THE EFFECTS OF MOMENTUM BIAS ON A GRAVITY GRADIENT STABILIZED SPACECRAFT WITH ACTIVE MAGNETIC CONTROL

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ABSTRACT

The improvements achieved by adding a momentum bias wheel to a Gravity Gradient (GG) stabilized spacecraft are evaluated. Mass, power, and computational processing requirements, as well as performance, are compared for three Attitude Determination and Control Subsystem (ADACS) scenarios.

Spacecraft which require low mass and power have long incorporated GG torques as a passive stabilization technique. The spacecraft is oriented in the general direction required by the mission, but the overall attitude and attitude rate errors are not exceptionally tight. In order to improve the spacecraft pointing accuracies, the GG stabilized ADACS can be augmented by an active control technique. Previously, the use of three TORQRODs™ was evaluated, with one oriented along each of the spacecraft axis. For this analysis, the incorporation of a small momentum bias wheel is shown to significantly improve the magnetic attitude control from a few degrees to a few tenths of a degree by providing additional gyroscopic stiffness. The ADACS impacts of a constant speed wheel vs an active pitch control loop are also compared.

Attitude control techniques are one part of the overall ADACS solution. The knowledge of how well the spacecraft attitude can be determined defines the net ADACS performance for a given mission scenario. If an accuracy of few degrees is sufficient, a novel approach is to determine the attitude simply from three-axis magnetometer data. The addition of an Earth horizon sensor to provide accurate roll and pitch information can improve the overall ADACS performance to approximately 0.5°, but at the expense of increased mass and power requirements.

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NOMENCLATURE

X,Y,Z	Spacecraft body roll, pitch, and yaw axis, respectively, with X along the velocity vector, Y along the negative orbit normal, and Z toward the nadir
$\underline{H}$	Total momentum vector
$\underline{\omega}$	Spacecraft body rate vector
$\underline{h}$	Wheel momentum vector = $\begin{bmatrix} 0 \\ h_2 \\ 0 \end{bmatrix}$
I	Inertia matrix of the spacecraft
$\underline{T}_{gg}$	Gravity gradient torque vector
$\underline{T}_d$	Disturbance torque vector
$\underline{T}_c$	Control torque vector
$\underline{M}$	Control dipole moment vector
$\underline{B}$	Earth magnetic field vector
$\underline{q}$	Quaternion vector associated with spacecraft attitudes
ϕ, θ, ψ	Roll, pitch, and yaw errors, respectively
A(.)	Direction cosine matrix
ω_o	Orbital rate
$\underline{\omega}_{sc}$	Spacecraft body rate excluding ω_o

INTRODUCTION

Passive stabilization techniques using GG torques have been in use for a long time, specifically for damping the libration motion of a spacecraft [1]. This technique does not use any additional sensors or actuators, if the spacecraft can be designed in such a way that it is GG stabilized. In the worst case, a GG boom may be required if the main body of the spacecraft itself can not be designed to satisfy the GG stabilization criterion due to mission dictated objectives. Even though this technique works well, it generally requires a long time to accomplish the libration damping (on the order of a few days). Moreover the attitude control errors are fairly loose (5° to

10°), which may be adequate to meet some mission requirements.

To improve the libration damping time and the attitude control errors, an active magnetic control technique using three TORQRODs has been suggested for a class of small satellites ranging in total mass from 40 to 200 kg [2]. This active control can reduce the libration damping time from days to a few orbits, and can achieve attitude control errors of less than 3° for roll, 2° for pitch, and 5° for yaw.

The above magnetic control technique is very efficient in significantly reducing the long libration damping time, but the attitude control accuracy is still moderate and may be unacceptable for some missions. Thus, in addition to the above magnetic control technique, a constant speed bias momentum wheel can be incorporated to decrease the attitude control errors to less than 0.2° per axis. This improvement in control accuracy is achieved at the expense of an increase in the mass and power required for the ADACS.

The above quoted errors are only those attributable to the attitude control portion of the ADACS, i.e., an assumption of "perfect" attitude knowledge was made. As inputs to the control algorithms, all attitude errors and respective rates can be estimated solely from magnetometer measurements using a Kalman Filter scheme [3]. It has been shown that this technique can provide attitude information to better than 1.8°, 2.4°, and 2.8° for roll, pitch, and yaw (3σ), respectively. The overall attitude accuracy can then be calculated as the Root-Sum-Squared (RSS) of the control and the knowledge errors.

A significant improvement in the overall attitude performance can be achieved by adding an Earth horizon sensor to provide a more accurate estimate of the spacecraft roll and pitch attitude. This improvement in attitude allows the addition of a pitch control loop around the bias momentum wheel. The pitch axis TORQROD is used for roll/yaw control, while the TORQRODs along the roll and yaw axes are activated to reduce the excess momentum accumulated in the wheel. The overall resulting attitude accuracies (3σ) are in the neighborhood of 0.3° for both roll and pitch, and 0.6° for yaw, but the ADACS mass and power requirements are again increased.

SPACECRAFT DYNAMICS

Equations of Motion

The dynamic equations of motions of the spacecraft are given by

$$\frac{d\underline{\mathbf{H}}}{dt} + \underline{\omega} \times \underline{\mathbf{H}} = \underline{\mathbf{T}}_d - \underline{\mathbf{T}}_c$$

where $\underline{\mathbf{H}} = I\,\underline{\omega} + \underline{\mathbf{h}}$

and the kinematic equations of motion of the attitude are given by

$$\frac{d\underline{\mathbf{q}}}{dt} = \frac{1}{2}\Omega(\underline{\omega}_{sc}) \cdot \underline{\mathbf{q}}$$

where

$$\Omega(\underline{\omega}_{sc}) = \begin{bmatrix} 0 & \omega_{zsc} & -\omega_{ysc} & \omega_{xsc} \\ -\omega_{zsc} & 0 & \omega_{xsc} & \omega_{ysc} \\ \omega_{ysc} & -\omega_{xsc} & 0 & \omega_{zsc} \\ -\omega_{xsc} & -\omega_{ysc} & -\omega_{zsc} & 0 \end{bmatrix}$$

$$\underline{\omega}_{sc} = \underline{\omega} - A(\underline{\mathbf{q}}) \cdot \begin{bmatrix} 0 \\ -\omega_0 \\ 0 \end{bmatrix}$$

$$A(\underline{\mathbf{q}}) = \begin{bmatrix} A_{11} & A_{12} & A_{13} \\ A_{21} & A_{22} & A_{23} \\ A_{31} & A_{32} & A_{33} \end{bmatrix}$$

$$= \begin{bmatrix} q_1^2-q_2^2-q_3^2+q_4^2 & 2(q_1q_2+q_3q_4) & 2(q_1q_3-q_2q_4) \\ 2(q_1q_2-q_3q_4) & -q_1^2+q_2^2-q_3^2+q_4^2 & 2(q_2q_3+q_1q_4) \\ 2(q_1q_3+q_2q_4) & 2(q_2q_3-q_1q_4) & -q_1^2-q_2^2+q_3^2+q_4^2 \end{bmatrix}$$

with $q_1^2 + q_2^2 + q_3^2 + q_4^2 = 1$

Assuming Z-X-Y Euler rotations, the attitude errors can be obtained as follows:

$$\phi = \arcsin^{-1}(A_{23})$$

$$\theta = \arctan^{-1}\left(-\frac{A_{13}}{A_{33}}\right)$$

$$\psi = \arctan^{-1}\left(-\frac{A_{21}}{A_{22}}\right)$$

Gravity Gradient Control With Magnetics

The baseline spacecraft is gravity gradient stable and incorporates three TORQRODs for active control. The control torque has two parts: one part is passive ($\underline{\mathbf{T}}_{cp}$) while the other is active ($\underline{\mathbf{T}}_{cm}$).

Defining $\underline{\mathbf{R}}_B$ as the vector representing the distance of the spacecraft Center of Gravity (CG) from the Earth center in the spacecraft frame, the GG torque in the spacecraft body frame can be expressed as [1]

$$\underline{\mathbf{T}}_{gg} = \frac{3\mu}{|R_B|^3} \cdot [\underline{\mathbf{r}}_B \times (I \cdot \underline{\mathbf{r}}_B)]$$

where
$\underline{\mathbf{r}}_B = \frac{\underline{\mathbf{R}}_B}{|R_B|}$
I = spacecraft inertia matrix
μ = Earth gravitation constant
= 3.986005×10^5 km^3/s^2

For circular orbits (as for this analysis), the gravity gradient torque vector is given by

$$\underline{\mathbf{T}}_{gg} = 3\,\omega_0^2 \cdot \underline{\mathbf{r}}_B \times (I\,\underline{\mathbf{r}}_B)$$

where $\underline{\mathbf{r}}_B = A(\underline{\mathbf{q}}) \cdot \begin{bmatrix} 0 \\ 0 \\ -1 \end{bmatrix}$

For this analysis, the passive control vector $\underline{\mathbf{T}}_{cp}$ is simply equal to $\underline{\mathbf{T}}_{gg}$.

Active Magnetic Control

The active control torque created by the TORQRODs is given by

$$\underline{\mathbf{T}}_{cm} = \underline{\mathbf{M}} \times \underline{\mathbf{B}}$$

where $\underline{\mathbf{B}}$ is the Earth's magnetic field vector and $\underline{\mathbf{M}}$ is the dipole moment vector generated by the TORQRODs (both in the spacecraft body frame). The $\underline{\mathbf{M}}$ vector is given by

$$\underline{\mathbf{M}} = \frac{\underline{\mathbf{m}} \times \underline{\mathbf{B}}}{|\underline{\mathbf{B}}|}$$

with $\underline{\mathbf{m}} = K_p \cdot \underline{\mathbf{a}} + K_d \cdot \dot{\underline{\mathbf{a}}}$

where $\underline{\mathbf{a}}$ = attitude error vector = $\begin{bmatrix} \phi \\ \theta \\ \psi \end{bmatrix}$

$\dot{\underline{\mathbf{a}}}$ = attitude rate vector = $\begin{bmatrix} \dot\phi \\ \dot\theta \\ \dot\psi \end{bmatrix}$

and K_p , K_d are 3 x 3 diagonal matrices corresponding to position and rate gain components, respectively. The total control torque applied to the spacecraft is given by

$$\underline{\mathbf{T}}_c = \underline{\mathbf{T}}_{cp} + \underline{\mathbf{T}}_{cm}$$

Bias Momentum Augmentation

The ADACS performance of a gravity gradient stable spacecraft with magnetic control can be significantly improved by the addition of a constant speed momentum wheel aligned with the pitch (Y) axis. For this analysis, a constant speed pitch wheel with a momentum of 0.375 N-m-s is used to provide bias momentum rigidity, but the active attitude control is provided by the TORQRODs. (An active pitch control loop via the wheel was not consider here with the magnetic attitude determination scheme. This is a viable alternative, which will be investigated in future analyses.)

The total momentum of the spacecraft with a constant bias level is given by

$$\underline{\mathbf{H}} = I\underline{\omega} + \begin{bmatrix} 0 \\ -h_2 \\ 0 \end{bmatrix}$$

where $h_2 = 0.375$ N-m-s

As before, the total control torque vector $\underline{T}_{cm}$ applied to the spacecraft is

$$\underline{T}_{cm} = \underline{M} \times \underline{B}$$

with $\underline{M}$ as defined previously.

SIMULATED PERFORMANCE AND HARDWARE SELECTION

A candidate spacecraft was selected for these analyses, with a circular sun synchronous polar orbit at an altitude of 800 km (432 nmi). The spacecraft was assumed to have a mass distribution such as to provide an inertia matrix which assures that the spacecraft is GG stable. Table 1 lists the spacecraft characteristics, the orbit parameters, and the initial conditions used for the following simulations.

The predicted ADACS performance both with and without the bias momentum augmentation was simulated via computer, and the attitude control results are presented in Figures 1 through 3 without the wheel, and in Figures 4 through 6 with the wheel. Gravity gradient, aerodynamic, solar pressure, and residual magnetic dipole effects were included. It must be emphasized that these simulations assume a perfect knowledge of the spacecraft attitude. As stated previously, the attitude determination errors must be RSS'd with the attitude control errors in order to define the overall pointing accuracy (3σ) of the ADACS.

The spacecraft is controlled through the use of three magnetic TORQRODs, one oriented along each of the spacecraft axes. A pair of two-axis magnetometers is used to measure the Earth's magnetic field vector with respect to the spacecraft body axes. This data is used to select the appropriate TORQROD combination for control, as well as to provide a real time estimate of the spacecraft three-axis attitude after processing via the spacecraft computer. A simple control electronics assembly is required for driving the TORQRODs and for interfacing the ADACS equipment with the spacecraft. The momentum wheel is driven at a constant speed of approximately 500 rpm to provide gyroscopic rigidity which improves the overall attitude control performance. The size, mass, and power characteristics of the hardware components required for these two control scenarios are summarized in Tables 2 and 3.

Table 1: Spacecraft Characteristics, Orbit Parameters, and Initial Conditions

Spacecraft Mass		45 kg
Spacecraft Inertia Matrix	I_{xx}	0.50 kg-m^2
	I_{yy}	0.65 kg-m^2
	I_{zz}	0.20 kg-m^2
Spacecraft CG-CP Distance *	ΔX	6.5 mm
	ΔY	6.5 mm
	ΔZ	13.0 mm
Residual Dipole Per Axis		0.05 Am^2
Orbit Altitude		800 km
Orbit Inclination		98.6°
Orbit Eccentricity		0.0
Bias Momentum		0.375 N-m-s
Initial Attitude		10° per axis
Initial Attitude Rate		0.0 deg/sec per axis

* Offset between the spacecraft Center of Gravity (CG) and Center of Pressure (CP)

Table 2: Gravity Gradient ADACS Hardware Component Summary

Qty	Component	Unit Dimensions (cm)	Unit Mass (kg (lb))	Unit Avg/Peak Power (W)
2	Two-Axis Magnetometer	11.5 x 5.9 x 2.6	0.25 (0.55)	0.04
3	1 Am^2 TORQROD	1.3 d x 12.7	0.08 (0.18)	0.025 / 0.25
1	ADACS Electronics Mag A/Ds / TQR Drivers	16.8 x 17.0 x 6.2	0.9 (2.0)	0.75
Total			1.7 (3.6)	0.9 / 1.6

Table 3: GG / Fixed Bias Momentum ADACS Hardware Component Summary

Qty	Component	Unit Dimensions (cm)	Unit Mass (kg (lb))	Unit Avg/Peak Power (W)
2	Two-Axis Magnetometer	11.5 x 5.9 x 2.6	0.25 (0.55)	0.04
3	1 Am^2 TORQROD	1.3 d x 12.7	0.08 (0.18)	0.025 / 0.25
1	T-Wheel (A-Size)	20.4 d x 8.0	2.5 (5.5)	1.0
1	ADACS Electronics Mag A/Ds / TQR Drivers Motor Driver	16.8 x 17.0 x 6.2	1.4 (3.0)	1.0
Total			4.6 (10.1)	2.2 / 2.8

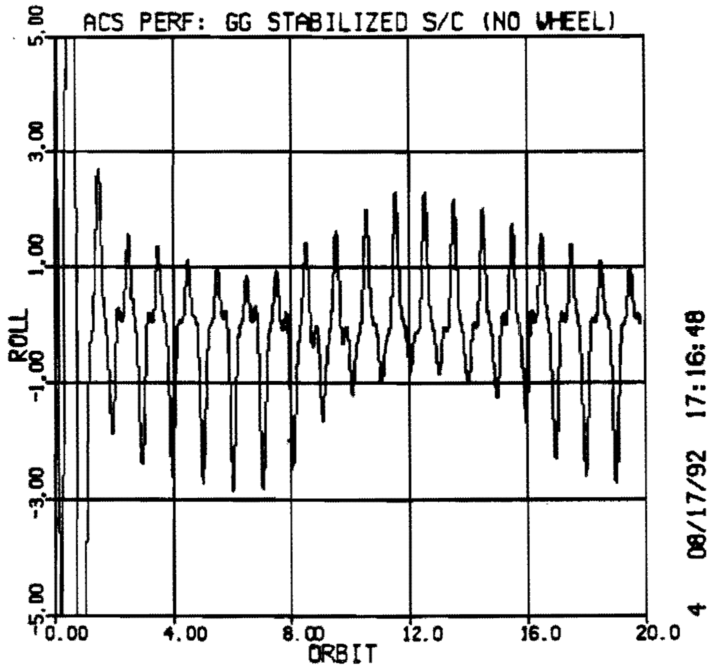

Figure 1a: Gravity Gradient Stabilization With Magnetic Control Roll Attitude Error (deg) vs. Orbit Number

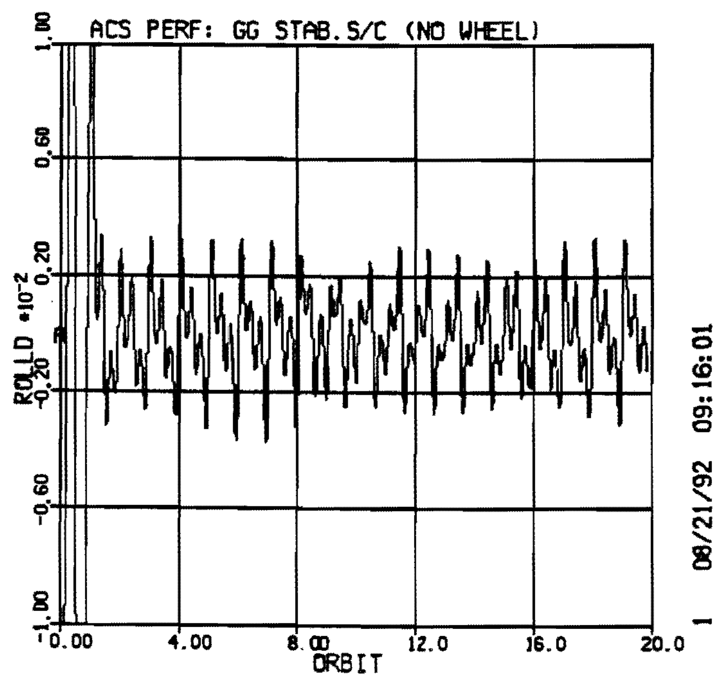

Figure 1b: Gravity Gradient Stabilization With Magnetic Control Roll Attitude Rate Error (deg/sec) vs. Orbit Number

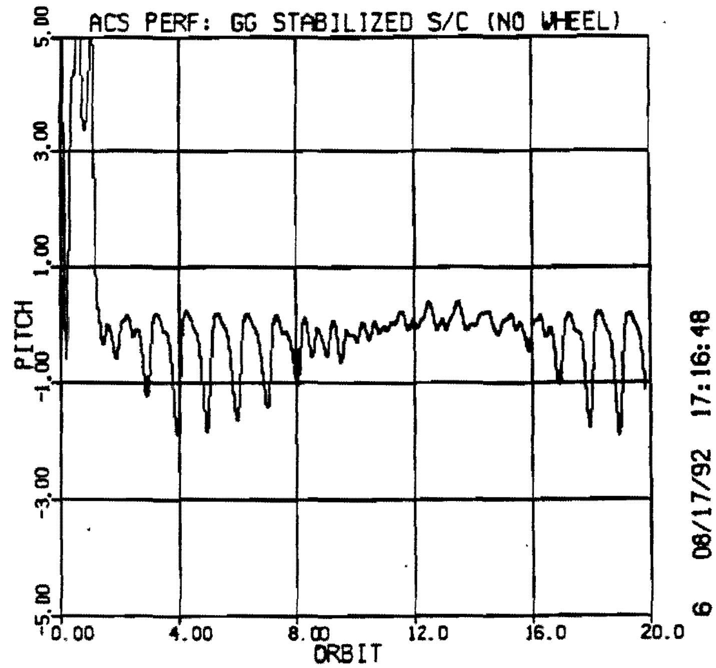

Figure 2a: Gravity Gradient Stabilization With Magnetic Control Pitch Attitude Error (deg) vs. Orbit Number

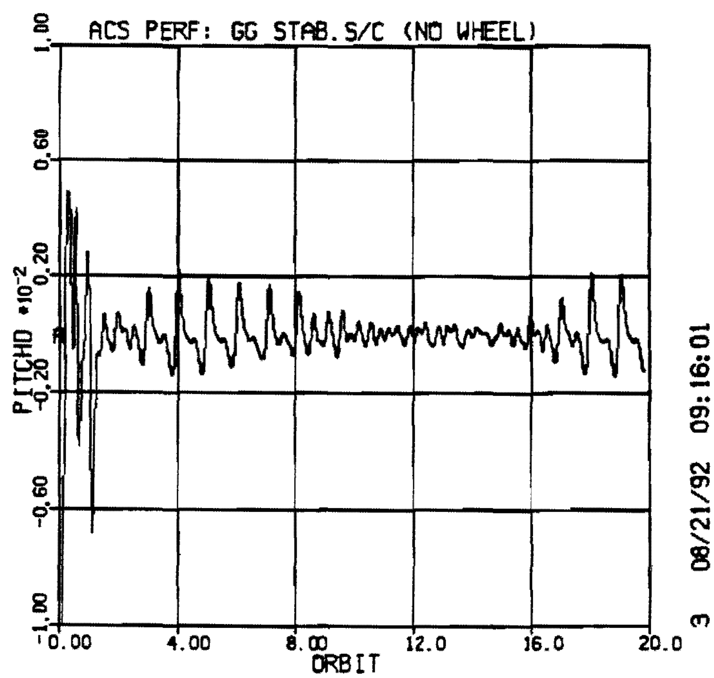

Figure 2b: Gravity Gradient Stabilization With Magnetic Control Pitch Attitude Rate Error (deg/sec) vs. Orbit Number

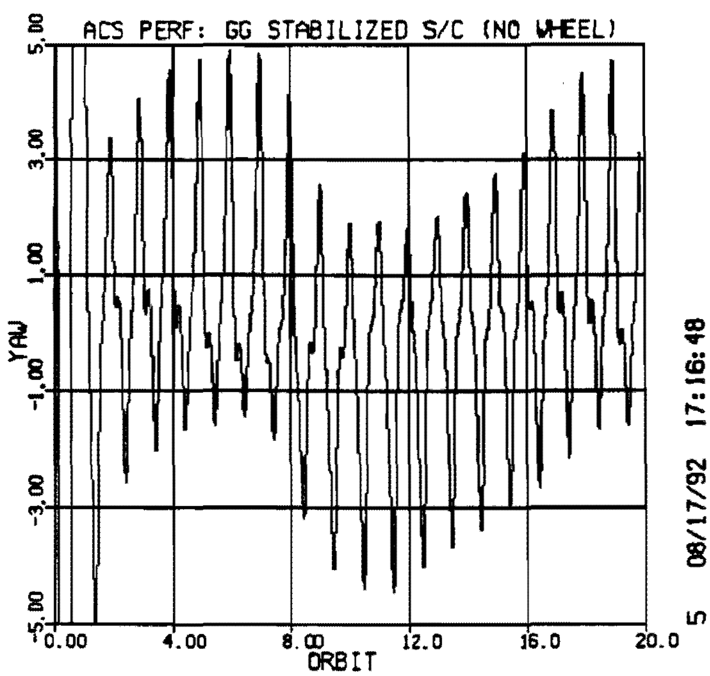

Figure 3a: Gravity Gradient Stabilization With Magnetic Control Yaw Attitude Error (deg) vs. Orbit Number

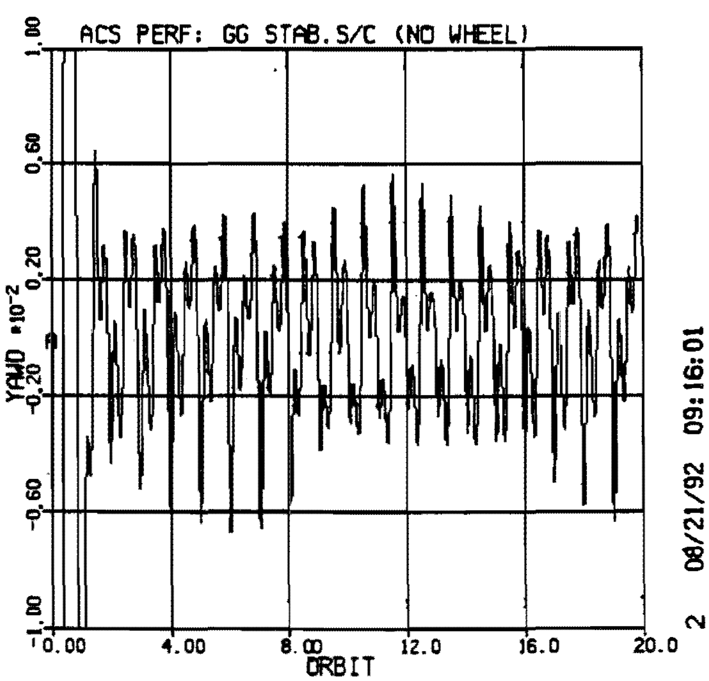

Figure 3b: Gravity Gradient Stabilization With Magnetic Control Yaw Attitude Rate Error (deg/sec) vs. Orbit Number

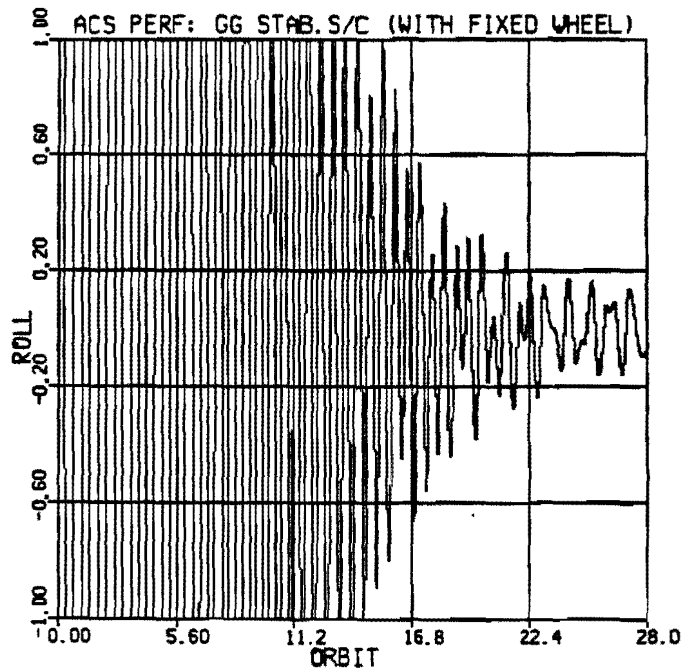

Figure 4a: Momentum Bias Stabilization With Magnetic Control Roll Attitude Error (deg) vs. Orbit Number

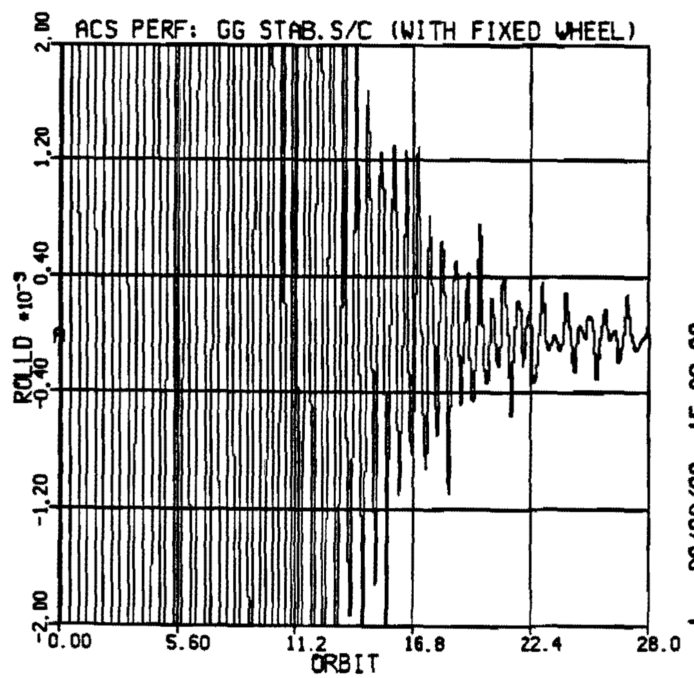

Figure 4b: Momentum Bias Stabilization With Magnetic Control Roll Attitude Rate Error (deg/sec) vs. Orbit Number

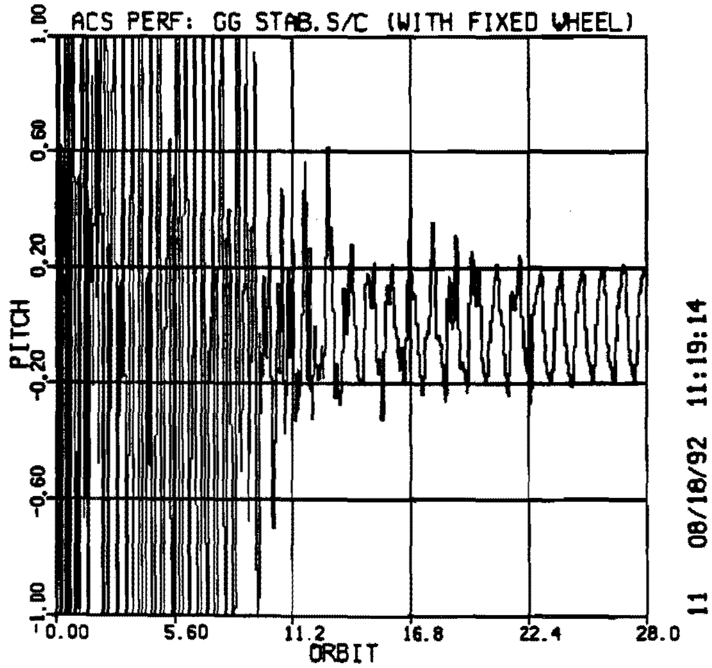

Figure 5a: Momentum Bias Stabilization With Magnetic Control Pitch Attitude Error (deg) vs. Orbit Number

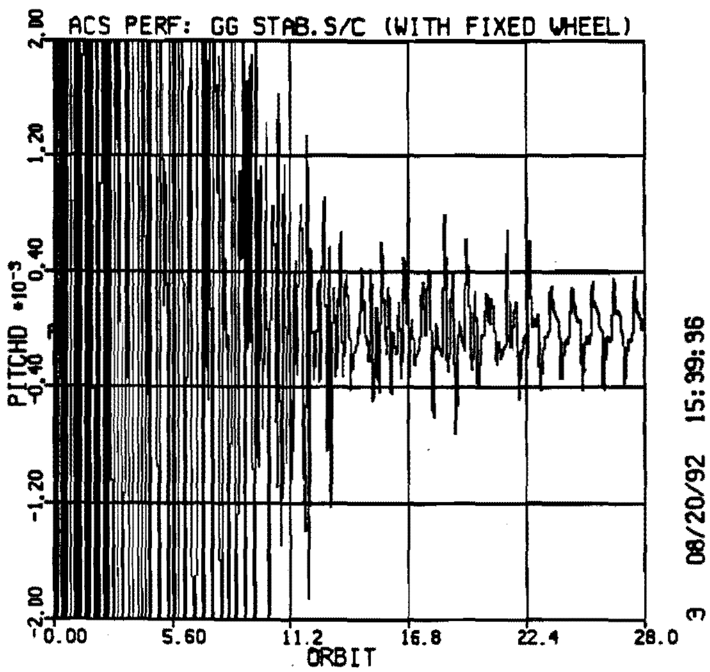

Figure 5b: Momentum Bias Stabilization With Magnetic Control Pitch Attitude Rate Error (deg/sec) vs. Orbit Number

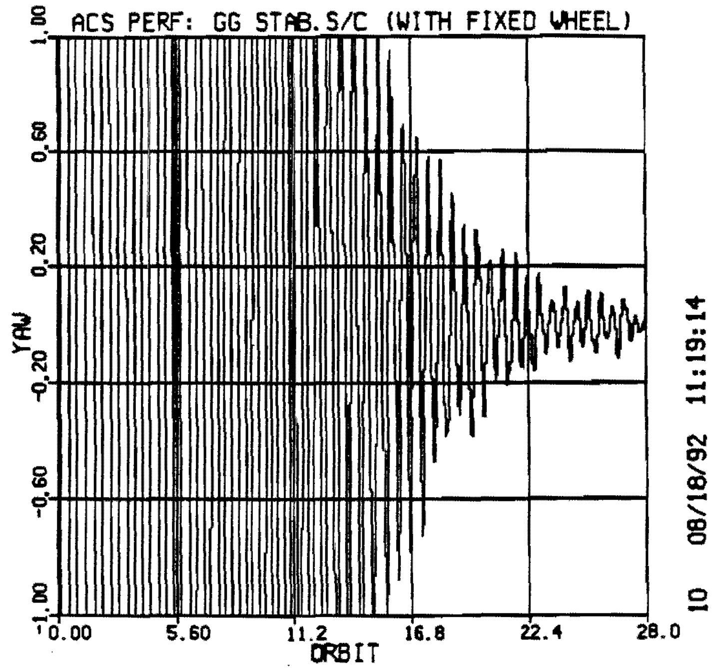

Figure 6a: Momentum Bias Stabilization With Magnetic Control Yaw Attitude Error (deg) vs. Orbit Number

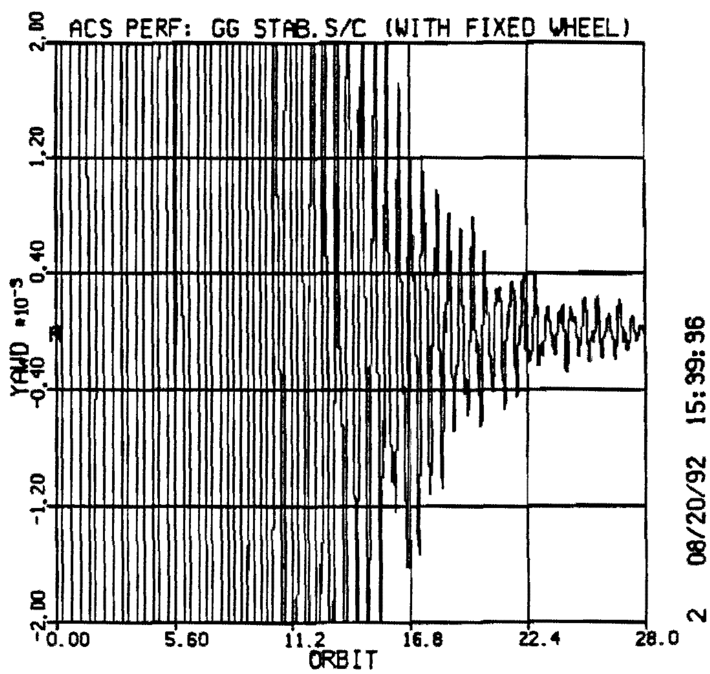

Figure 6b: Momentum Bias Stabilization With Magnetic Control Yaw Attitude Rate Error (deg/sec) vs. Orbit Number

All three TORQRODs are identical in design. Each TORQROD consists of a magnetic core and a coil, which provides a dipole moment of 1.2 Am^2 with an input voltage of 5V. When appropriately activated, the dipole moment interacts with the Earth's magnetic field to produce a control torque. Power is supplied to the TORQRODs directly from the spacecraft batteries. These units were built and successfully flown for the DARPA Special Communication Satellite Cluster (SCSC) series of spacecraft.

The magnetometer is a two-axis fluxgate unit which was specifically designed for low mass and low power consumption. ITHACO developed a similar magnetometer for the SCSC program. The magnetometer is self contained in that one box incorporates the two sensing coils, as well as the necessary processing electronics. The +5V regulated power is supplied directly from the spacecraft. Two units provide the required three-axis spacecraft measurements, as well as a limited redundant capability.

The momentum wheel is a fully capable bi-directional reaction wheel built specifically for small and medium sized spacecraft and designed for low power economical situations. For this application, the ITHACO A-Size wheel is suggested, which provides a nominal momentum of 0.375 N-m-s at approximately 500 rpm. The +6V power, as well as the regulated +5V and ±15V input power requirements, are supplied directly from the spacecraft. This wheel was developed under a NASA Small Business Innovative Research (SBIR) grant. A total of seventeen (17) A-Size wheels have been built or are currently on order.

The ADACS electronics interfaces with the spacecraft computer, both magnetometers, the three TORQRODs, and the momentum wheel. Data from the magnetometers are transferred via analog-to-digital converters to the microprocessor for evaluation in the control laws. Based upon these results, TORQROD status/polarity signals are provided to the appropriate TORQROD driver circuits on the ADACS electronics board. These driver circuits then activate the selected TORQROD, as necessary. The motor driver electronics portion is designed to maintain the wheel at a constant speed of 500 rpm.

AUTONOMOUS MAGNETIC ATTITUDE DETERMINATION

The Autonomous Magnetic Attitude Determination (AUTOMAD) technique has been developed to provide three-axis attitude information for Earth orbiting satellites, using only geomagnetic field measurement data. The two-axis magnetometers are used to measure the local geomagnetic field, the strength and orientation of which depend upon the orbital position and local orientation of the satellite. Comparisons of the measured field characteristics with a known geomagnetic field model, such as the International Geomagnetic Reference Field (IGRF), provide navigation information, which allows the determination of the satellite orbit parameters and instantaneous position. This knowledge of the spacecraft ephemeris can then be used with subsequent magnetic field measurements, which, when compared to a known geomagnetic field model, provide three-axis attitude position and rate information.

The magnetic navigation (MAGNAV) portion of AUTOMAD can provide reasonable spacecraft positional information for various orbits, both circular and elliptical, with altitudes ranging from 300 to 1000 km and with inclinations from equatorial to polar. For most scenarios, the convergence of the total positional error to less than 5 km is achieved within 10 obits. The Magnetic Attitude Determination Subsystem (MADS) can subsequently estimate three-axis attitude and attitude rates. A 1σ attitude accuracy of better than 1° for all three axes can be achieved, with initial attitude estimate errors as large as 60°. Details of both the implementation and performance of MADS and MAGNAV are provided in References 3 through 7. Approximately 2900 source lines of code are required for AUTOMAD, with an additional capacity of 128K bytes needed for data storage. Typically, an update from AUTOMAD is provided to the control algorithms at a frequency of once every ten seconds.

Once the spacecraft attitude data has been calculated, the information is used in the on-board control loops in order to maintain the proper spacecraft orientation via the activation of the TORQRODs. In addition, the navigational data can be used to autonomously command the appropriate actions to correct, modify, or alter the satellite orbit, pointing, or operational configuration, if desired.

AN EARTH HORIZON SENSOR ALTERNATIVE

As previously stated, a significant improvement in the overall ADACS performance can be accomplished if an Earth horizon sensor is used for providing spacecraft attitude information. An economical option to adding a separate sensing device, is to replace the momentum wheel with an ITHACO T-SCANWHEEL®. This device is a momentum wheel with an integral Earth sensor. The momentum wheel provides momentum bias and control torques about the pitch axis of the spacecraft. An angled scan mirror coupled to the shaft of the wheel stimulates an off-axis horizon sensor to provide pitch and roll attitude information. By using the same motor and bearings for the momentum wheel and Earth sensor, the overall power consumption is reduced and the system reliability is enhanced.

A single T-SCANWHEEL is aligned such that the spin axis is parallel to the spacecraft pitch axis. Roll and pitch error signals are provided via the sensor portion of the T-SCANWHEEL. Pitch errors are maintained via a standard wheel control loop; roll errors are controlled via the proper activation of the pitch axis TORQROD. Yaw is quarter orbit coupled to roll through the momentum bias, i.e., a yaw error becomes controllable as a roll error after the spacecraft has traversed 90° of orbital arc. Since the momentum wheel is used to provide excess momentum storage, as well as for pitch control, a momentum desaturation control law is necessary. When the wheel momentum exceeds the selected threshold, the desaturation is achieved via the two TORQRODs along the roll and yaw axes. (If pitch offsets are acceptable, than the pitch momentum can be unloaded via tach feedback control and/or a pitch bias.)

A scanning sensor provides two axis attitude information. By noting the points in the scan where the Field of View (FOV) sees space/Earth or Earth/space discontinuities and relating the scanner orientation to the spacecraft axes, pitch and roll attitude information are generated. The phase (pitch) axis is simply the average of the cross-on and cross-off points. The primary advantage of this axis is that the results are not affected by the altitude, the scan cone angle, or the height of the atmosphere (trigger height), assuming it to be the same at the two crossings. The elevation (roll) axis can be determined from the separation of the crossings or the pulse width, provided that the spacecraft altitude, scan geometry, and trigger height are all perfectly known. The elevation axis is the most difficult axis to calibrate. For the comparisons made here, a processed attitude determination accuracy of 0.3° for roll and pitch is assumed as worst case (3σ) for a single scanning Earth horizon sensor. (This accuracy is highly dependent upon the spacecraft orbit, the amount of onboard processing, and the characteristics of the particular sensor used.)

The pitch control torque is calculated as [1]

$$T_{cy} = K_{py}\theta + K_{dy}\dot{\theta}$$

where K_{py} and K_{dy} are pitch loop position and derivative gains, respectively. For roll/yaw control, the value for the pitch TORQROD dipole is determined as [1]

$$M_{cy} = -K_1 B_1 \phi - K_2 B_3 \dot{\phi} - K_3 B_3 \phi$$

The momentum desaturation dipole values (X and Z axis TORQRODs) are evaluated as [8]

$$M_d = \begin{bmatrix} M_{dx} \\ 0 \\ M_{dz} \end{bmatrix} = -K_u (\underline{\mathbf{B}} \times \underline{\Delta \mathbf{H}}) = \begin{bmatrix} K_u B_3 \Delta H_2 \\ 0 \\ -K_u B_1 \Delta H_2 \end{bmatrix}$$

where $\Delta H_2 = h2_{cur} - h2_{nom}$
$h2_{cur}$ = current wheel momentum
$h2_{nom}$ = nominal wheel momentum
K_u = desaturation gain

The net control dipole vector is thus given by

$$\underline{M} = \begin{bmatrix} 0 \\ M_{cy} \\ 0 \end{bmatrix} + \begin{bmatrix} M_{dx} \\ 0 \\ M_{dz} \end{bmatrix}$$

and the corresponding magnetic control torque vector is provided as

$$\underline{T}_{cm} = \underline{M} \times \underline{B}$$

The size, mass, and power characteristics of the required hardware components are summarized in Table 4, while the predicted ADACS performance with the T-SCANWHEEL is presented in Figures 7 through 9.

It should be noted that the degradation in the yaw performance is due to the fact that yaw is only being controlled through the passive quarter-orbit roll/yaw coupling, rather than through an active estimate of the yaw error. It may be possible to improve this situation by maintaining a modified version of AUTOMAD, at the expense of the processing requirements on the host computer. This will be evaluated in future analyses.

SOFTWARE REQUIREMENTS

Current spacecraft designs routinely incorporate an on-board computer to accommodate the control and processing functions associated with the power, thermal, telemetry/command, and attitude determination/control subsystems. The processor requirements for the ADACS scenarios discussed here are summarized in Table 5.

The ADACS software implements all of the spacecraft specific attitude determination and control algorithms and is resident within the spacecraft host computer. The ADACS Flight Program Software (AFPS) architecture suggested here is modular and composed of various Real-time Task Modules (RTM). The modular approach of selecting (versus developing) software programs from a library set of modules allows the software to be "tailored" to the ADACS hardware and to the ADACS solution, and provides lowered maintenance costs and lower recurring development costs.

Table 4: GG / Variable Bias Momentum ADACS Hardware Component Summary

Qty	Component	Unit Dimensions (cm)	Unit Mass (kg (lb))	Unit Avg/Peak Power (W)
2	Two-Axis Magnetometer	11.5 x 5.9 x 2.6	0.25 (0.55)	0.04
3	1 Am^2 TORQROD	1.3 d x 12.7	0.08 (0.18)	0.025 / 0.25
1	T-SCANWHEEL® (A-Size)	20.4 d x 19.0	3.2 (7.0)	1.0 / 5.5 *
1	ADACS Electronics Mag A/Ds / TQR Drivers Motor Driver Attitude Decoder Signal Conditioner	16.8 x 17.0 x 12.4	2.2 (4.9)	3.8
Total			6.2 (13.5)	5.0 / 10.1

* Average wheel power at 500 rpm / peak wheel power at full torque at 500 rpm

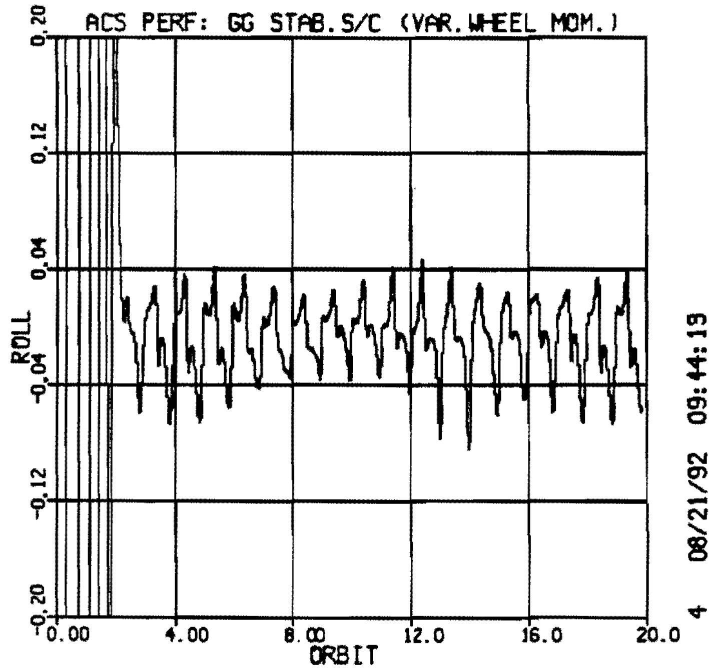

Figure 7a: Momentum Bias Stabilization With Wheel Pitch Control Roll Attitude Error (deg) vs. Orbit Number

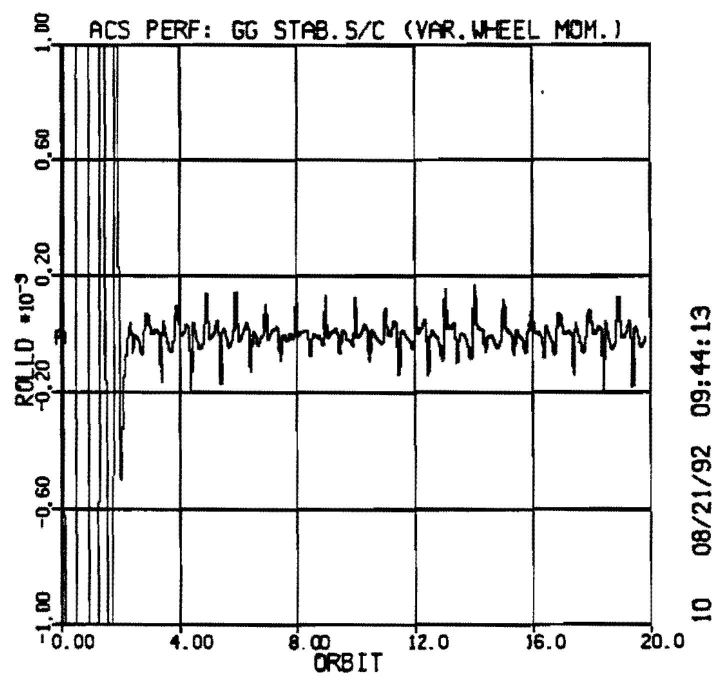

Figure 7b: Momentum Bias Stabilization With Wheel Pitch Control Roll Attitude Rate Error (deg/sec) vs. Orbit Number

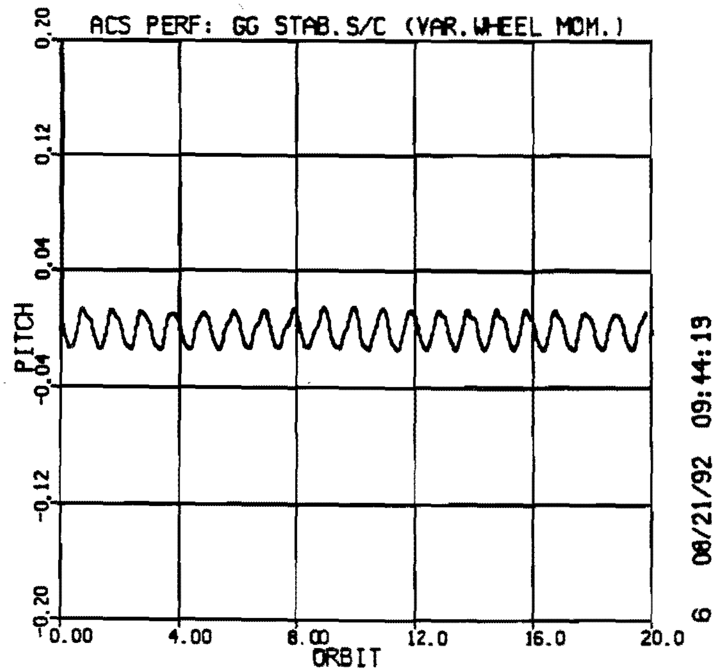

Figure 8a: **Momentum Bias Stabilization With Wheel Pitch Control Pitch Attitude Error (deg) vs. Orbit Number**

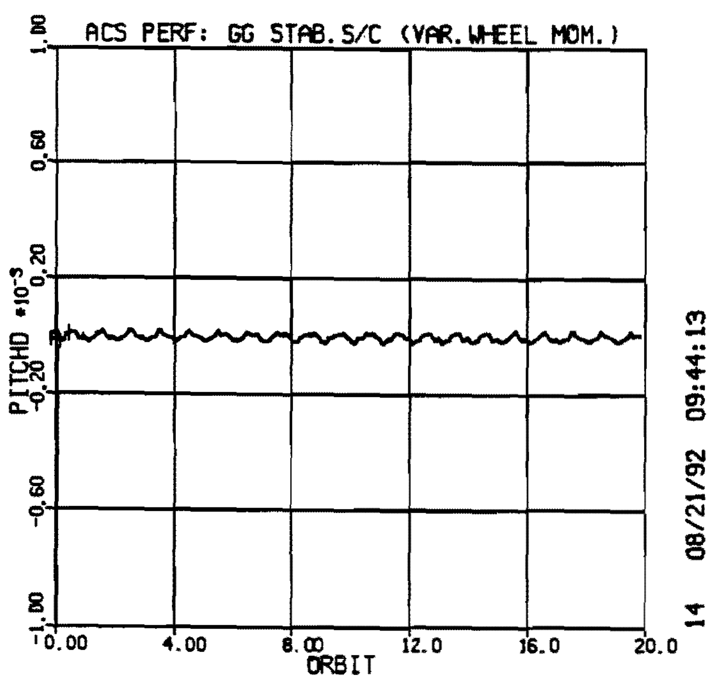

Figure 8b: **Momentum Bias Stabilization With Wheel Pitch Control Pitch Attitude Rate Error (deg/sec) vs. Orbit Number**

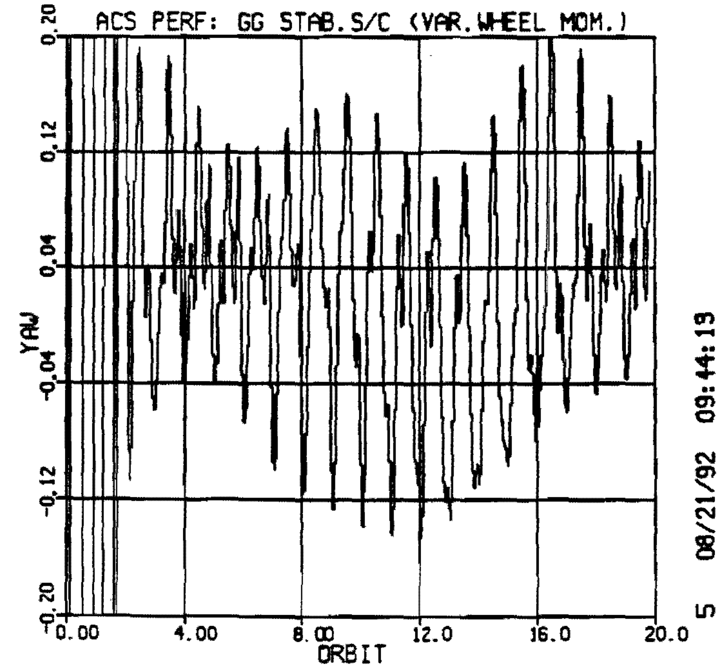

Figure 9a: Momentum Bias Stabilization With Wheel Pitch Control
Yaw Attitude Error (deg) vs. Orbit Number

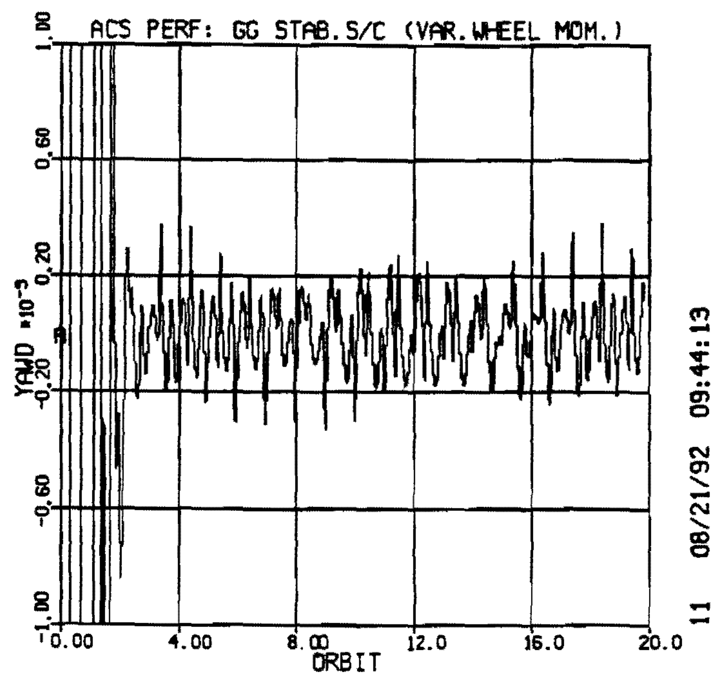

Figure 9b: Momentum Bias Stabilization With Wheel Pitch Control
Yaw Attitude Rate Error (deg/sec) vs. Orbit Number

Table 5: ADACS Software Processing Requirements

System	Source Lines of Code (SLOC)	Memory (Bytes)	Max Frequency of Operation (Hz)
Gravity Gradient Plus Magnetics	3435	217310	10.0
Fixed Bias Momentum	3535	219910	10.0
Variable Speed Bias Momentum	1415	36790	10.0

The software structure is tailored to each of the specific mission scenarios, based upon the following criteria.

1) The types and number of operational modes defined for the spacecraft
2) The hardware component subroutines based on the ADACS hardware complement defined for the mission
3) The subroutines required to perform the attitude determination algorithms and control laws defined for the mission type
4) The subroutines that provide interrupt level communication

The AFPS is structured with a logical hierarchy of drivers, subroutines, and data, whose operation is coordinated by the Operating System (OS).

The concept of building the flight software from a number of generic modules has several advantages. The maximum heritage is preserved from one spacecraft ADACS configuration to the next, thus eliminating much of the need for recurring development and qualification costs, as well as reducing risk. The maintainability is enhanced since the software changes little from mission to mission and the reliability is also enhanced for the same reason. Finally, configuration control is simplified as the need for new documentation and flowcharts is minimized. (This preliminary estimate is based upon the use of a LYNX OS incorporated into an 80386 processor with an 80387 coprocessor. The final OS has not yet been selected.)

ADACS PERFORMANCE SUMMARY

The overall ADACS mass, power, and performance summary is provided in Table 6. The attitude control errors are the peak steady state errors estimated from Figures 1 through 9. The attitude determination or knowledge errors for the MADS concept are assumed to be no worse than those observed from the Low-power Atmospheric Compensation Experiment (LACE) spacecraft [5]. The attitude determination errors for the Earth sensing scenario are based upon values observed from the LANDSAT spacecraft missions. The overall attitude accuracy (3σ) is calculated as the RSS of the control and the knowledge errors.

An extremely low mass and low power consuming ADACS is provided through the use of magnetometers, TORQRODs, and the spacecraft computer, if the attitude performance is acceptable. However, as shown in Table 6, a significant improvement in the three-axis control of the spacecraft can be achieved by incorporating the bias momentum wheel.

Based upon the performance of the LACE spacecraft, magnetic attitude determination to better than 3.0° per axis is readily achievable. Since this method was not the primary means of attitude determination for LACE, the spacecraft was not optimized for this technique. Several additional requirements concerning such items as magnetometer placement, spacecraft residual magnetism, data time tagging, etc., should be implemented on a spacecraft which intends to

Table 6: ADACS Performance Summary

System	Mass (kg)	Power[¤] (W)	Axis	Attitude Errors (deg) Control*	Knowledge†	Accuracy§
Gravity Gradient Plus Magnetics	1.7	0.9 / 1.6	Roll	2.8	1.8	3.3
			Pitch	1.9	2.4	3.1
			Yaw	5.0	2.8	5.7
Fixed Bias Momentum	4.6	2.2 / 2.8	Roll	0.16	1.8	1.8
			Pitch	0.20	2.4	2.4
			Yaw	0.12	2.8	2.8
Earth Sensor Plus Variable Speed Bias Momentum	6.2	5.0 / 10.1	Roll	0.08	0.3	0.31
			Pitch	0.02	0.3	0.30
			Yaw	0.2	0.6	0.63

¤ Average / Peak
† 3σ accuracy from flight data
* With perfect attitude knowledge
§ RSS Control and Knowledge

use MADS for primary data acquisition. These enhancements are expected to reduce the attitude determination errors by at least a factor of two, with a similar improvement in the overall attitude accuracy.

If an Earth horizon sensor is added, a typical momentum biased ADACS is achieved. As shown in Table 6, the overall system performance is significantly improved, at the expense of increased mass, power, and cost requirements.

REFERENCES

[1] Wertz, J.R. ed., Spacecraft Attitude Determination and Control, D. Reidel Pub. Co., (Boston, 1978).

[2] F. Martel, P. K. Pal, and M. Psiaki, "Active Magnetic Control System For Gravity Gradient Stabilized Spacecraft", Second Annual AIAA/Utah State University Conference On Small Satellites, September 1988

[3] M. Psiaki, F. Martel, P. Pal, "Three-Axis Attitude Determination via Kalman Filtering of Magnetometer Data", Journal of Guidance, Control, and Dynamics, Volume 13, Number 3, May-June 1990

[4] M. Psiaki and F. Martel, "Autonomous Magnetic Navigation for Earth Orbiting Spacecraft", Third Annual AIAA/Utah State University Conference On Small Satellites, September 27, 1989

[5] S. M. Fox, P. K. Pal, and M. Psiaki, "Magnetometer-Based Autonomous Satellite Navigation (MAGNAV)", 13th Annual AAS Guidance and Control Conference, February 3-7, 1990

[6] S. M. Fox, P. K. Pal, M. L. Psiaki, and M. E. Mook, "Magnetic Attitude Determination Subsystem Performance on the LACE Spacecraft", 14th Annual AAS Guidance and Control Conference, February 2-6, 1991

[7] M. L. Psiaki, L. Huang, and S. M. Fox, "Ground Tests of Magnetometer-Based Autonomous Navigation For Low Earth Orbiting Spacecraft", AIAA Guidance, Navigation, and Control Conference, August 1991

[8] A. Craig Stickler and K. T. Alfriend, "An Elementary Magnetic Attitude Control System", AIAA Mechanics and Control Conference, August 5-9, 1974