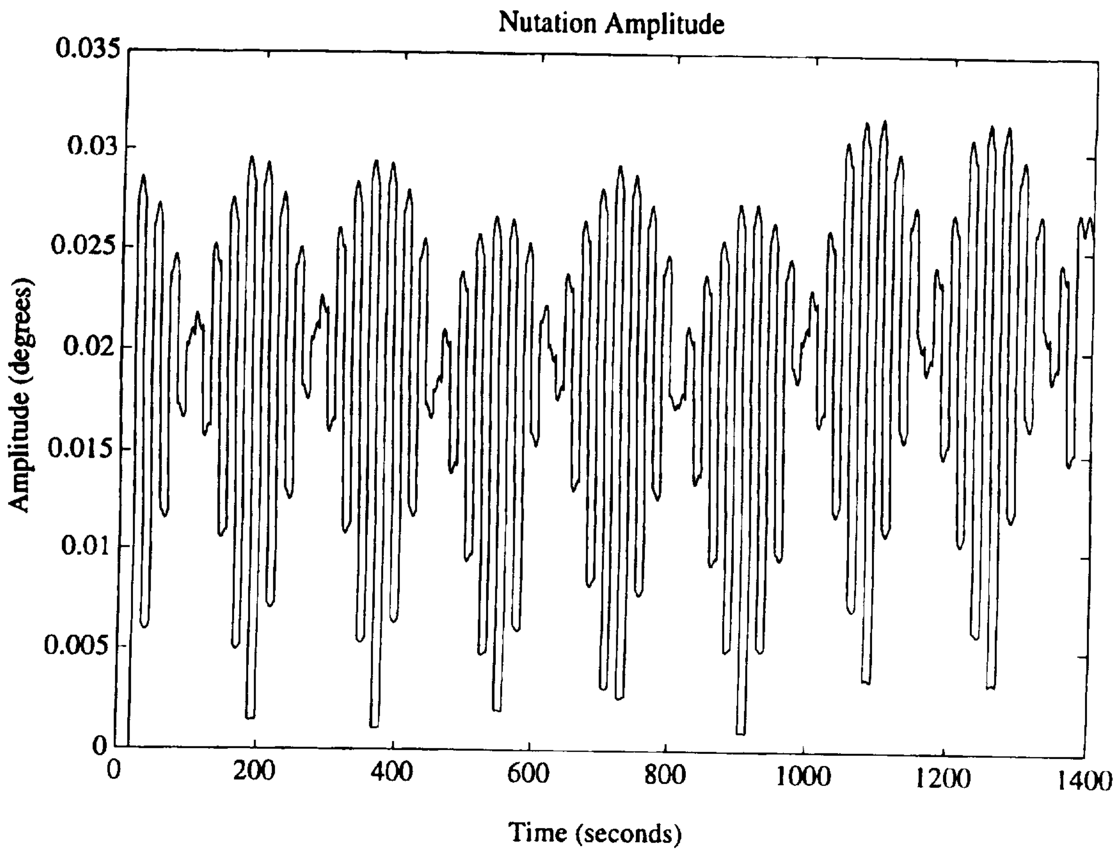

Figure 4. Dynamics Simulation of ACE Nutation During an Attitude Maneuver

IV. NUTATION MONITORING

Following each attitude maneuver, ground operations personnel will need to quantify the amount of nutation induced. From the preceding analysis, we can expect nutation to be somewhat less than 0.025°. The nutation angle can be computed from the telemetered sensor data provided that the sensor accuracy is less than the actual nutation angle. The ACE Sun sensor is the Adcole model 27990. It is accurate only to ~0.25°, so it will not be useful for observing small nutation angles. The star tracker is the Ball Aerospace (BASG) model CT-631. It is accurate to ~0.025° (~1.5 arc-min) and therefore may provide observability of post-maneuver nutation angles.

The CT-631 sensor has a 20° x 20° field of view (FOV) and is typically used on three-axis stabilized spacecraft. Some modifications to the onboard star searching and tracking algorithms allow its use on a spinning spacecraft, such as ACE. For ACE, the star tracker is mounted with its boresight 90° form the spacecraft Z-axis. As the spacecraft spins, the star tracker views a swath of the sky that is 20° x 360°. Due to star tracker processing limitations, this swath cannot be viewed as a continuum, but must be divided into 900 "pickets". Each picket is a region of the sky which is 0.4° x 20°. The FOV swath is viewed as a "picket fence", which has all 900 pickets placed in adjunction. Figure 5 shows the pickets which the ACE star tracker searches during one spacecraft revolution. During each full spacecraft revolution, the star tracker views every 8th picket and searches for stars within those pickets. During subsequent spacecraft revolutions, the "picket fence" is advanced by one picket, and every 8th picket is again observed. After 8 complete revolutions (nominally every 96 seconds), the entire 20 degree field of view band has been searched.

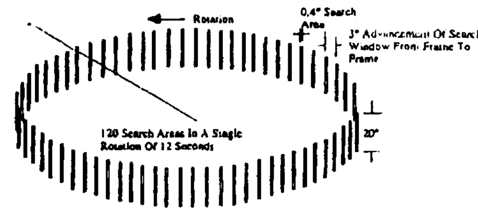

Figure 5. ACE Star Tracker Sky Search

For a nutating spacecraft, a fixed celestial object (such as a star) varies in declination sinusoidally relative to a spacecraft-mounted sensor (such as a star tracker) with an amplitude equal to twice the nutation angle. This effect is illustrated in Figure 6. The high-frequency sinusoidal curve represents the actual declination angle relative to the sensor and the 8 squares represent the observed declination angle, sampled every 96 seconds. The spacecraft nutation angle can then be observed by collecting enough simultaneous observations of the object such that the object passes through the full range of declination angles.

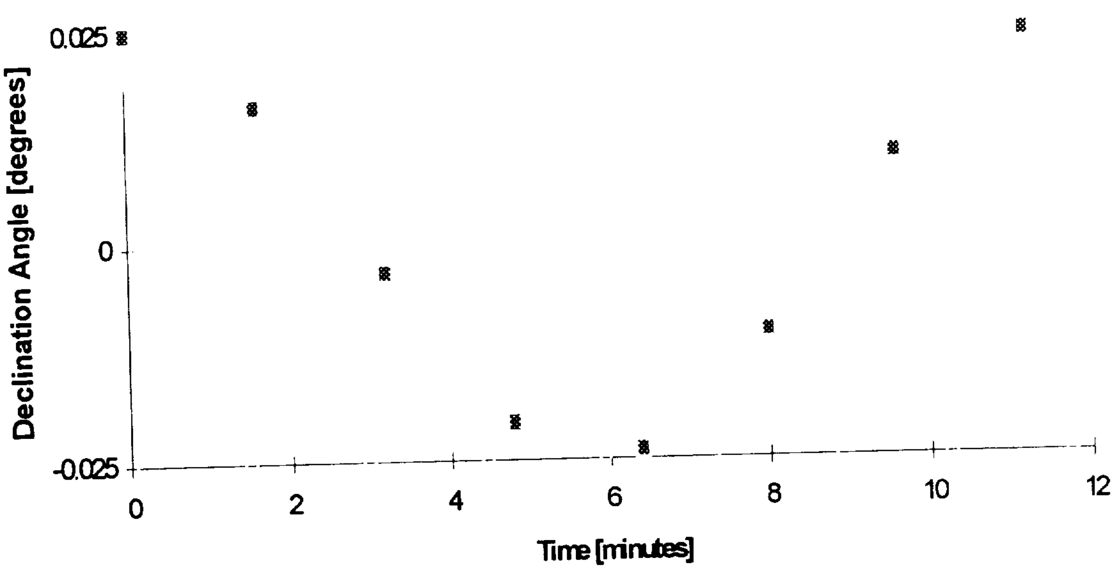

Figure 6. Effect of Nutation on Star Tracker Observations

The object also precesses at the inertial nutation rate, Ω_I, defined as

$$\Omega_I = \left(\frac{I_3}{I_T}\right) \cdot \Omega_s \qquad (11)$$

where

I_3 = the major moment of inertia ,
I_T = the transverse moment of inertia, defined as $(I_1 \cdot I_2)^{1/2}$, and
Ω_s = the spacecraft spin rate.

For ACE, the ratio I_3/I_T is always fairly close to a value 1.5. This means that between subsequent observations of a given star (after 8 spacecraft revolutions), the star has processed along its sinusoidal path (relative to the sensor) by approximately $1.5 \cdot 8 \cdot 360$ degrees, or approximately 12 complete revolutions. Since a sine curve repeats itself after each complete 360 degree cycle, subsequent observations of a given star by the ACE star tracker will show little change in declination. Thus many star observations will need to be collected in order to determine the nutation angle. The number of observations required is a function of actual spin rate and the inertia ratio. Figure 6 illustrates this effect for ACE, using typical values of 5 RPM for the spin rate and 1.483 for the inertia ratio. After about 8 samples, or 11.2 minutes, the observed declination angle has gone through the full range of declination angles.

In the general case, the amount of time required to observe the full range of declination angles depends on the actual inertial nutation rate, which is a function of both the actual spin rate and inertia ratio, as implied in equation 11. A parametric study will be performed to see how that function behaves for ACE. In general, it is desirable to minimize the amount of time needed to observe the nutation effect.

For ACE, the spin rate is required to be maintained at 5.0 ± 0.1 RPM. The inertia ratio, I_3/I_T, will be approximately 1.494 at beginning-of-life and, as propellant is expelled, will decrease to a value approximately 1.471 at end-of-life. Thus, the following relations can be used:

$$\Omega_s = \Omega_0 + \varepsilon_\Omega \qquad (12)$$

where

Ω_s = actual spacecraft spin rate,
Ω_0 = nominal spin rate, $\pi/6$ rad/sec,
ε_Ω = allowable spin rate variation, $|\varepsilon_\Omega| \leq \pi/300$ rad/sec,

and

$$\frac{I_3}{I_T} = 1.5 - \varepsilon_I \qquad (13)$$

where

ε_I = variation in inertia ratio, a small positive number.

These two equations can be substituted into the original equation to give the inertial nutation rate as

$$\Omega_I = (1.5 - \varepsilon_I) \cdot (\Omega_0 + \varepsilon_\Omega) \qquad (14)$$

which when expanded gives

$$\Omega_I \cong 1.5 \cdot \Omega_0 + 1.5 \cdot \varepsilon_\Omega - \Omega_0 \cdot \varepsilon_I \qquad (15)$$

After 8 complete revolutions, the phase shift, $\Delta\phi$, of the star will be

$$\Delta\phi = (8 \cdot 2\pi) \cdot \frac{\Omega_I}{\Omega_0}$$

or

$$\Delta\phi = 24\pi + 24\pi \cdot \frac{\varepsilon_\Omega}{\Omega_0} - 16\pi \cdot \varepsilon_I \qquad (16)$$

The term 24π can be ignored, since it is a multiple of 2π, and will not effect the phase shift computation. The absolute value of the remaining terms can be used to express the magnitude of the phase shift, whether left or right. The equation then becomes

$$\Delta\phi = \left| 24\pi \cdot \frac{\varepsilon_\Omega}{\Omega_0} - 16\pi \cdot \varepsilon_I \right| \tag{17}$$

This angle, $\Delta\phi$, again represents a small phase shift along the sinusoid, achieved after 8 successive revolutions. The nutation angle can only be characterized after many such small phase shifts take the observation through a complete phase cycle of 2π. The time, t, in minutes, required to complete this cycle is then

$$t = 8 \cdot \left(\frac{2\pi}{\Omega_0}\right) \cdot \left(\frac{2\pi}{\Delta\phi}\right) \cdot \left(\frac{1\,\text{min}}{60\,\text{sec}}\right) \tag{18}$$

which reduces to

$$t = \left| \frac{45 \cdot \varepsilon_\Omega}{\pi} - 5 \cdot \varepsilon_I \right|^{-1} \text{ minutes} \tag{19}$$

By using spin rates in units of RPM, the equation simplifies to

$$t = \left| 1.5 \cdot (\Omega_s - \Omega_0) + 5 \cdot \left(\frac{I_3}{I_T}\right) - 7.5 \right|^{-1} \tag{20}$$

Equation 20 is plotted in Figure 7 using the full range of values for Ω_s and I_3/I_T.

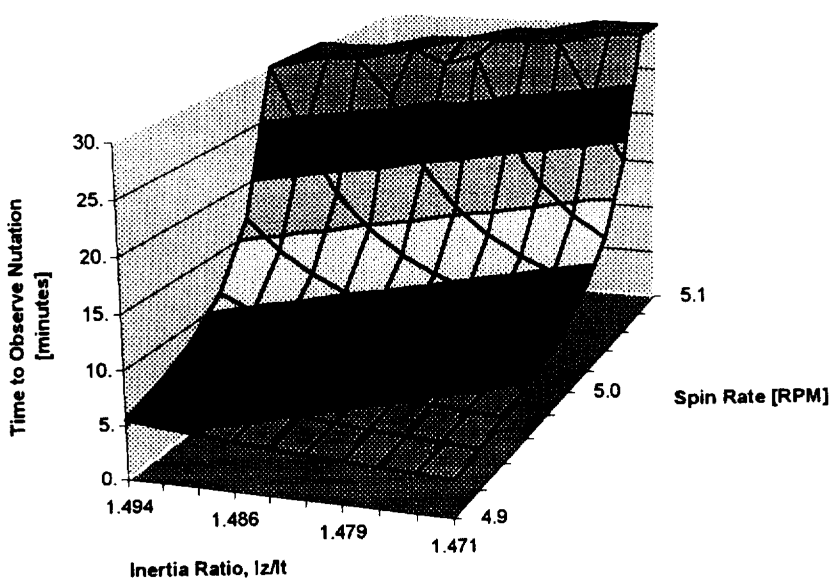

Figure 7. Nutation Observation Times

Figure 7 shows graphically the time, in minutes, required to observe the ACE nutation angle for a given spin rate and inertia ratio. The graph is truncated on the vertical axis at 30 minutes; approximately 20% of the combinations of spin rate and inertia ratio are above this plateau and will require more than 30 minutes of star data to characterize the nutation angle. This indicates that for randomly chosen values of spin rate and inertia ratios, we can observe the nutation angle within 30 minutes 80% of the time. This percentage can be increased to nearly 100% by maintaining a spin rate of less than 5 RPM. Nutation observation time also decreases as the inertia ratio decreases; this will be the trend as the life of the mission goes on, and hydrazine fuel is periodically used to perform maneuvers.

V. SUMMARY AND CONCLUSIONS

Strategies have been developed for planning and executing ACE attitude maneuvers that will allow spacecraft nutation to be controlled. Nutation monitoring techniques have also been identified. In summary,

- by properly subdividing the attitude maneuvers, the induced nutation can be kept low
- there is an advantage to using one thruster pair pulse per revolution instead of two
- performing the maneuvers with the radial instead of the axial thrusters will reduce the induced nutation by a factor of nearly 1/2
- maintaining a spin rate of less than 5 RPM will assist in quickly assessing the post-maneuver nutation angle. As a secondary advantage, a lower spin rate will provide the spacecraft with less gyroscopic stiffness and allow attitude maneuvers to be performed more efficiently.

VI. ACKNOWLEDGMENTS

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