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VOLUME I: TECHNICAL SUMMARY

by P. C. Wheeler, R. G. Nishinaga, J. G. Zaremba, and H. L. Williams

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TRW SYSTEMS
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ABSTRACT

This study is directed toward establishing the feasibility of utilizing a Semi-Active Gravity Gradient System (SAGS) for controlling the attitude of an earth-oriented spacecraft. The control configuration employs an active reaction wheel for pitch attitude control. Roll/yaw control is achieved by operating the pitch wheel with a momentum bias, and by gimballing the wheel and coupling it to the vehicle through an energy removal mechanism to provide roll/yaw damping. Long-term momentum buildup is prevented by gravity gradient restoring torques.

The vehicle configuration assumed for this study is that of a scaled down (smaller and lighter) Nimbus. A major departure from the Nimbus structural configuration is the probable presence of a two-degree-of-freedom solar array. The additional array freedom compensates for the satellite's lack of yaw maneuverability. A single inertia mast is included for increased gravity gradient restoring torques.

These investigations have dealt with both the performance analysis and implementation aspects of the SAGS control configuration. The results of the former study phase indicate that steady-state roll/yaw accuracies on the order of one to two degrees are readily attainable with this concept, while pitch accuracy levels of one-half to one degree present no difficulties for the nominal mission and spacecraft here considered. (In this regard it should be stressed that these results are strongly related to the spacecraft configuration and the resulting thermal boom bending and magnetic disturbances;

over

on a smaller, magnetically "cleaner" vehicle accuracies of a few tenths of a degree in all axes would be a reasonable design goal.) The controller parameter values selected for fine control are completely acceptable for acquisition.

Implementation studies have resulted in two preliminary mechanical designs, both of which incorporate all mechanical functions required for attitude control (i.e., horizon sensing and control torque generation). These designs differ primarily in the mechanization of the horizon sensing system. Indications are that a control system weight as low as 25 pounds (including signal processing and control electronics, but not the solar array control system or the inertia augmentation assembly) can be achieved, with a nominal power consumption of 14 watts.

NASA CR-593 summarizes the results of these investigations; detailed analyses are presented in NASA CR-594.

AUTHOR

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1.0 INTRODUCTION

Many current and projected satellite applications require the alignment of one axis of the spacecraft with local vertical to relatively high degree of precision. In some cases sufficient accuracy (a few degrees) may be obtained using a totally passive gravity gradient system with appropriate energy dissipation devices. When extreme accuracy is required, a totally active system, including precision sensors, reaction wheels and pneumatic jets, may be indicated. Of course, the complexity of such an attitude control system can impose severe weight, power, and reliability penalties. The Semi-Active Gravity Gradient System (SAGS) which is the subject of these investigations offers a potential compromise (in terms of both performance and complexity) between the active and passive approaches. Indeed, for low altitude earth-pointing applications, the accuracy potential of the SAGS control concept is competitive with that offered by much more intricate fully active control systems using horizon sensors.

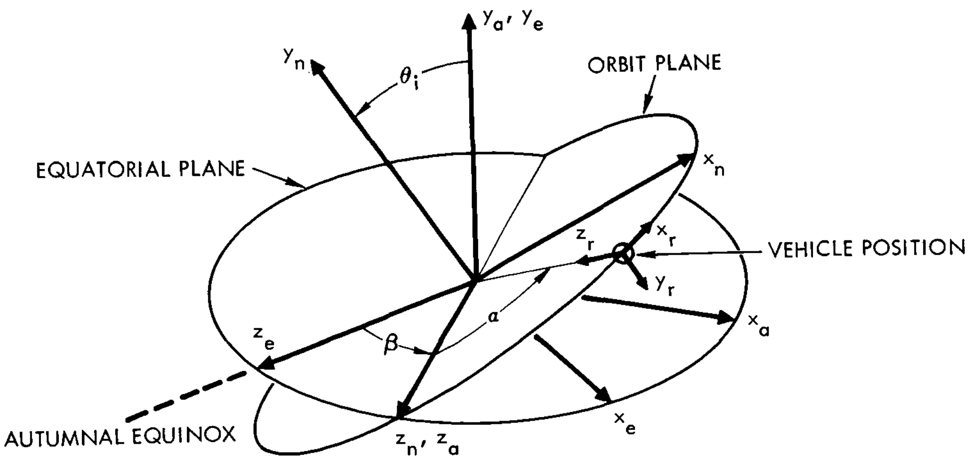

Figure 1 Definition of Nominal Spacecraft Attitude

1.1 The Control Problem

The nominal attitude requirement considered for this study is that the vehicle control axes be aligned with the (x_r, y_r, z_r) orbital reference frame of Figure 1; that is, the yaw axis of the spacecraft is to be aligned toward the earth and the pitch axis must be normal to the orbit plane. Associated with this requirement are two distinct control problems:

(i) Nominal attitude orientation must be acquired from the potentially large attitude errors and angular velocities existing immediately after orbital injection.

(ii) The required attitude accuracy must be maintained for the duration of the mission in the face of environmental and internal disturbances

In many applications an additional control requirement is introduced by the ground rule that a solar array-storage battery power supply be considered; namely, the face of the solar array must be maintained normal to the vehicle-sun line. Although the control problems associated with the solar array are straightforward, the motion of the array may introduce significant internal disturbances.

1.2 The SAGS Control Configuration

The basic SAGS control configuration consists of a single reaction wheel with its spin axis (nominally) along the pitch axis of the vehicle. This wheel is mounted in a single (roll) degree of freedom suspension which allows its spin axis to move in the pitch/yaw plane of the spacecraft. Gimbal motion is constrained by a spring restraint, a damping mechanism, and a set of stops.

For small error control the wheel speed may be characterized by small (bounded) perturbations about a bias level. The presence of a pitch bias momentum provides stiffness for roll/yaw attitude motions. Roll/yaw perturbations result in motion of the vehicle relative to the wheel; this motion produces damping through the dissipation mechanism in the gimbal system.

Active pitch control is provided by controlling the wheel speed about its bias level. A horizon scanner provides pitch attitude error information. Momentum build-up is prevented by gravity gradient torques.

Efficient solar energy conversion can be assured by controlling the attitude of the solar panels relative to the vehicle. The array drive will provide continuous motion of the array about the drive shaft in response to error signals provided by array-mounted sun sensors. Because the sun may be out of the orbit plane, a second degree of solar array freedom may be necessary. This can be provided by a hinge in the solar array drive shaft. Motion of the sun relative to the orbit plane will necessitate occasional (e.g., weekly) incremental changes in the hinge angle; this control can be effected by ground command.

1.3 Mechanization of the SAGS Configuration

The primary components of the SAGS control configuration are the gimballed reaction wheel assembly, an inertia augmentation assembly, a horizon scanner for obtaining attitude error information, and the electronics associated with generating meaningful attitude data from the scanner output and using it, together with tachometer signals, to control the reaction wheel speed.

From the viewpoint of evaluating this control concept, the gimballed reaction wheel assembly is of greatest significance since it alone is unique to the SAGS configuration. Major elements of this assembly include the motor/reaction wheel unit, the suspension (i.e, a pair of torsion wires along the axis of freedom), a damping mechanism (employing the effects of magnetic hysteresis, eddy currents, or fluid viscosity) and a case in which the reaction wheel/ motor unit is suspended via the torsion wires.

Although the horizon scanner may well be a separate unit, it is possible to incorporate this function into the gimballed reaction wheel assembly. In this case, the speed of the wheel (which consists of small perturbations about a bias level) provides the scanning action required for horizon sensing. This approach yields a single mechanical unit (the Gimballed Reaction Wheel/Scanner) which provides the entire attitude control function.

An inertia augmentation assembly is generally required to provide significant gravity gradient restoring torques for momentum control. The basic element in this unit will be an extensible mast (e.g., a deHavilland boom) with a mass at its remote end; one or more of these sub-assemblies may be employed, generally extending along the yaw axis of the spacecraft. In some cases it may be desirable to add to the flexibility of the system by providing the capability of extending or retracting the mast and/or altering its angle of departure from the spacecraft via ground commands. Of course the benefits of such provisions must be traded against the additional system complexity which they imply.

1.4 Interface Considerations

The extent to which the attitude control assembly constrains other spacecraft sub-systems, either by interfering with their normal operation or by requiring increases in their level of performance, is of considerable importance. Of equal significance is the degree to which the other sub-systems might limit the applicability of this control configuration.

Of the major spacecraft sub-systems, only the communication system, the power system, and the structure interact significantly with the SAGS control assembly; the first of these is of concern only to the extent that it limits the command (and telemetry) capability while the major power system interface is the possible need for a two-degree-of-freedom solar array. The most significant interface is with the spacecraft structure.

The ACS/structure interface concerns primarily the horizon scanning function (which requires a certain unobstructed field of view) and the inertia augmentation assembly. In the latter case the major requirement is that the mast not interfere with the mission of the spacecraft (as an example, a mast extending below the sensory ring on the Nimbus spacecraft would be intolerable); in addition, the mast mounting point is generally constrained.

Horizon scanner field of view considerations can have a significant impact upon the ACS. For example, if a scan cone with a full 360 degree FOV is not feasible, a scanner output processing scheme must be employed which will discard a significant position of each scan cycle without degrading ACS performance. This factor is even more significant if the scanner is included in the reaction wheel, because the position of the scan cone relative to the vehicle will be a function of the gimbal deflection angle. The extent to which this might be a problem depends strongly upon the satellite configuration and where in the vehicle the scanner assembly is mounted. In the case of an integrated wheel/scanner assembly the effect of gimbal motion can be reduced greatly by using the scanner output only when the gimbal deflections are small (i.e., when the roll and yaw attitude errors are small). The results of this study indicate that the scanner field-of-view factor places no significant limitation upon the applicability of the SAGS control concept.

1.5 Study Program

The major elements of the SAGS feasibility evaluation study are related to the following three tasks:

- Task I: Fine Control Performance Evaluations
- Task II: Acquisition Performance Evaluations
- Task III: Implementation Evaluations

The fine control performance studies consist primarily of evaluating the merits of various configuration alternatives (e.g., one versus two degrees of freedom in the wheel suspension), selection of such parameters as the wheel bias momentum to give desirable performance, determination of the effects of disturbances upon the control accuracy, and, finally, an evaluation of the absolute accuracy attainable with the SAGS control configuration for a representative application.

The acquisition studies are directed toward establishing a plausible acquisition sequence and verifying its performance via analyses and simulation studies.

The implementation evaluations concern the effect of the mechanization requirements of the SAGS control concept upon its over-all feasibility. Topics of this study include the wheel suspension, the reaction wheel/motor assembly, damping mechanisms and earth sensing capabilities.

In order to evolve significant numerical results (e.g., estimates of absolute control accuracy and of control system weight), a specific vehicle configuration and mission have been considered. The vehicle structure (Figure 2) is similar to Nimbus, but of smaller dimensions and with an over-all weight of 500 lbs. Major elements include a Nimbus-type sensory ring (R), a control box (C) housing the solar array drive and attitude control system components, an inertia mast (M) with a tip mass to enhance the gravity gradient restoring torques and two solar panels (P) which are connected to

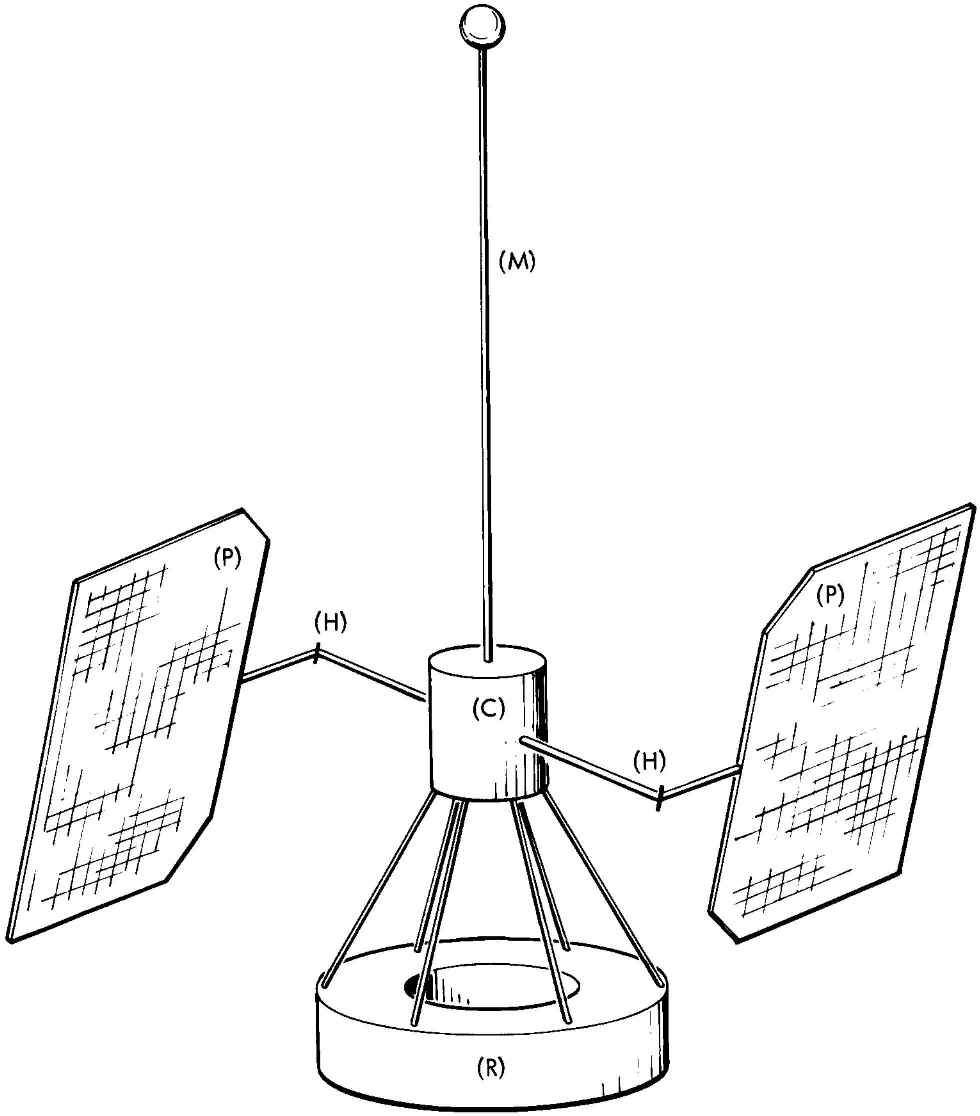

Figure 2 Typical SAGS Structural Configuration

the solar array drive shaft through hinges (H). A detailed evaluation of this configuration is presented in the SAGS First Quarterly Report (Reference 1). The physical dimensions and inertia properties associated with Figure 2 are presented in the next section of this report. The orbit assumed for these studies is circular with an altitude of 750 nautical miles.

2.0 FINE CONTROL PERFORMANCE EVALUATIONS

The lifetime of a spacecraft is typically composed of a short period of time (a day or less) during which the nominal attitude is initially acquired and a much longer period during which the attitude control system must maintain the required control accuracy in the presence of environmental and internal perturbing influence. For this reason, the system design must emphasize fine control performance rather than attempting to "optimize" acquisition performance. The design approach taken in the investigations here reported has been to evolve a system based upon achieving desirable fine control performance (within realistic mechanization constraints) and then to test the acquisition performance and, if necessary, alter the design to achieve acceptable acquisition operation. These investigations have indicated that a system selected on the basis of steady-state performance is also a good one from the viewpoint of acquisition.

At the outset of this study a major ACS option was the choice of a suspension configuration. The choice was between a system with both roll and yaw gimbal freedom, or one with either roll freedom or yaw freedom. A preliminary analysis shows that the single (roll) degree-of-freedom gives the most favorable fine control performance. Subsequent investigations were, therefore, limited to the roll SDF suspension.

The major design parameters for the SAGS control configuration are the amount of damping and spring restraint in the gimbal, the momentum capacity of the reaction wheel, (i.e., the bias momentum and the momentum range), the motor torque level, the method of best utilizing the horizon scanner output, and the degree of inertia augmentation. Of these, the motor torque, the momentum range and the scanner processing scheme have been decided by pitch capture considerations (Section 3); the remaining parameters are determined by the requirements of roll/yaw fine control.

The numerical evaluations of this section (e.g., disturbance torque computations) are based upon the vehicle structural dimensions of Figure 3 (see Reference 1 for development of this configuration) and an orbital altitude of 750 nautical miles. The boom length, L, has been selected (in conjunction with the tip mass) on the basis of allowing a steady pitch offset of no more than 0.5° in the presence of a constant disturbance of 3×10^{-5} ft-lb.

The resulting 52 foot mast with a 15 lb. tip mass yields the deployed inertia distribution shown in Table I. Note that it will generally be possible to extend the rod further (or retract it) to obtain a more favorable inertia distribution if in-orbit performance indicates such a requirement.

	Boom Undeployed	Boom Deployed*
I_x (roll)	130	1500
I_y (pitch)	100	1500
I_z (yaw)	80	100

* Rounded to nearest 50 slug-ft^2 increment

Table I. Spacecraft Inertia Properties
($L = 52$ ft, $M_T = 0.47$ slugs)

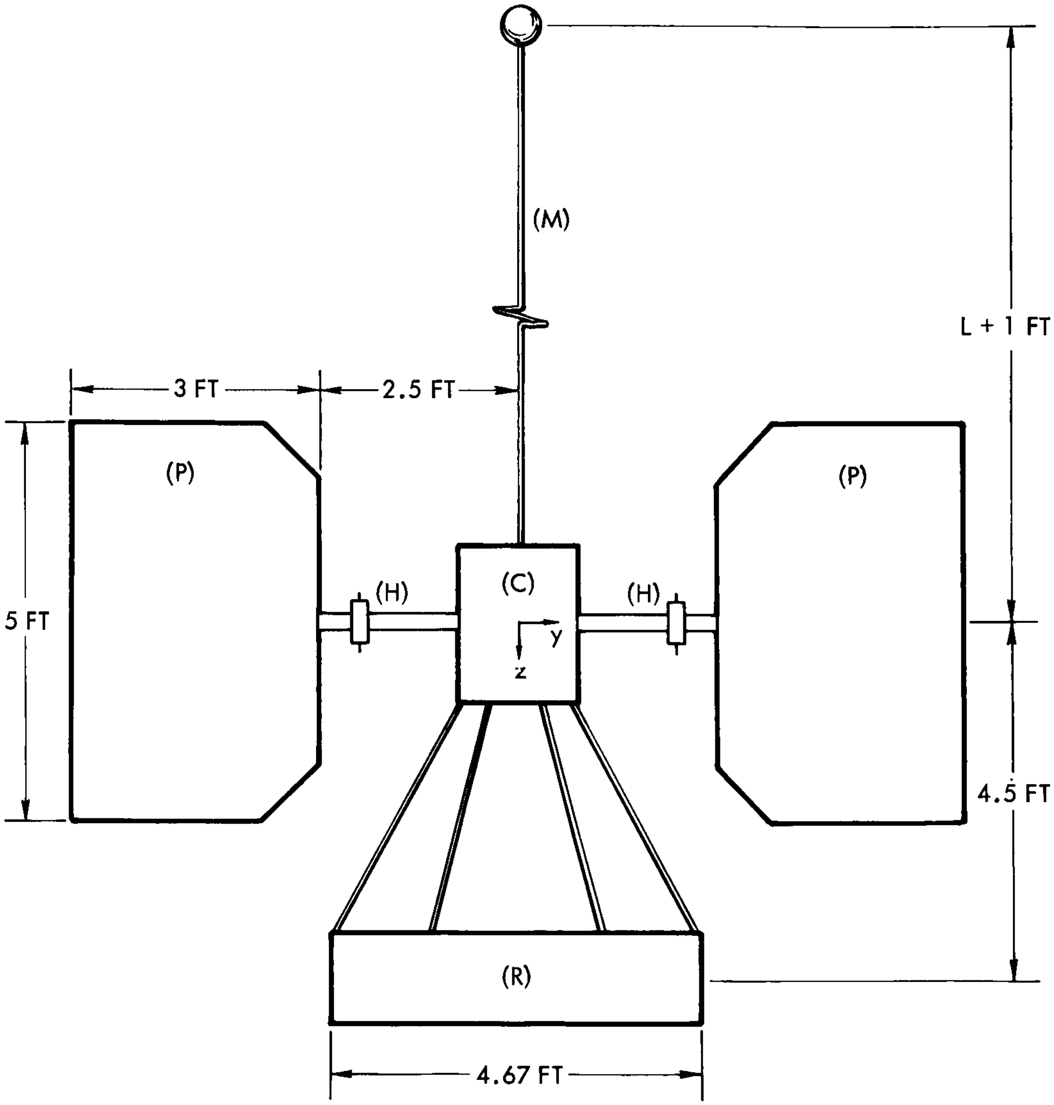

Figure 3 Vehicle Dimensions for Numerical Evaluations

2.1 Pitch Fine Control

For small attitude errors pitch motion is decoupled from roll/yaw operation.[1] Pitch performance, therefore, does not depend upon the values chosen for the suspension parameters or upon the bias momentum of the pitch wheel. In fact the most fundamental question in designing the wheel control loop is the selection of the compensation loop.

Figure 4 shows the pitch control loop block diagram. Major components include the reaction wheel motor (which has an essentially "flat" torque characteristic over its momentum range), a pulse ratio modulator which furnishes an on-off drive voltage to the motor, and a tachometer used both for compensation and to inhibit the momentum range of the wheel under transient (acquisition) conditions. The pulse ratio modulator, here represented by its slow-signal input/output characteristic, is shown in more detail in Figure III-2 of Appendix III. Notice that the effects of boom bending are neglected. This assumption can generally be made valid by an appropriate choice of the boom properties (i.e., diameter and wall thickness); in particular the boom natural frequency must not be near the motor torquing frequency required to maintain the bias speed in the presence of windage.

[1]Possible sources of coupling include dynamical effects as well as gimbal motion in the case of the integrated wheel/scanner assembly. However, these effects are all second-order.

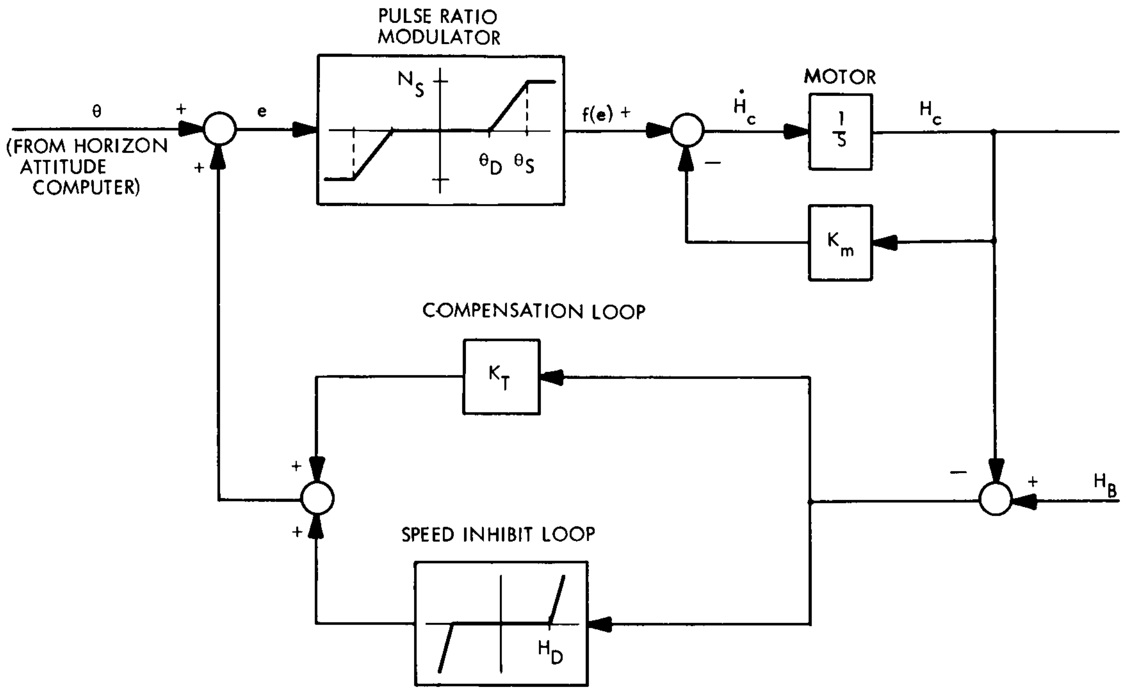

Figure 4 Reaction Wheel Control System

The reaction wheel control system of Figure 4 must provide pitch attitude control while maintaining the wheel speed within a small neighborhood of the nominal bias speed. These functions must be performed in the presence of environmental disturbances which cannot be precisely estimated; thus, the design evolved must be one which will function (perhaps with somewhat reduced attitude accuracy) in the presence of abnormal perturbing effects.

Consider the operation of the pitch control channel, without the compensation loop, in the presence of an excessive steady disturbance, $T_{dy}(0)$. Such a perturbation will produce a steady attitude offset given by

$$\Theta(0) = \frac{T_{dy}(0)}{3\,\omega_o^2\,(I_x - I_z)} \tag{2.1}$$

If, as well may be the case, $\Theta(0)$ is in excess of the modulator deadband, the wheel will be accelerated until the deviation of the wheel momentum from its bias level exceeds the allowed upper limit. At this point, the effectiveness of pitch control (for example, in reacting to periodic pitch disturbances) will be seriously impaired.

The presence of the compensation network eliminates the degradation in pitch performance noted above. By adding to the error signal a term proportional to the incremental wheel momentum, steady-state operation can be reached without excessive speed excursions. The system is converted from one in which $\dot{H}_c$ is proportional to Θ to one in which $H_c - H_B$ is proportional to Θ. Moreover, the damping provided by the compensation loop eliminates the need for a lead network in the error signal feedback path. Again it should be noted that the effects of boom bending have been neglected; Nimbus D design studies have indicated that these effects may necessitate the inclusion of attitude rate feedback (i.e., a lead network).

The parameters to be specified are the modulator deadband (Θ_D), the modulator saturation level (Θ_S), the motor torque level (N_S), the momentum range of the wheel (H_D), and the tachometer feedback gain (K_T). Of these only Θ_D, Θ_S and K_T are of concern here, because the values of N_S and H_D recommended for pitch capture (5 in-oz and

1 ft-lb-sec, respectively) far exceed the requirements of fine control.

Owing to the effects of the ever present windage torque $K_m H_c$, the static operating point of the system will always be in the region $\theta_D < |e| < \theta_S$. As a result, the control system will, in the absence of disturbances, hold θ equal to zero instead of allowing the limit cycle motion which would occur in the absence of biased wheel operation. Although this performance will occur in spite of the value chosen for θ_D, it is still desirable to select values of θ_D which are comparable to the required attitude accuracy. Reasonable values for θ_D and θ_S in the present instance are 0.5 degree and 1.0 degree, respectively.

The considerations associated with specifying K_T are developed in Appendix III, and summarized in Table II. There is clearly a trade-off between the steady-state attitude response to orbit-frequency

Response Characteristic	Effect of Increasing K_T
θ (0)	unaffected
θ $(j\omega_o)$	increases
$H_c(0)-H_B$	decreases
H_c $(j\omega_o)$	decreases

Table II Effect of Increasing K_T Upon Steady-State Pitch Performance disturbances and pitch wheel excursions. (Figure 5 indicates more specifically the effect of K_T upon control accuracy for the inertia properties of Table I and an orbital rate of 10^{-3} rad/sec.) A tachometer gain of 0.1 rad/ft-lb-sec was found to be a good compromise in this study. With disturbance torques of $T_{dy}(0) = 3 \times 10^{-5}$ ft-lb and $T_{dy}(j\omega_0) = 4 \times 10^{-5}$ ft-lb the error responses are $\theta(0) = 0.50^\circ$ and $\theta(j\omega_0) = 0.26^\circ$. This performance, predicted via linear analysis, was verified by analog simulation. Subsequent to these simulation studies a detailed disturbance torque analysis gave estimates of $T_{dy}(0) = 1.6 \times 10^{-5}$ ft-lb and $T_{dy}(j\omega_0) = 1.2 \times 10^{-4}$ ft-lb; the corresponding performance levels are $\theta(0) = 0.27^\circ$ and $\theta(j\omega_0) = 0.78^\circ$, for a peak pitch attitude error of 1.05°.[2]

It is of interest that the results presented in Appendix III show that the significant response characteristics (Table II) are, for sufficient wheel torque levels, independent of the modulator characteristic. This phenomenon, observed for small error transient response as well, occurs because the modulator acts through the motor to maintain the error signal, e, just outside the deadband (by a distance such that the average motor drive torque equals the windage and friction). This portion of the system can then be regarded as a high-gain amplifier and the precise shape and gain of its input/output

[2]The largest contribution to this estimate of $T_{dy}(j\omega_0)$ occurs as a result of thermal boom bending with the sun in the orbit plane (Appendices IV and VI). This effect can be reduced significantly (by an order of magnitude) by employing a coated boom.

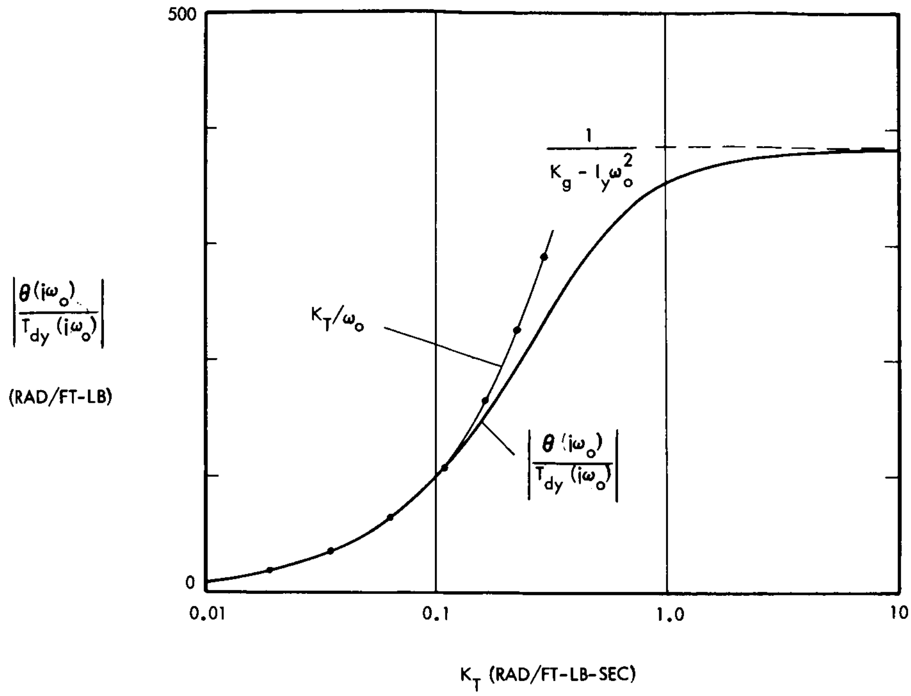

Figure 5 Effect of Tachometer Gain upon Pitch Attitude Error (ω_o = 0.001 rad/sec)

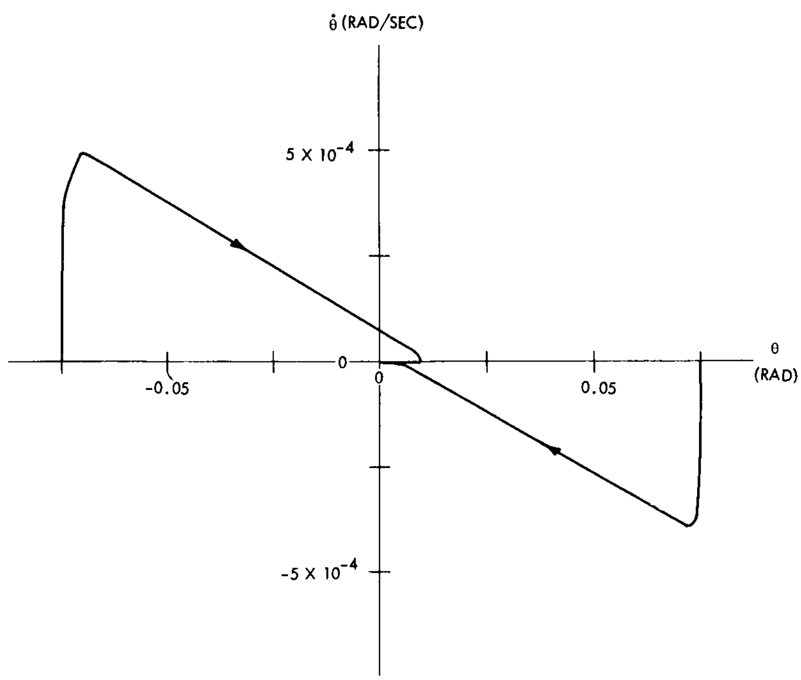

Figure 6 Pitch Transient Response

characteristic is of little moment.

The transient response of this system is shown in the phase trajectory of Figure 6. Note that the terminal response is composed of a short-lived, high amplitude exponential and a long-term low amplitude exponential. This result is demonstrated analytically, as well, in Appendix III.

Another compensation scheme, involving integral tachometer feedback, was investigated via analog simulation. To mechanize this method of compensation the tachometer gain of Figure 4 is replaced by a lag network with a time constant on the order of 10^4 seconds (Reference 2). This system was found to give superior attenuation of the response to orbit rate disturbances; however, it gave adequate damping of small attitude transient motions only with a lead network in the path from the horizon attitude computer to the modulator. It appears that the additional complexity is warranted only in cases when the orbit rate disturbance is significantly greater than the values predicted above (as might be the case in the presence of large values of orbital eccentricity - - for example $\epsilon = .10$). Even in these cases the proportional tachometer compensation scheme can be made to provide an adequate degree of attenuation of orbit frequency disturbances (by reducing K_T) while still providing acceptable bounds upon wheel momentum variations.

2.2 Roll/Yaw Fine Control

2.2.1 Suspension Configuration

The primary criterion in selecting those parameters which affect roll/yaw performance is the maintenance of a high degree of steady-state accuracy in the presence of the expected environmental disturbances. On this basis, the most promising suspension configuration must be selected and the reaction wheel bias momentum must be chosen.

The wheel may be suspended in any one of four possible geometrical configurations (Appendix IV). These possibilities include two two-degree-of-freedom suspensions (differing significantly for large errors only, as a function of the ordering of the roll and yaw gimbals) and two single-degree-of-freedom configurations (one with a roll gimbal and one with a yaw gimbal). A preliminary analysis (Appendix IV) indicated that d.c. yaw accuracy is extremely sensitive to k_z, the yaw gimbal spring restraint, for any system possessing a yaw degree of freedom; large values of k_z are required for tight yaw control and, in fact, yaw accuracy in the presence of constant disturbances in best for an infinite spring (i.e., no yaw degree of freedom). On the other hand, the d.c. yaw response with a single (roll) gimbal is not a function of k_x, the roll spring constant. The response to disturbances at orbital frequency is relatively unaffected by the gimbal configuration. Thus, from a performance viewpoint (and certainly from the standpoint of mechanization) a suspension with a roll gimbal (i.e., one in which the wheel momentum is always in the pitch-yaw plane of the vehicle) is preferred. Subsequent studies have dealt exclusively with this configuration.

The technique utilized to damp motion of the reaction wheel assembly relative to the vehicle is of considerable interest. For velocity-dependent dampers (e.g., an eddy current damper), the system's steady-state performance can be established conclusively via linear analyses such as those herein reported. Hysteresis dampers, on the other hand, require that the system be studied via simulation. In attempting to do so (using the existing Generalized Spacecraft Simulation) it was found that the presence of the high frequency dynamics associated with the small gimbal inertias resulted in extremely inefficient computer operation (i.e., computation at speeds on the order of real time). Omission of the gimbal inertias (as was done successfully with a proportional damper in the acquisition simulation) eliminates the gimbal differential equation altogether, since the remaining gimbal torques (those due the damper, the spring restraint and the presence of the wheel momentum) are functions which depend only upon the gimbal position and (in the case of the damper) upon the sign of the gimbal rate. It appears that this problem could be resolved by solving iteratively for the gimbal deflection using the reduced gimbal equation; however, time did not allow adaptation of this approach to the Generalized Spacecraft Simulation in the present study period. In spite of these analytic and simulation difficulties the design of a suitable hysteresis damper can easily be effected based upon TRW experience gained for passive systems. Therefore, although no concrete data is available for SAGS roll/yaw performance with a hysteresis damper, the implementation of this approach can be undertaken with a high level of confidence. Indeed, past experience has indicated performance advantages for hysteresis damper systems. Furthermore, use of a hysteresis damper rather than an eddy current mechanism can yield a reduction of several pounds in the weight of the control system.

2.2.2 Stability Considerations

In establishing the values of H_c (the bias momentum), k (the gimbal spring restraint) and c (the gimbal damping coefficient), certain basic constraints are imposed by the requirement that the nominal orientation be stable in the absence of external disturbances. An analysis reported in depth in the First Quarterly Report (Reference 1) imposed the following constraints on the roll gimbal suspension:

$$
\begin{aligned}
&H_c < 4\,\omega_o\,(I_y - I_z) \\
&H_c < \omega_o\,(I_y - I_z) \\
&H_c < \frac{4\,k\,\omega_o\,(I_y - I_z)}{k + 4\omega_o^2(I_y - I_z)} \\
&c > 0
\end{aligned}
\qquad (2.2)
$$

where gimbal inertias are neglected. With equal roll and pitch inertias, the wheel momentum is clearly required to be negative, corresponding to the case in which the wheel momentum adds vectorially to the orbital angular momentum of the spacecraft. Observe that the equilibrium orientation which occurs with a yaw error of 180° is made unstable with H_c sufficiently negative. In practice the stability margin of this system is quite high, since values of H_c on the order of $\omega_o I_y$ are required for acceptable yaw performance.

2.2.3 Parameter Selection

An extensive study was undertaken on the TRW "On-Line" computer to determine the effect of the various system parameters (H_c, k, c) upon the sensitivity of the roll and yaw errors to roll/yaw disturbances at frequencies ranging from $\omega = 0$ to $\omega = 4\,\omega_o$. This simplified study, reported in detail in Reference 1, established the following recommended parameter ranges:

$$\begin{aligned} &\omega_o I_y \le c \le 2\,\omega_o I_y \\ &0 < k \le .5\omega_o^2\, I_y \\ &2\omega_o I_y \le H_c \le 4\omega_o I_y \end{aligned} \qquad (2.3)$$

Additional constraints are imposed by implementation considerations; for example, choosing the lower limits of (2.3) for the wheel momentum and the damping coefficient yields a considerable weight dividend (Appendix VII). Considering these factors the following parameter values were selected for subsequent detailed performance investigations:

$$\begin{aligned} I_y &= 1500 \text{ slug-ft}^2 \\ c &= 1.5 \text{ ft-lb per rad/sec} \\ k &= 0.75 \times 10^{-3} \text{ ft-lb/rad} \\ H_c &= -3.0 \text{ ft-lb-sec} \end{aligned} \qquad (2.4)$$

where an orbit rate of 10^{-3} rad/sec is assumed.

2.2.4 Detailed Performance Evaluation

The terminal phase of the roll/yaw design study has involved a detailed assessment of the absolute performance capabilities of the SAGS control configuration for a representative spacecraft (Figure 2) in a typical orbit (Appendix IV). Utilizing a detailed digital program developed for roll/yaw frequency response evaluations (including the effects of such factors as products of inertia, the displacement of the wheel assembly from the vehicle center of mass, and gimbal inertias), the error/torque influence coefficients - e.g., $|\phi(j\omega)/T_{dx}(j\omega)|$ - were computed and plotted as functions of frequency (as, for example, in Figure 7).

In order to evaluate the pointing accuracies, a detailed disturbance torque analysis has been completed (Appendix VI). Effects considered are magnetic moments, thermal bending in the inertia mast, solar pressure, eccentricity perturbations and misalignments between the principal axes of inertia and the control axes. Two extreme cases, one with the sun in the orbit plane and one with the sun normal to the orbit plane, were considered, with the following additional assumptions:

- o The residual spacecraft magnetic moment is no greater than 5×10^{-5} ft-lb-gauss (7.3 amp-ft^2) along any axis of the vehicle or the array.

- o The inertia mast is of uncoated Berylium-Copper with a length of 52 ft, a diameter of 0.5 in, a thickness of 0.002 in, and a 15 lb tip mass.

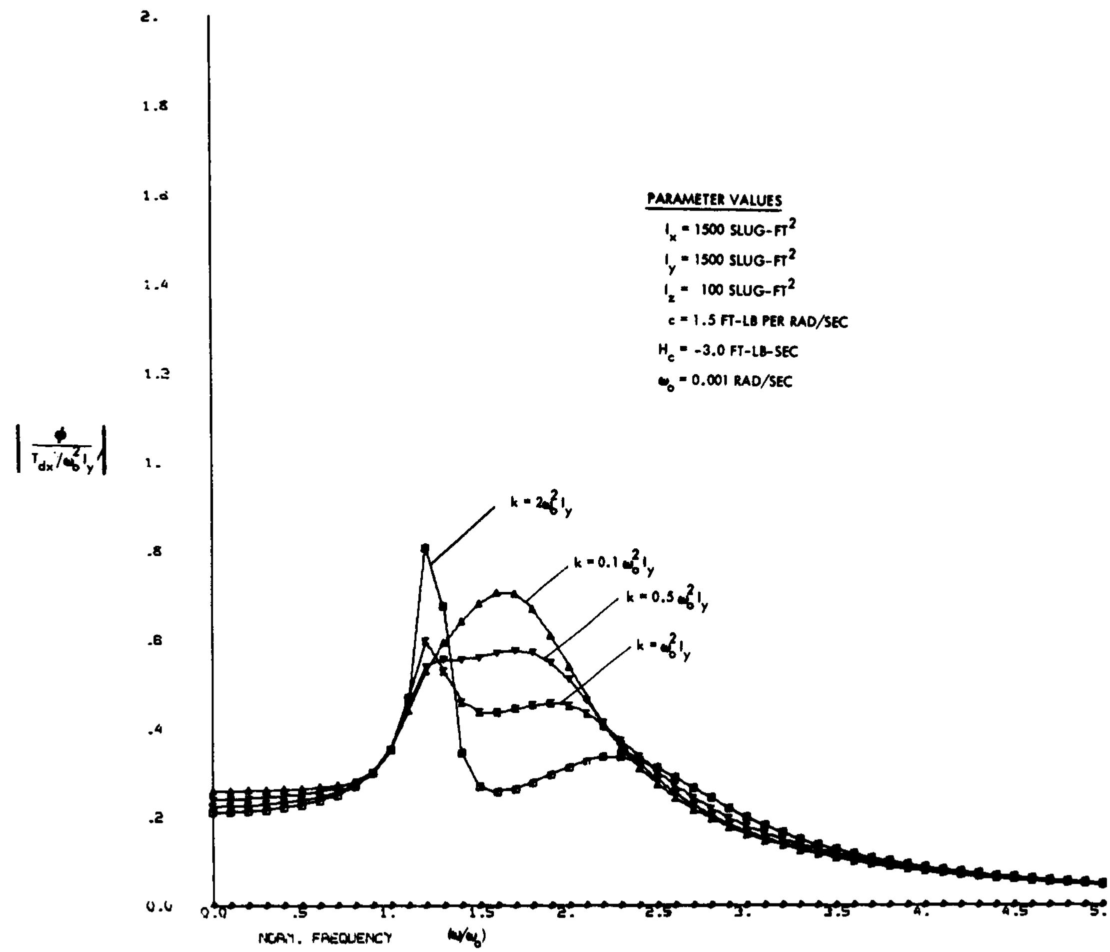

Figure 7 Roll Response to Roll Disturbances

- Control/principal axis misalignments are 2° about the pitch and roll axes prior to boom deployment; the boom is deployed precisely along the negative yaw control axis.
- Orbital eccentricity is 0.01; the nominal orbital altitude is 750 nautical miles.

The dominant disturbance components were found to be at zero and orbital frequency; these torque components, with the relative significance of the various torque sources, are summarized in Table III (for additional components and a more complete description of the study see Appendices IV and VI).

Table IV summarizes the system performance, as developed in Appendix IV, with the parameters of (2.4). Two observations are of interest. First, it is not surprising to find that such factors as gimbal inertia and displacement of the wheel assembly from the vehicle cm have no significant effect upon performance. Second, it should be stressed that there has been no strenuous effort made to "optimize" performance by means of multiple design iterations. The performance levels exhibited here, acceptable for a wide variety of applications, can be improved either by providing increased roll/yaw stiffness (with a probable increase in system weight due, for example, to a heavier reaction wheel assembly) or by configuring the spacecraft so as to reduce the environmental disturbances (e.g., striving for a high degree of magnetic cleanliness, and by coating the Be-Cu inertia mast).

	Sun in Orbit Plane						Sun Normal to Orbit Plane					
	Roll		Pitch		Yaw		Roll		Pitch		Yaw	
	d.c.	ω_o	d.c.	ω_o	d.c.	ω_o	d.c.	ω_o	d.c.	ω_o	d.c.	ω_o
Magnetic Moments	.22*	1.65	1.60	2.10	.16	.87	.30	2.55	.45	1.95	.15	1.80
Thermal Bending	.05	.10	.10	6.50	0	0	7.30	0	0	0	.25	0
Solar Pressure	0	0	0	.60	0	0	.85	0	0	0	0	0
Misalignments	1.85	0	.85	0	.10	0	2.50	0	.75	0	.05	0
Eccentricity	0	.10	0	2.60	0	.10	0	.05	0	2.60	0	.02
Total**	2.12	1.85	2.55	11.8	.26	.97	11.0	2.60	1.20	4.55	.45	1.82

* Entries are $10^5 T_d$ in ft-lb

** Contributions assumed to be "in phase"

Table III. Constant and Orbital Frequency Disturbance Torques

Disturbance Source	Steady-State Errors* (deg)					
	ϕ		ψ		γ_x	
	Case I	Case II	Case I	Case II	Case I	Case II
Magnetic Moment, Solar Radiation, Boom Bending	.76	2.0	1.45	2.41	.78	1.7
Control Axis Misalignment	.20	.26	.02	--	.16	.21
Orbital Eccentricity (1%)	.01	.01	.02	.01	--	--
Total	.97	2.27	1.49	2.42	.94	1.91

* Note: Case I - Sun in orbit plane
Case II - Sun normal to orbit plane

Table IV. Steady-State Roll/Yaw Errors ($k = .5\,\omega_o I_y$; $c = \omega_o I_y$; $H_c = -2\omega_o I_y$; $I_y = 1500$ slug-ft^2)

It should be noted that the total effect of orbital eccentricity cannot be determined via linear analysis, since eccentricity introduces periodic coefficients as well as additional forcing terms. A preliminary evaluation of the possible effect upon roll/yaw performance was undertaken using the Generalized Spacecraft Simulation, with a pure pitch bias system (wheel momentum always along the pitch spacecraft axis) and no damping. With an orbital eccentricity of 0.05 and an initial yaw error of two degrees the subsequent roll and yaw errors were never greater than 1.2^o and 2.0^o, respectively. These results, while inconclusive, suggest that the SAGS control configuration will provide acceptable fine control in the presence of moderate orbital eccentricities.

3.0 ACQUISITION PERFORMANCE EVALUATIONS

The importance of fine control performance and its dominant role in the determination of system parameters has been stressed in the previous section. Although of less significance from a long-term operational standpoint, acquisition performance can, if inadequate, negate the entire mission. In general, acquisition performance must satisfy two requirements: (i) it must, above all, attain the nominal spacecraft attitude orientation successfully from a set of reasonable initial conditions, and (ii) it must do so within a certain upper time limit. Specification of the initial conditions and of the time allowed for initial acquisition is a matter which depends upon the spacecraft and its mission requirements.

For a satellite employing the SAGS control configuration initial acquisition will generally consist of three phases:

(i) Rate Damping - during which the initial angular velocities following injection are reduced to a low level.

(ii) Roll/Yaw Acquisition - during which the pitch (wheel) axis is aligned normal to the orbit plane.

(iii) Pitch Acquisition - during which the yaw spacecraft axis is aligned with the local vertical (toward the earth).

The rate damping phase will commence at injection; its character (indeed, whether it is essential) will depend upon the nature of the injection stage. If the vehicle is injected from a spinning stage one of the several available despin mechanisms may be used, while with a fully-stabilized injection the rate removal mechanism, if required, may consist of three rate gyros and a low impulse three-axis pneumatic system. In any event when, following rate damping, the solar array and the inertia mast are deployed, the angular velocities of the spacecraft will be less than orbit rate. From this point the normal attitude control mechanism must complete the acquisition maneuver.

Within this framework there are still significant operational alternatives. For example, the reaction wheel speed may be maintained fixed until roll/yaw acquisition is complete (thus decoupling pitch acquisition from roll/yaw acquisition), or the pitch control loop may be enabled immediately after the completion of the deployment phase. In the latter case, scan cone/vehicle intersection due to large gimbal motions must be avoided. Moreover, the effect of large wheel speed variations upon roll/yaw acquisition could be significant. For these reasons the alternate course has been favored in the following discussions.

Conceptually it is useful to consider the two terminal acquisition phases to be decoupled (as they will in fact be, if pitch attitude control is disabled until roll/yaw acquisition is complete). Roll/yaw acquisition is accomplished via the combined effects of gravity gradient torques and the gyroscopic torques induced by the presence of the reaction wheel momentum bias. The gyroscopic torque will cause the wheel to seek a condition of alignment with the spacecraft's angular velocity vector. The mechanisms which couple the rotor to the vehicle assure that this motion can be an equilibrium state only when gimbal deflections are absent. The gravity gradient torques will perturb the total system angular momentum until it is aligned normal to the orbit plane.[3] Roll/yaw acquisition will terminate with the vehicle either oscillating or spinning about the pitch axis, with the wheel gimbal undeflected, and with the wheel momentum directed along the spacecraft's orbital momentum (i.e., normal to the orbit plane in the usual right-hand sense). This phase should require from two to ten orbits for the system studied here.

Removal of the pitch spin rate can be achieved by running the wheel momentum alternately between its upper and lower limits with a frequency of two cycles per satellite revolution, in response to the horizon scanner output. The satellite will then capture with the yaw axis pointed either toward or away from the earth's center. In the latter case a turnover maneuver must be executed, either by means of the reaction wheel or by retracting and reextending the inertia mast. The entire pitch acquisition maneuver should require no more than five orbits.

A reasonable constraint upon acquisition is that this maneuver be accomplished within the confines of the control configuration selected for fine control. As reported in Section 2, a single (roll) degree-of-freedom wheel suspension provides favorable fine control performance and is, of course, more easily implemented than a two degree-of-freedom mechanism. Acquisition studies have been undertaken for the roll gimbal configuration only.

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[3] Unimportant singular situations can exist; for example if the vehicle rates following deployment are <u>identically</u> zero, and if the yaw axis is precisely normal to the orbit plane with $I_x = I_y$, no torques can result and the vehicle would remain in this attitude. Of course, such (unstable) singularities are of academic interest only.

3.1 Roll/Yaw Acquisition

The system performance in converting the spacecraft attitude variations after rate removal into a pure pitch motion about the orbital momentum vector has been considered both analytically and via digital simulation. Although the analytic efforts have been unrewarding, the simulator studies have yielded concrete results. Both aspects of these investigations are detailed in Appendix I.

One important characteristic of the semi-active gravity gradient control concept examined during this study is that, unlike many gravity gradient configurations, it results in a unique terminal yaw orientation. The only stable yaw attitude is with the relatively large wheel momentum bias adding vectorially to the orbited momentum of the spacecraft; the equilibrium in which these momenta are opposed has been shown to be unstable (Appendix I).

Successful completion of roll/yaw acquisition will be signalled by the alignment of the pitch axis (y_b) with the y_r axis of Figure 1. In this condition the direction cosine matrix relating the (x_b, y_b, z_b) frame to the (x_r, y_r, z_r) frame will be of the form:

$$\begin{bmatrix} \bar{x}_b \\ \bar{y}_b \\ \bar{z}_b \end{bmatrix} = \begin{bmatrix} \cos\theta & 0 & -\sin\theta \\ 0 & 1 & 0 \\ \sin\theta & 0 & \cos\theta \end{bmatrix} \begin{bmatrix} \bar{x}_r \\ \bar{y}_r \\ \bar{z}_r \end{bmatrix} \qquad (3.1)$$

Thus the element a_{22} is a good measure of the state of the roll/yaw acquisition maneuver.

A representative set of simulated roll/yaw acquisitions is presented in Appendix I. These runs (which assume a proportional damper) center around the set of baseline system parameters of Table V.[4] Notice that normalized as well as numerical values are given because, as is demonstrated in Appendix II, these results will apply equally to any situation in which the <u>normalized</u> parameters are retained, regardless of the orbit rate (ω_o) and the pitch inertia (I_y). These baseline values were selected from considerations of fine control performance and of implementation requirements.

	Baseline Value Normalized	Numerical
Orbit rate, ω_o (rad/sec)	ω_o	0.001
Pitch inertia, I_y (slug-ft^2)	I_y	1500.
Roll inertia, I_x (slug-ft^2)	I_y	1500.
Yaw inertia, I_z (slug-ft^2)	$.067\ I_y$	100.
Bias momentum, H_c(ft-lb-sec)	$-2\omega_o\ I_y$	-3.0
Damping coefficient, c (ft-lb per rad/sec)	$\omega_o\ I_y$	1.5
Spring constant, k(ft-lb/rad)	$0.1\ \omega_o^2 I_y$	1.5×10^{-4}
Gimbal stop, γ_s(deg)	--	30

Table V. Baseline Roll/Yaw Acquisition Parameter Values

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[4] Note that $\bar{H}_c = H_c\ \bar{y}_b$ is the wheel momentum with the gimbal undeflected; therefore, H_c must be negative so that $\bar{H}$ will add to the orbital momentum at the desired stable equilibrium.

Figure 8 shows the variation of a_{22} during an acquisition with the baseline system parameters for a case in which the vehicle is initially stationary in inertial space with a roll attitude error of 89 degrees. The roll/yaw acquisition time (i.e., here the time required to make a_{22} permanently greater than 0.95) is 3.3 orbits. The set of runs presented in Appendix I is summarized in Table Vl.[5]

Based upon the digital computer simulation of roll/yaw acquisition, it appears that the suspension parameter which most affects acquisition performance is the damping coefficient (see, for example, runs 1, 6 and 7 of Table VI). The spring constant seems to have relatively little effect upon acquisition performance (runs 1 and 8).

The major conclusion to be drawn from the simulator study is that the baseline system parameters, selected from the viewpoint of providing good steady-state roll/yaw performance, will also assure acceptable acquisition behavior.

3.2 Pitch Acquisition

This maneuver places distinct and significant requirements upon the momentum range of the wheel, the torque level within this momentum range, and the characteristics of the horizon attitude computer.

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5 Although each of these runs terminated in bounded pitch oscillations (thus eliminating the requirement for a subsequent pitch capture), it is not safe to conclude that this would always be the case.

Run No.	Fig.	Convergence Time ($a_{22} > 0.95$)	Terminal Pitch Motion*	Remarks
1	8, I - 1	3.3 revs	Bounded pitch oscillation ($a_{33} < 0$)	Baseline parameter values; 89° initial roll error.
2	I - 2	1.5 revs	Bounded pitch oscillations ($a_{33} > 0$)	Baseline parameter values; 89° initial yaw error.
3	I - 3	7.9 revs	Bounded pitch oscillations ($a_{33} > 0$)	Baseline parameter values; 179° initial roll error.
4	I - 4	2.5 revs	Bounded pitch oscillations ($a_{33} < 0$)	Gimbal stop at 45°; compare with Run 5.
5	I - 5	3.4 revs	Bounded pitch oscillations ($a_{33} > 0$)	Gimbal stop at 20°; compare with Run 4.
6	I - 6	3.3 revs	Bounded pitch oscillations ($a_{33} < 0$)	Effect of twice baseline damping; compare with Runs 1 and 7.
7	-	5.0 revs	Bounded pitch oscillations ($a_{33} < 0$)	Effect of one-half baseline damping; compare with Runs 1 and 6.
8	I - 7	4.0 revs	Bounded pitch oscillations ($a_{33} < 0$)	Effect of 10 times baseline spring constant; compare with Run 1.
9	I - 8	4.1 revs	Bounded pitch oscillations ($a_{33} < 0$)	Effect of zero net initial angular momentum; compare with Run 1.

* $a_{33} < 0$ indicates "upside-down" pitch attitude

Table VI. Summary of Representative Roll/Yaw Acquisition Runs

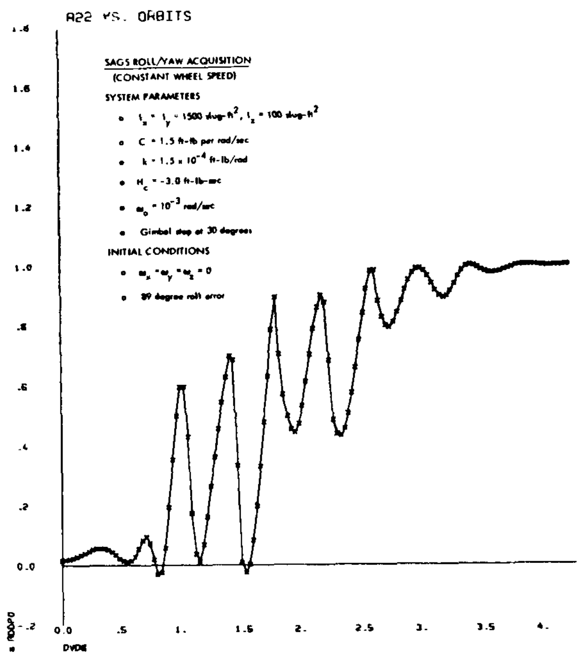

Figure 8 Roll/Yaw Acquisition with Baseline Parameters: a_{22} v. Orbits

Pitch capture is effected by cycling the reaction wheel speed between its upper and lower limits (i.e., through an increment $2\ H_D$), in synchronization with the pitch tumbling motion, on the basis of the processed horizon scanner output. In order to be effective, the wheel momentum increments (and the resultant pitch rate increments) must be properly phased with the pitch attitude; that is, the magnitude of the pitch rate should be decreased by $2H_D/I_y$ when at attitudes of maximum potential energy ($\Theta = 0,\pi$) and increased by equal increments when the potential energy is a minimum. In this way the tumbling rate will be removed without requiring a secular change in wheel momentum, and capture (that is, the reduction of the tumbling motion to bounded pitch motion) will ensue.

Notice that for proper synchronization between wheel control and vehicle motion the attitude error signal should be zero at $\Theta = \pi/2$ and $3\pi/2$, as well as at the origin. As shown in Appendix I, either the "sin2Θ" processing method, or an appropriate blanked scheme will provide this characteristic. Although the "sinΘ" technique of reducing the horizon scanner output is inadequate for capture purposes, it is useful in providing a closed-loop turnover capability with wheel control.

Figure 9 shows a typical pitch capture with "sin2Θ" processing, for $H_D = 1$ ft-lb-sec and a wheel torque level of 5 in-oz. Additional examples are presented in Appendix I. Notice that the acquisition time and H_D are inversely related; for a momentum bias level of -3.0 ft-lb-sec (corresponding to a motor speed of 1500 rpm) a control range of 1 ft-lb-sec is reasonable upper limit, in that it will

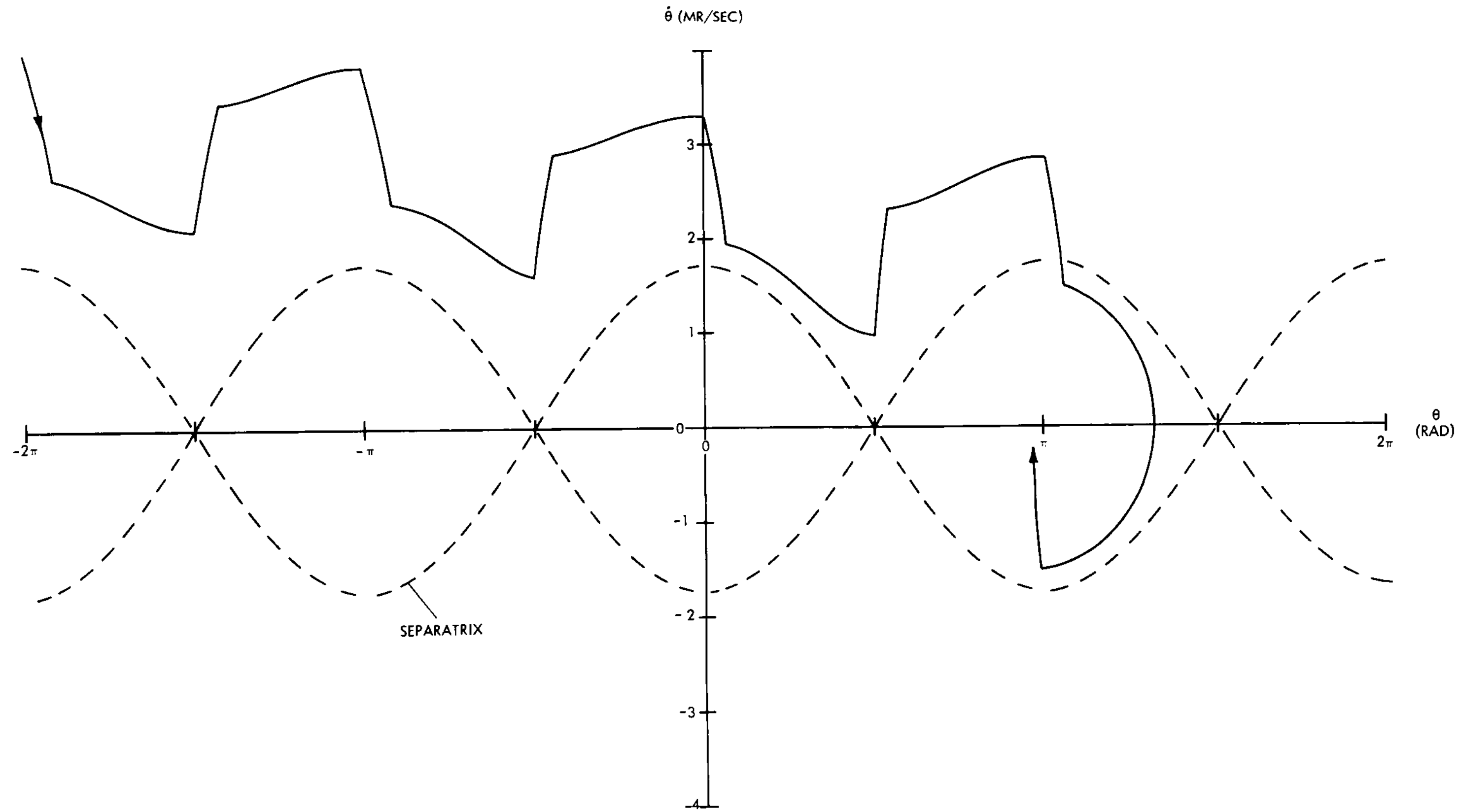

Figure 9 Pitch Capture with "Sin 2θ" Processing
(H_D = 1 ft-lb-sec, T_m = 5 in-oz)

maintain a range of wheel speed (1000 - 2000 rpm) suitable for horizon scanning. The pitch torque level must then be sufficient to produce the required momentum increment during a small pitch attitude variation.

Although a blanked processing scheme will also produce pitch capture, the acquisition time will be at least twice what it is with "sin2θ" processing. For this reason the unblanked characteristic should be chosen if permitted by the vehicle structure.

Pitch capture may terminate with the vehicle "upside-down" ($\bar{z}_b = - \bar{z}_r$); Figure 9 is an example of this behavior. In this event a turnover maneuver must be executed, preferably using the reaction wheel. As is shown in Appendix I, this maneuver can be effected (in the case of an unblanked scanner) by switching from "sin2θ" processing to "sinθ" processing when capture is complete. The resulting motion is shown in Figure 10. Note that this closed loop technique (requiring a single ground command) is preferable to an open-loop maneuver; however, if significant blanking is required the turnover will probably require some degree of open loop operation.

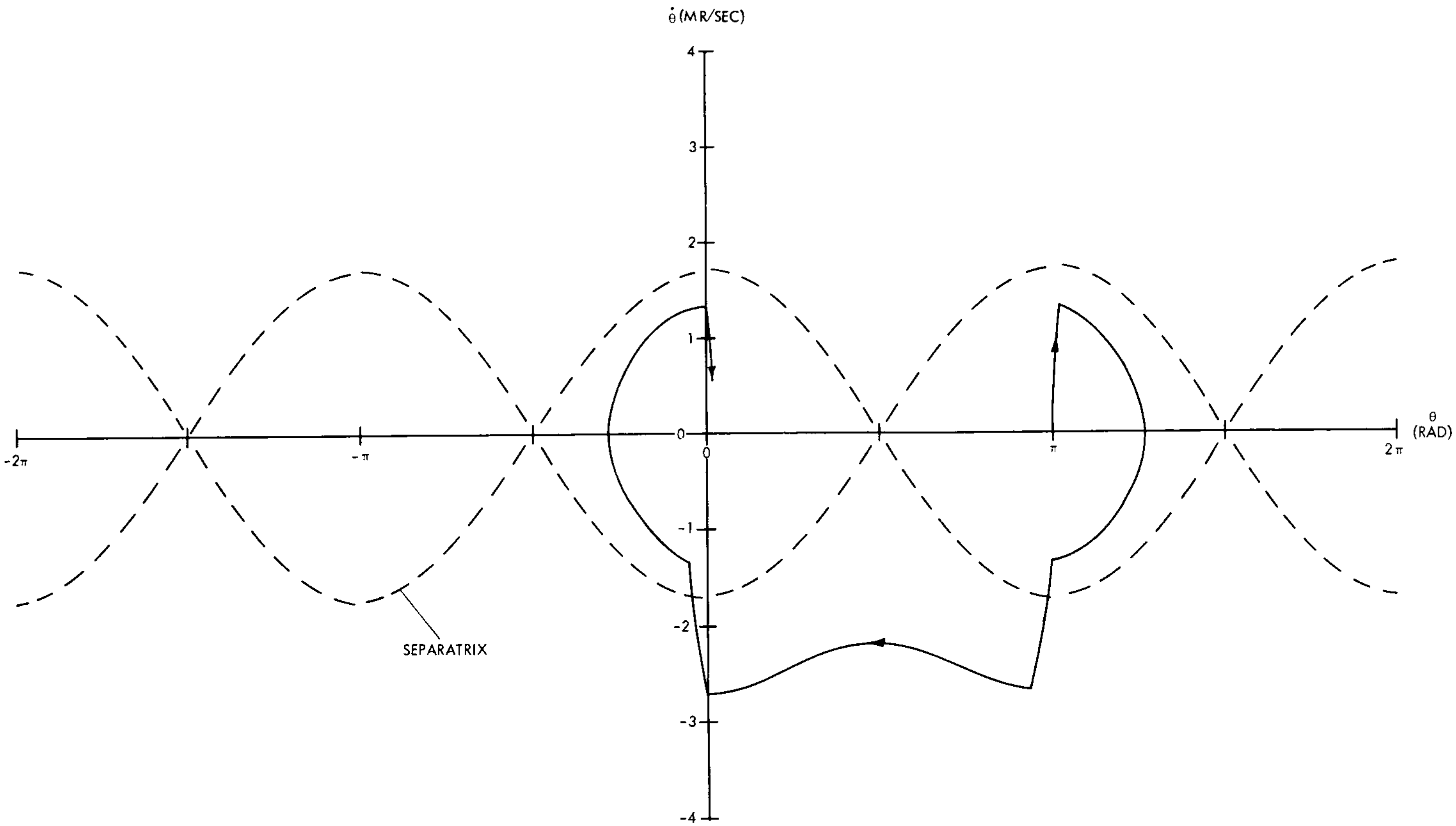

Figure 10 Turnover with "Sin θ" Processing (H_D = 1 ft-lb-sec, T_m = 5 in-oz)

4.0 IMPLEMENTATION EVALUATIONS

This section summarizes the results of the implementation studies detailed in Appendix VII. Major emphasis has been devoted to the mechanical aspects of the implementation problem, since these factors present the greatest test of the feasibility of implementing the SAGS control concept; however, a brief description of the control electronics and signal processing is included in Appendix VII.

Figure 11 illustrates the more significant features of the mechanical controller unit. Major elements include the assembly housing, the suspension system, a damping mechanism, the reaction wheel/motor unit and a horizon sensor system. It is noteworthy that this design permits all mechanical attitude control functions (i.e., attitude sensing and control torque generation) to be effected by one unit; although a separate scanner mechanism is possible (and may be desirable under certain circumstances) the present approach was considered here because it presents the greatest potential for reduced weight and power, and increased reliability.

Two detailed designs have been evolved during the course of these investigations. These differ primarily in the configuration of the optical system. In both designs, radiant energy passes through the housing via a germanium window to a prism and objective lens which rotate with the reaction wheel (thus providing the desired conical scan pattern). However, the configuration of Figure 12 features a case-mounted sensing element with optical signal trans-

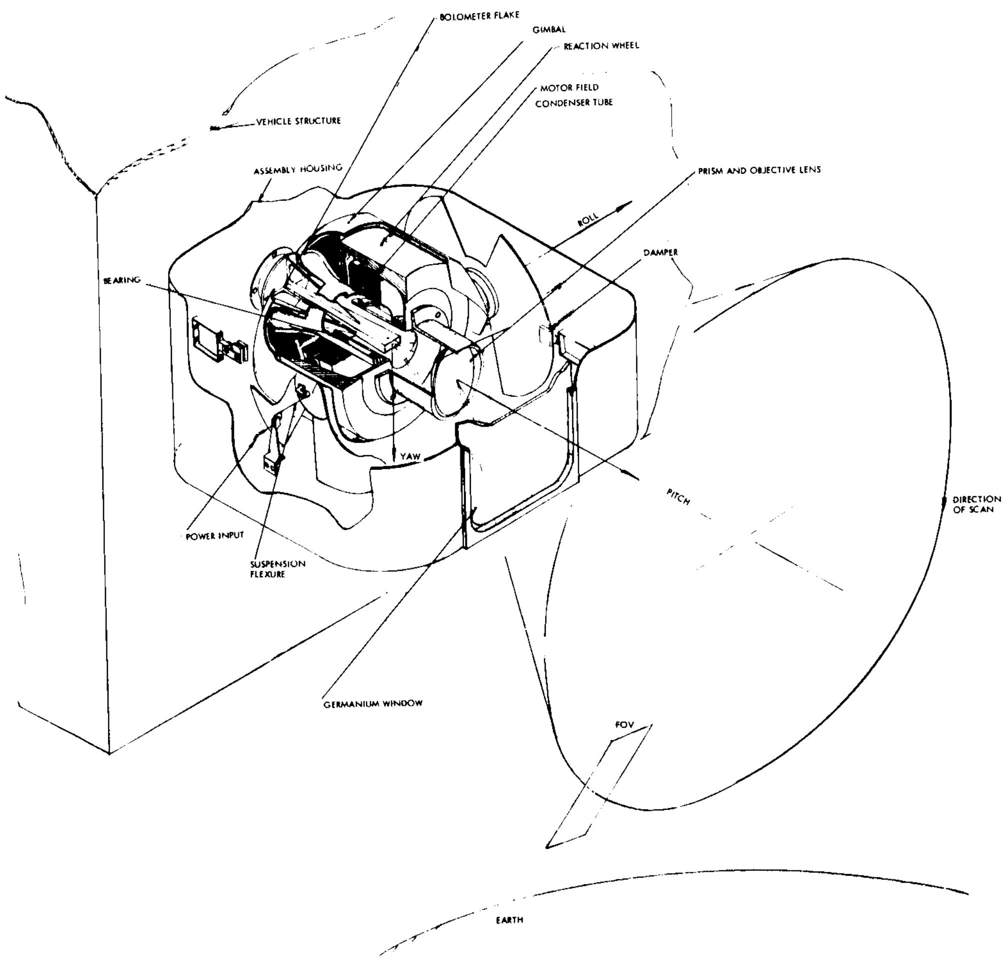

Figure 11 Conceptual View of Gimballed Reaction Wheel/Scanner Assembly

mission from the gimballed wheel assembly to the housing, while in the alternate design (Figure 13) the bolometer is affixed to the motor so that signal transmission through the suspension must be electrical. These configurations differ further in that one (Figure 12) employs a hysteresis damper and the other an eddy current mechanism.

The major characteristics of these units are summarized in Table VII. The subsequent paragraphs describe the various subsystems in more detail. Additional data, design details and alternate approaches are to be found in Appendix VII.

4.1 Motor/Reaction Wheel Assembly

The motor assembly employed is an inside-out, two-phase (400 ~) induction machine. Single-phase "on-off" power is provided (via the torsion wire suspension) as demanded by the control electronics, with the necessary phase-shifting provided by four capacitors. Elements of the horizon sensing system are mounted in a channel provided in the motor unit. A magnetic pickoff provides motor speed information and reference pulses for horizon sensor signal processing.

The reaction wheel is constructed almost entirely of aluminum alloy materials.[6] The two thin section ultra precision radial bearings are of symmetrical deep groove design with one integral shield

[6] The designs indicated in Figures 12 and 13 were originally evolved for a reaction wheel momentum of 7 ft-lb-sec, based upon preliminary estimates of the momentum bias requirements. Subsequent detailed analyses showed a bias momentum of 3 ft-lb-sec to be adequate. In order to arrive at weight and power estimates more consistent with this reduced momentum requirement (without performing a complete mechanical redesign) the rotor material was changed from stainless steel to aluminum alloy. It should be noted that the weight and size of the assembly would decrease somewhat if stainless steel were retained with the diameter of the wheel (and the housing dimensions) reduced accordingly. Other detailed aspects of the design (e.g., the motor bearings and the torsion wires) would also be affected somewhat if a more optimum mechanical design were developed. However the basic configuration would be unaltered.

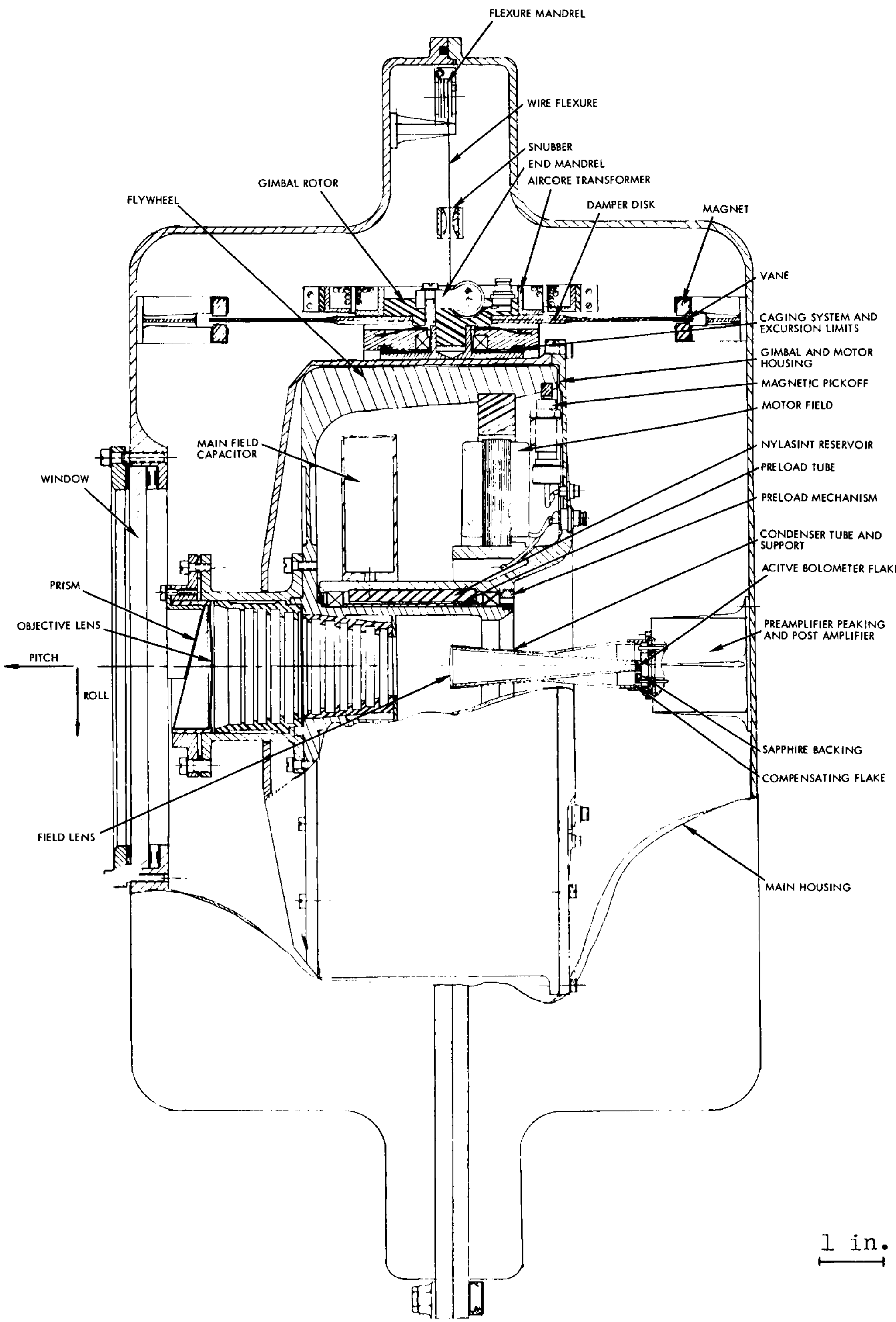

Figure 12 Detailed Design with Case-Mounted Bolometer and Hysteresis Damper

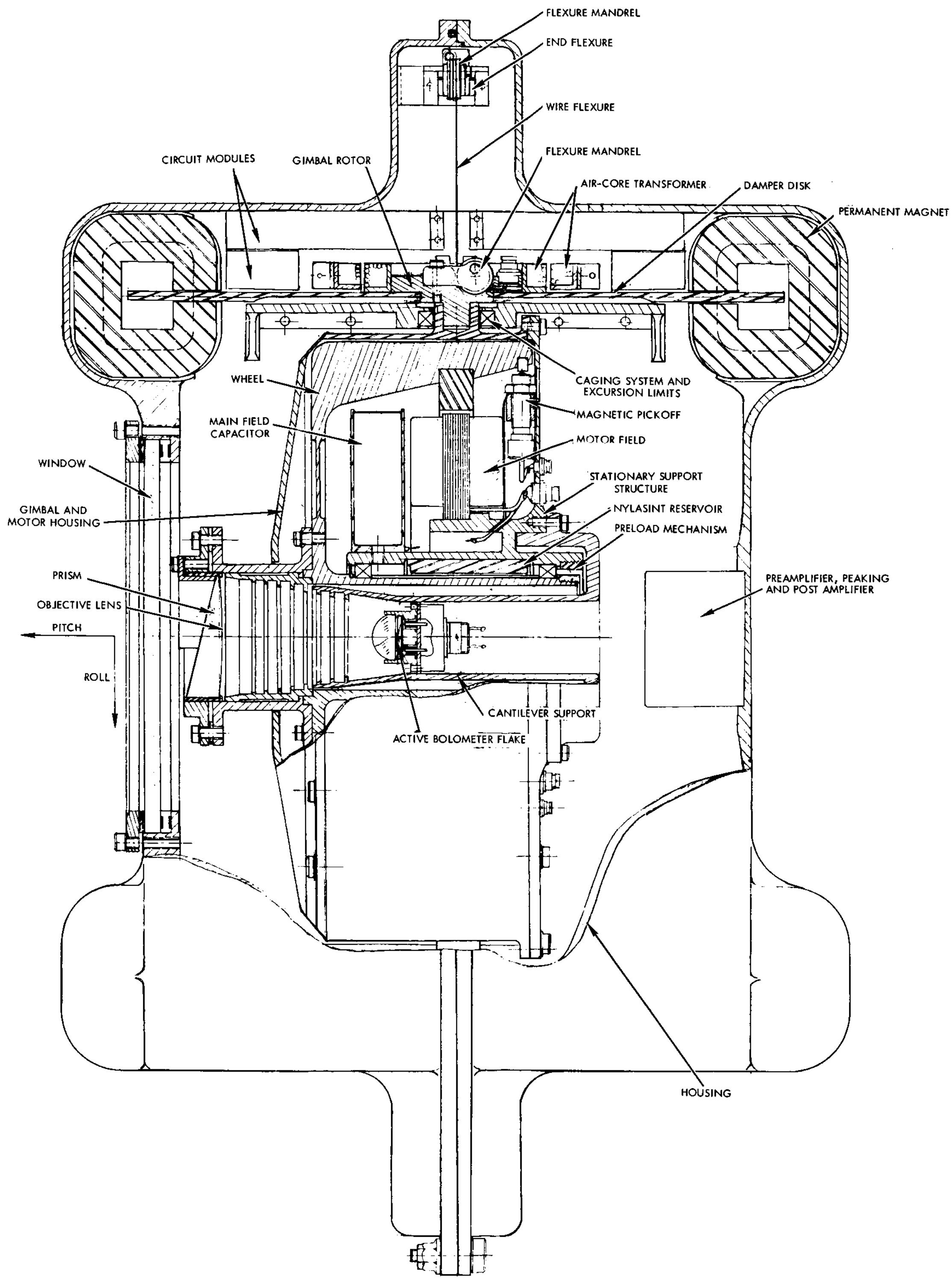

Figure 13 Detailed Design with Wheel-Mounted Bolometer and Eddy Current Damper

Characteristic	Value
Reaction Wheel/Motor Assembly	
o Nominal control torque	5.5 in-oz
o Momentum at bias speed	3 ft-lb-sec
o Controllable speed range	1500 ± 500 rpm
Horizon Sensor System (CO_2 band)	
o Bolometer input power (case mounted bolometer)	5.05 microwatts
o Bolometer input power (wheel mounted bolometer)	9.40 microwatts
Power Consumption*	
o Motor power	
- nominal	5 watts
- maximum	32 watts
o Electronics	
- nominal	9 watts
- maximum	14 watts
Controller Dimensions (excluding flexure housings)	13" x 10" x 10"
Controller Weight	
o Total mechanical unit**	
- with hysteresis damper	22.5 lb
- with eddy current damper	28.5 lb
o Housing (including window)**	8.0 lb
o Electronics Assembly	4 lb

*Maximum power will be required only during the pitch acquisition maneuver.
**These values can be reduced by approximately 3 lbs by fabricating the housing from magnesium; further improvement may be realized by reducing the case dimensions.

Table VII Summary of Controller Characteristics facing outwards towards the sides of the assembly. Alternate balls are slightly undersized and serve as idler type spacers for the load carrying balls. Such a design tends towards reduction of the internal sliding friction. Axial preload of the bearings is accomplished as shown on Figure 12 to prevent vibration impacts. Bearings are normally oil lubricated with lubricant retention within the bearing promoted by a porous Nylasint oil reservoir.[7] The wheel itself is so designed that when in a severe vibration environment it deflects sufficiently to gap the existing clearance space between the outer wheel surface and the gimbal structure thus limiting the load transmission to the shaft and bearings. The stationary parts are as light as possible consistent with good design practice.

4.2 Suspension System and Housing

The controller housing is a two-piece, aluminum webbed structure, the interior of which is pressured (with helium) at 0.15 atmosphere. The necessary scanner field-of-view is afforded by a large (6" x 8") germanium window.

The suspension system consists of a pair of torsion wire flexures and a caging mechanism. Each torsion wire is wrapped around a mandrel and fixed by a retaining screw. The outboard mandrels are mounted to the housing via cantilever end-flexures

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[7] Bearings are lubricated with a light, general purpose diester instrument oil (with a Plexol base), while the oil reservoir is impregnated with Plexol 201 oil. With these oils and the designs shown, the optical effects due to oil vaporization should be negligible.

which provide a means to apply preloading to the wires and also afford a degree of protection for the wires. The torsion wires, made of 0.013" diameter beryllium-copper, were chosen on the basis of such factors as power transmission capabilities, torsional and lateral stiffness, and heat transfer considerations.

A passive caging mechanism, incorporating both translational and rotational stops, protects the gimballed reaction wheel assembly and the suspension during periods of abnormal excitation.

4.3 Damper Mechanism

Three of the many possible sources of energy dissipation merit serious consideration for this application. Of these, the two electromagnetic techniques (eddy-current and hysteresis) appear most promising, while the third (viscous shear) does not appear to be compatible with other components of this device; in this regard mechanization of a fluid damper, either as a separate sealed unit or by flotation of the entire gimballed assembly, leads to significant problems.

Controllers using hysteresis and eddy current dampers are shown in Figure 12 and 13, respectively. Notice that the hysteresis damper is considerably more compact than the other; the corresponding weight difference is approximately 6 lbs. However, this comparison depends heavily upon the application because the eddy current damper weight is a strong function of the required amount of energy removal (e.g., if c is doubled in the present case the damper weight increases by about 3 lbs) while the hysteresis damper will weigh on the order of one pound for any reasonable energy dissipation requirement. The primary problem associated with the hysteresis damper is the fact that its performance cannot be established via simple linear response techniques. Rather, complex simulations must be employed, the development of which was outside the scope of this study. A gross sizing of a hysteresis device is easily accomplished from past experience, and future refinements will have little effect upon the salient characteristics (i.e., size and weight) of the hysteresis damper.

4.4 Horizon Sensing System

Major elements of both horizon sensing system designs are the prism and objective lens (mounted on the rotating part of the motor assembly) and the detector. However the design of Figure 12 is unique in that the bolometer is mounted in the stationary housing while the prism and objective lens are located in the gimballed portion of the unit. The resulting "bend" in the optical axis (which occurs when the torsion wire suspension deflects) is accommodated by a pyramidal condensing channel with its entrance located at the intersection of the optical axis with the gimbal axis. Gimbal deflections as large as 20 degrees will not result in any appreciable change in efficiency.

The alternate design of Figure 13 includes the bolometer as part of the gimballed wheel unit. This more conventional design provides greater optical efficiency (see Table VII), but requires electrical (rather than optical) transmission of the raw attitude information to the case. This can be accomplished via hard wires (with the possibility of significant restraint torques on the gimballed assembly) or by means of more exotic indirect techniques as outlined in Appendix VII.

4.5 Signal Processing and Control Electronics

The circuitry required by error signal conditioning and reaction wheel speed control is indicated in the block diagram of Figure 14. Provisions are included for implementing either the "sin2θ" or the "sinθ" processing schemes (described in Appendix I). The reaction wheel control electronics supply an "on-off" motor drive voltage based upon the conditioned error information and wheel speed information. Maintenance of the wheel speed within the control range (1000 to 2000 rpm) required by momentum bias and scanning considerations is assured by a high-gain speed inhibiting loop with an appropriate deadband.

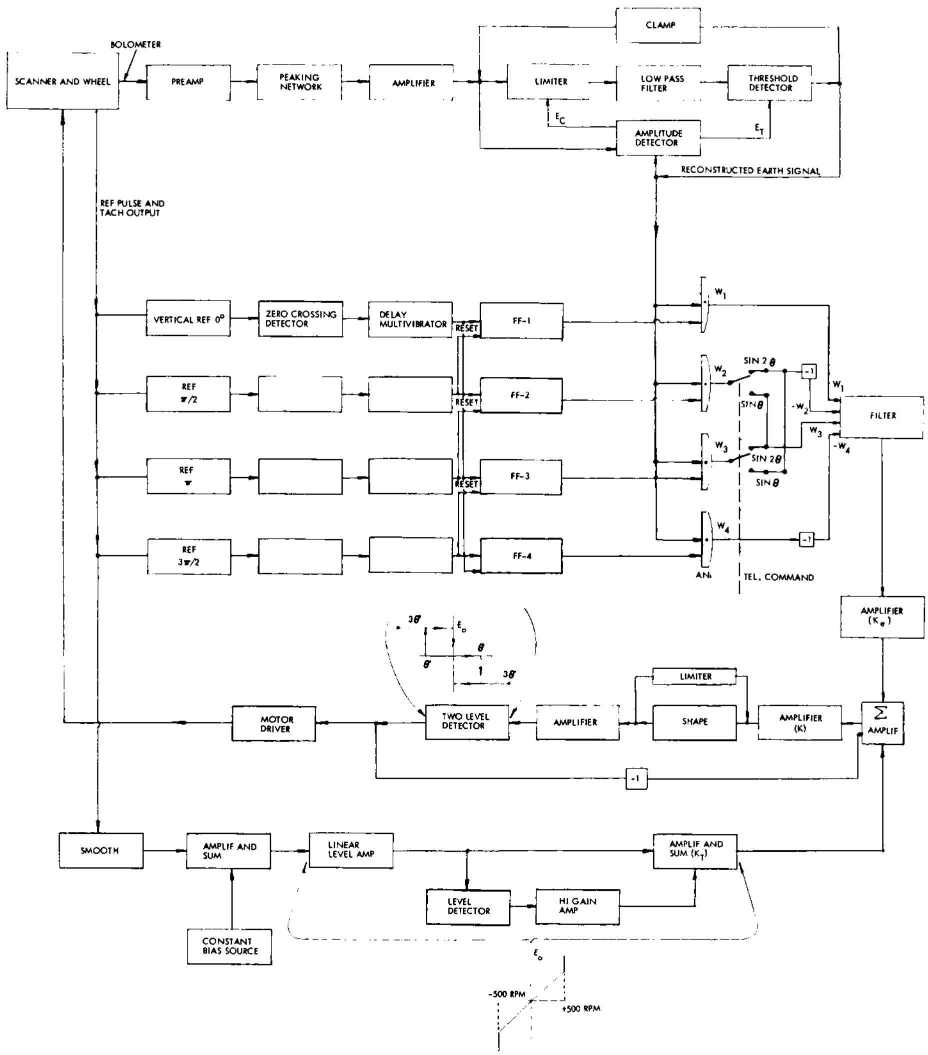

Figure 14 Signal Processing and Control Electronics

5.0 CONCLUSIONS

This study has established the operational feasibility of the SAGS control concept and has, as well, shown that this control technique can be readily mechanized. Major detailed results (with numerical data based upon the spacecraft configuration and mission specified in Section 2.0) are itemized below:

(i) Steady-state roll, pitch and yaw pointing accuracies of approximately 1.0°, 1.0° and 1.5°, respectively, can be readily achieved for the nominal case (sun in the orbit plane) with the vehicle configuration here considered. The primary contributions to the roll/yaw errors are magnetic torques and thermal boom bending; both sources can be attenuated considerably, the former by striving for a high degree of magnetic cleanliness and the latter by coating the inertia mast. If these steps are taken it is not unreasonable to expect two-fold improvements in the roll/yaw accuracies without any change in the controller parameters. For a vehicle having more favorable structural properties, roll/yaw accuracies on the order of a few tenths of a degree are a reasonable design goal, while the attainable pitch accuracy is limited only by the horizon scanner.

(ii) The system parameters selected for desirable fine control performance are compatible with good acquisition operation. Assuming acquisition to begin with the vehicle stationary in inertial coordinates, but with an arbitrary attitude, roll/yaw acquisition will require from two to ten orbits.

Subsequent pitch capture (and turnover, if required) should take place within three orbits. Note, however, that full power output should be available from the solar array during this terminal maneuver.

(iii) The performance levels summarized above can be attained from a single mechanical unit which performs both the control actuation and the attitude sensing function. The weight of this unit with a proportional damper is approximately 28 lbs. However, TRW experience indicates that equivalent (if not better) performance can be obtained with a hysteresis damper, in which case a reduction of 6 lbs in the controller weight will be achieved. Modifications in the housing (notably, use of magnesium in its fabrication) will save an additional 3 lbs. Including 4 lbs of control electronics, total ACS weights as low as 25 lbs are achievable for the mission, spacecraft, and performance levels herein considered. Nominal power consumption is 14 watts, with a peak drain of 46 watts during pitch acquisition.

The more significant spacecraft and controller parameters are summarized in Table VIII.

Parameter	Value
Spacecraft roll inertia (slug-ft^2)	1500
Spacecraft pitch inertia (slug-ft^2)	1500
Spacecraft yaw inertia (slug-ft^2)	100
Wheel bias momentum (ft-lb-sec)	-3.0
Wheel momentum range (ft-lb-sec)	-2.0 to -4.0
Nominal motor torque (in-oz)	5.0
Gimbal spring restraint (ft-lb/rad)	10^{-4} to 10^{-3}
Damping coefficient (ft-lb per rad/sec)	1.5

Table VIII Summary of Vehicle and Controller Parameters

6.0 NEW TECHNOLOGY

This study concerns a new method of controlling the attitude of an earth-oriented spacecraft using a single reaction wheel. Roll/yaw control is achieved by operating the reaction wheel with a momentum bias, and by gimballing the wheel and coupling it to the vehicle through an energy-removal mechanism to provide roll/yaw damping. Pitch control is provided by controlling the wheel speed so that, for small attitude errors, the momentum differs from the bias momentum by an increment which is essentially proportional to the pitch attitude error (as indicated by a horizon scanner). This technique, developed by TRW personnel, is considered to be a new concept.

In addition to establishing the performance potential of the above concept, this study has considered its implementation. These investigations have resulted in original detailed designs employing a reaction wheel suspended on torsion wires (see Section 4.0 for a detailed description). In the mechanization of this concept, the horizon sensing function has been incorporated into the reaction wheel, with the bias speed providing the necessary scan. Although this reaction wheel/scanner concept is not new, an approach taken in solving the optical problems introduced by gimballing such a unit is an original one, employing a technique which allows optical transfer of the error signal from the gimballed wheel to the case of the assembly (Figure 12). It should be noted, however, that the reaction wheel/scanner approach is not required for implementation of this control concept; a separate scanner may be employed.

A patent disclosure pertaining to the above developments has been filed.

7.0 REFERENCES

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NASA-Langley, 1966 CR-593

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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