ATTITUDE DETERMINATION AND CONTROL FOR THE MULTI-MISSION SPACE PAYLOAD PLATFORM (SPP)

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1. INTRODUCTION

The first artificial satellite, Sputnik, was a "smallsat" of sort with its 58 cm. diameter and mass of just under 84 kg. Since then, design trends have produced progressively larger spacecraft tracking the launch vehicle throw capabilities. This approach, however, is being challenged today and, although large satellites will continue to meet global requirements cost effectively, there is a need to accommodate specific payloads with smaller spacecraft buses.

The economy of scale that called for larger satellites in the past, is now focusing on a multi-mission design approach. A general purpose spacecraft bus can be designed to support a wide range of payloads: meteorological, surveillance, communication, and scientific. General Electric Astro Space has developed such a reliable and cost effective vehicle, the **Space Payload Platform** (Photo 1), with a common core bus weighing 130 kg. (dry weight), available in both spin-stabilized and three-axis stabilized configurations. Compatible with both Air Launch Vehicle (ALV -Pegasus) and Standard Small Launch Vehicle (SSLV-Taurus), the SPP allows up to a 300 kg, 260 Watt payload to be injected directly into a low altitude circular or elliptic orbit. This basic capability can be augmented by an on-board hydrazine propulsion system to achieve, correct, and/or maintain higher altitude orbits. To accommodate these various programs, a flexible Attitude Determination and Control System (ADACS) has been implemented such that only a minimum of alterations are required for different missions.

In this paper, we shall explore ADACS design approach for several configuration options.

2. THE MULTI-MISSION BUS CONCEPT

The challenge in designing a single multi-mission bus is in reconciling the terms "standard" and "multi-mission". This is achieved by handling a variety of missions with simple combinations of common, "off-the-shelf" spacecraft elements. By so doing, the danger of designing a large, unwieldy vehicle by trying to simultaneously accommodate all of the potential mission parameters is avoided.

For the spacecraft design in general, and the ADACS design in particular, the two main constraints are the launch vehicle lift capability and the payload orientation and stabilization requirements. A significant number of applications, particularly those related to Earth observation and communications, are geocentric pointing. This requirement dominates the ADACS design. Orbits can be circular, elliptical, highly elliptical with low perigee passages, sun-synchronous, etc. Additionally, once achieved, an orbit may or may not need to be controlled. The selection of sensors/actuators will differ depending on whether a three-axis or a spin stabilized configuration is required. From a careful survey of the possible missions, several configurations have been defined and are now examined:

- With or without propulsion augmentation
- Three axis- vs Spin-stabilized transfer orbit
- Three Axis- vs Spin-stabilized mission
- Earth vs Inertial Pointing

2.1. PROPULSION AUGMENTATION

The level of propulsion capability is determined mostly by the need to supplement the launch vehicle capability and/or to attain and maintain the operational orbit.

If required, the optional spacecraft propulsion system can be provided either in a one 5 lb_f or four 1 lb_f. thruster configuration. If the latter case is selected, propulsion can be used for orbital injection and correction as well as for attitude control and momentum management, thus eliminating the need for magnetic torquers. A secondary benefit provided by the propulsion subsystem is the nutation damping generated by fuel slosh energy dissipation. In the absence of propulsion, a passive ball-in-tube nutation damper may be required.

2.2. LAUNCH SEQUENCE - FROM LAUNCHER SEPARATION TO MISSION CONFIGURATION

Several types of launch scenarios are possible:

1. - Spinning transfer orbit to a spinning mission
2. - Spinning transfer orbit to a three-axis stabilized mission
3. - Continuous three-axis stabilized mode

Figures 1 and 2 depict a launch from an ALV, supplemented by the satellite's own propulsion for injection into the operational orbit. This scenario is of type 2 defined above. The spacecraft is initially spinning prior to launch vehicle separation. Several ground-based spin axis attitude determination methods can be used. The so-called "Cone Intercept" method based on Sun Sensor Assembly (SSA) and Horizon Sensor Assembly (HSA) data is well known and has been successfully applied many times.

Following orbit insertion, the spacecraft is erected on its orbit via the "Dual Spin Turn" (DST, Patented, Ref. 1) which requires simply turning on the Momentum Wheel (MWA). From then on the mission is performed in a 3-axis stabilized mode using a bias momentum. The pitch axis (nominally aligned with the orbit normal) is controlled by modulating the MWA speed, while the other two axes, coupled via the bias momentum, are magnetically controlled. Momentum management is likewise performed by the magnetic torquers.

It is easy to imagine the launch sequences corresponding to a scenario of type 1 or 3. For a continuous spinner, no DST is required, while a continuous three-axis stabilized transfer to orbit saves the weight and complexity of a spinning phase ADACS sensor complement for a three axis stabilized mission.

3. ADACS EQUIPMENT VERSUS MISSION

As briefly discussed in the preceding section, the complement of ADACS sensors and actuators depends upon the mission and/or the availability of propulsion. Figure 3 identifies the sensors and actuators corresponding to a particular mission. Table 1 lists the ADACS components and gives additional design details.

Weights and power consumption vary from mission to mission. The following two cases were taken from typical missions and are given as references:

Configuration	Subsystem Weight	Consumption
Spinner	6.5 kg.	17 watts
3-axis stabilized	17.0 kg.	35 watts

The successful ADACS configurations presented here have evolved from heritage concepts on other GE Astro Space satellite buses. In particular, the attitude acquisition sequence involving the use of SSA/HSA combination and the Dual Spin Turn (DST) scenario are flight proven operations which present little or no risk.

On orbit orientation can alternatively be achieved by the so-called "cross-product law" (Ref. 3). A 3-axis magnetometer senses the direction of the Earth's magnetic field relative to the spacecraft and drives three orthogonal magnetic torquers to align the wheel momentum with the orbit normal.

The "Bias Momentum" configuration in itself is simple, reliable, and has good pitch control performance. It was already the preferred candidate for smallsat applications in a comprehensive survey performed almost twenty years ago (Ref. 4). At the time, the only disadvantage was a not yet fully developed magnetic torque system for momentum dumping and roll/yaw control. Momentum management using magnetics is now well understood and has been used on a series of successful GE Astro Space low Earth orbit vehicles (DMSP, Tiros, Landsat: Ref. 2).

4. ADACS DESIGN APPROACH

Three specific missions were chosen and analyzed to size the Attitude Determination and Control Subsystem. They are representative of the Mission/Payload summary shown on Table 2. The orbit characteristics and the relevant mass properties are listed in Table 3. The spacecraft axis convention and the orbit orientations are illustrated in Figure 4.

4.1 ENVIRONMENTAL DISTURBANCE ANALYSIS

Equations for the aerodynamic, gravity gradient, solar pressure and residual torques were derived for the three missions listed in Table 3.

Mission 1 disturbances are dominated by an aerodynamically induced momentum growth that occurs at each perigee passage. This growth is proportional to the offset between the spacecraft center of mass and center of pressure and can be controlled by proper orientation of the three solar arrays. The solar arrays are feathered (edge-on to the flight path) during perigee passages to minimize the aerodynamic drag on the vehicle.

Profiles of the aerodynamic disturbance torque and momentum build-up corresponding to a 0.75° uncertainty in feathering the arrays are given in Figure 5. Combined gravity gradient, solar pressure and residual dipole disturbances produce much smaller transverse and pitch momentum growth independent of the array angles.

For Mission 2, combined disturbances, dominated by gravity gradient and solar pressure, generate disturbance torque and Momentum accumulation as shown on Figure 6.

For Mission 3 (a spinner), the disturbances are a combined spin-averaged aerodynamic and solar pressure torques. Corresponding profiles are given in Figure 7.

Table 4 summarizes the environmental disturbances for the three study cases.

4.2 ADACS ACTUATORS SIZING

Sizing the control system to meet specifications in the more demanding environment without incurring too severe a penalty in terms of weight and power consumption for the "milder missions" is the goal of a standard design. In the previous paragraph, the environmental torque profiles for the three case missions were described. The actuator sizing now proceeds based on satisfying the mission-specific pointing requirements.

The momentum wheel capacity is determined by the allowable attitude angular error and the vector sum of pitch and transverse disturbance momentum. Table 4 shows that the largest transverse secular momentum growth is 0.91 N-m-second/orbit for Mission 1. The relationship between the wheel momentum H_w, the transverse momentum buildup H_d and the corresponding attitude error θ is:

$$H_w - H_z = \frac{H_d}{\mathrm{Tan}\,(\theta)} \approx \frac{T(0) + \frac{T(2\omega_o)}{3} + \frac{\pi}{2}\sqrt{T_{xI}^2 + T_{yI}^2}}{\omega_o\,\theta} \qquad (1)$$

Where T(0) is the constant body-fixed torque vector predominantly caused by the aerodynamic pressure, $T(2\omega_0)$ is the body-fixed vector containing periodic torques of two times orbit rate and higher predominantly caused by the gravity gradient, Tx_I and Ty_I are the transverse inertially fixed torques predominantly caused by the solar pressure, H_z is the accumulated pitch axis momentum, and ω_o is the orbit rate (rad/sec).

If the Mission 1 attitude error θ is to be kept within 2°, the corresponding H_W requirement is 26 N-m-sec. It is clear that for such a low perigee orbit, keeping the aerodynamic torque small is essential if one wants to avoid the weight penalty of a large bias momentum wheel. On the other hand, Mission 2 could meet a pointing of 0.2° with a 10 N-m-sec. momentum wheel.

The 193 km. Mission 1 perigee was chosen to dramatize the effect of the aerodynamic drag. This drag, proportional to the atmospheric density, changes drastically at about 200 km, and missions with perigee above this altitude can be flown with significantly smaller wheels at comparable pointing performances.

A control algorithm (Patent applied for) has been developed for the SPP vehicle that takes advantage of the otherwise detrimental drag effect . The invention slightly adjusts the individual array angles of attack from nominal to control momentum growth during each perigee passage. Because both the normal and tangential drag components are used, the adjustments are small and the solar array power loss is negligible.

The size of the magnetic torquers is driven by the needs of the roll/yaw control authority and the management of the momentum unloading. A model of the Earth's magnetic field is available either as calculated from the spacecraft ephemeris or measured directly with a magnetometer. The cross-product magnetic torque law can then be used to control the spacecraft momentum and the attitude autonomously. The required torquer dipole M, can be selected by equating the transverse disturbance momentum in one orbit to the control momentum developed using the cross product law in the presence of the Earth magnetic field, B:

$$H_d = C^T \alpha \int_0^{2\pi} C\,[\,\widetilde{M}\,B\,]\,d\beta \qquad (2)$$

Where β is the true anomaly, C is the transformation matrix from the vehicle to the inertial frame, the tilde represents the skew-symmetric matrix, and α is the desired control authority margin.

At an altitude of 850 km., for instance, a 10 Ampere-turn -meter^2 magnetic torquer generates a torque of about 0.15 Newton-meter. This is of the same order of magnitude as the torque delivered by most momentum wheels used to control the pitch axis.

Simulations were performed to determine the attitude control and unloading capabilities using magnetics for various orbits with inclinations greater than 28°. For Mission 2, for instance, momentum dumping of about 0.12 N-m-sec./orbit/Atm^2 is feasible. Thus, unloading the 0.17 N-m-sec. per orbit accumulation about the pitch axis in Mission 2 requires a 2 Atm^2 torquer. Unloading the 0.91 N-m-sec. pitch momentum growth under the same orbital conditions requires a 10 Atm^2 torquer for Mission 1.

Momentum unloading can also be done with thrusters instead of magnetic torquers. A trade-off study was done for Mission 1 indicating that magnetics was preferable if the accumulated momentum exceeded 0.2 N-m-sec./orbit over a one year mission life.

Figures 8, 9 and 10 show simulated SPP yaw-roll attitudes for Missions 1,2 and 3 using the cross-product magnetic unloading law in conjunction with the solar array feathering control algorithm and a suitable estimate of the yaw attitude. Even with the worst-case mission 1 disturbances, the attitude errors are within the desired 0.1 deg goal.

5. CONCLUSION

The GE Astro Space Attitude Determination and Control System (ADACS) for the Space Payload Platform (SPP) is versatile and available in both Spin-stabilized and Three-axis-stabilized configurations. It features simultaneous momentum and attitude control using the vehicle bias momentum together with magnetic torquers and a solar array orientation control system.

Minimal modifications are required from mission to mission, most of them being in the controller firmware and thus easily implemented. The system supports launch from an ALV or a SSLV and can be used with or without propulsion augmentation depending upon the mission.

REFERENCES

1) U. S. Patent 4,275,861, " Orientation of Momentum Stabilized Vehicles," C. H. Hubert, June 30, 1981.

2) A. Craig Sticker/T. Alfriend," An Elementary Magnetic Attitude Control System," AIAA Paper # 74-923, 1974.

3) J. Berg, "DMSP S-15 Autonomous Momentum Dumping," GE ASTRO SPACE Internal Correspondence, 5D-ADC-302, June 1988.

4) Alfred I. Sibilla et al, "The Role of Small Satellites in Space Applications Missions," Journal of the British Interplanetary Society - Vol. 24 1971.

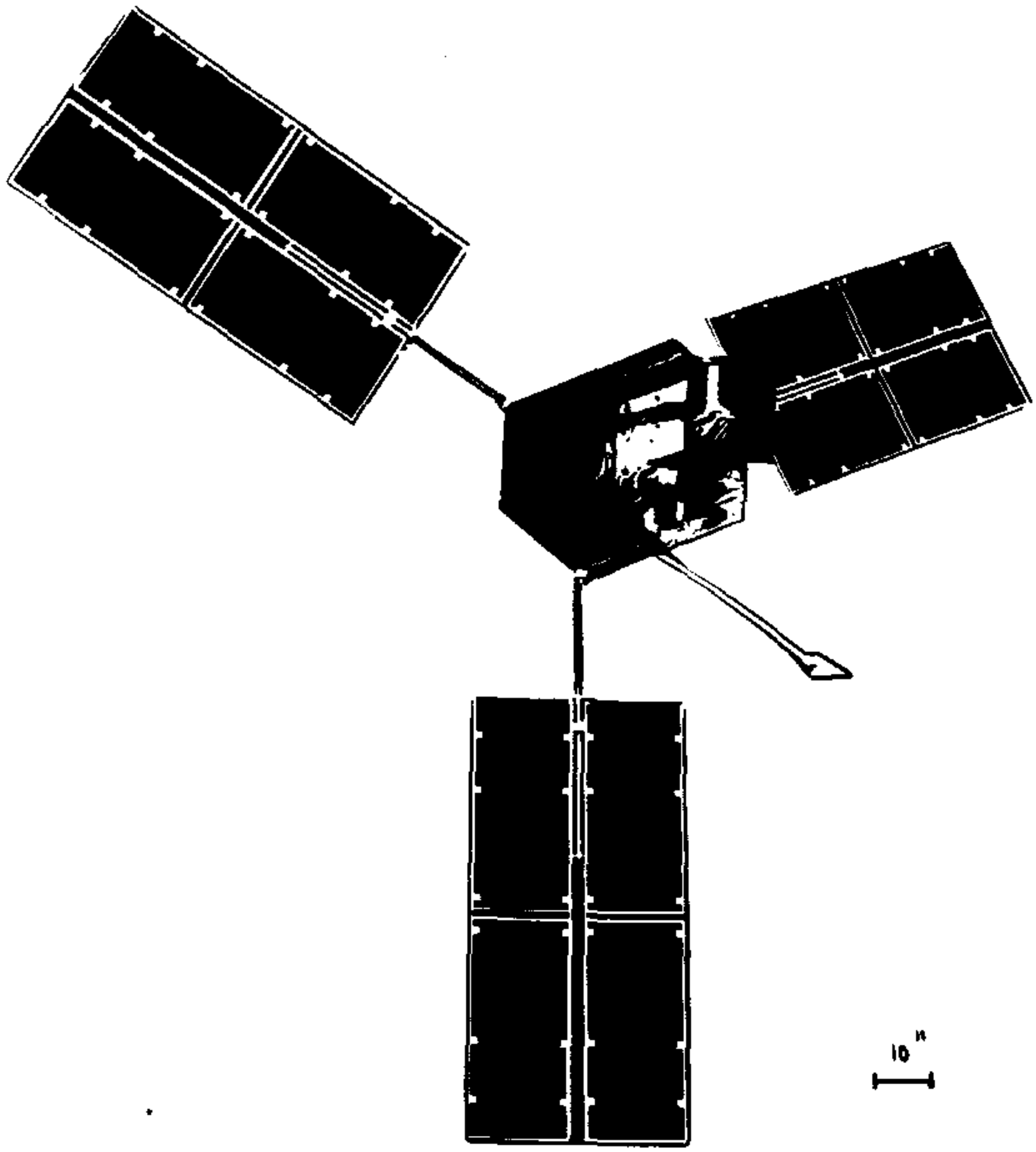


Photo 1. Space Payload Platform (SPP) in Operational
orbit configuration

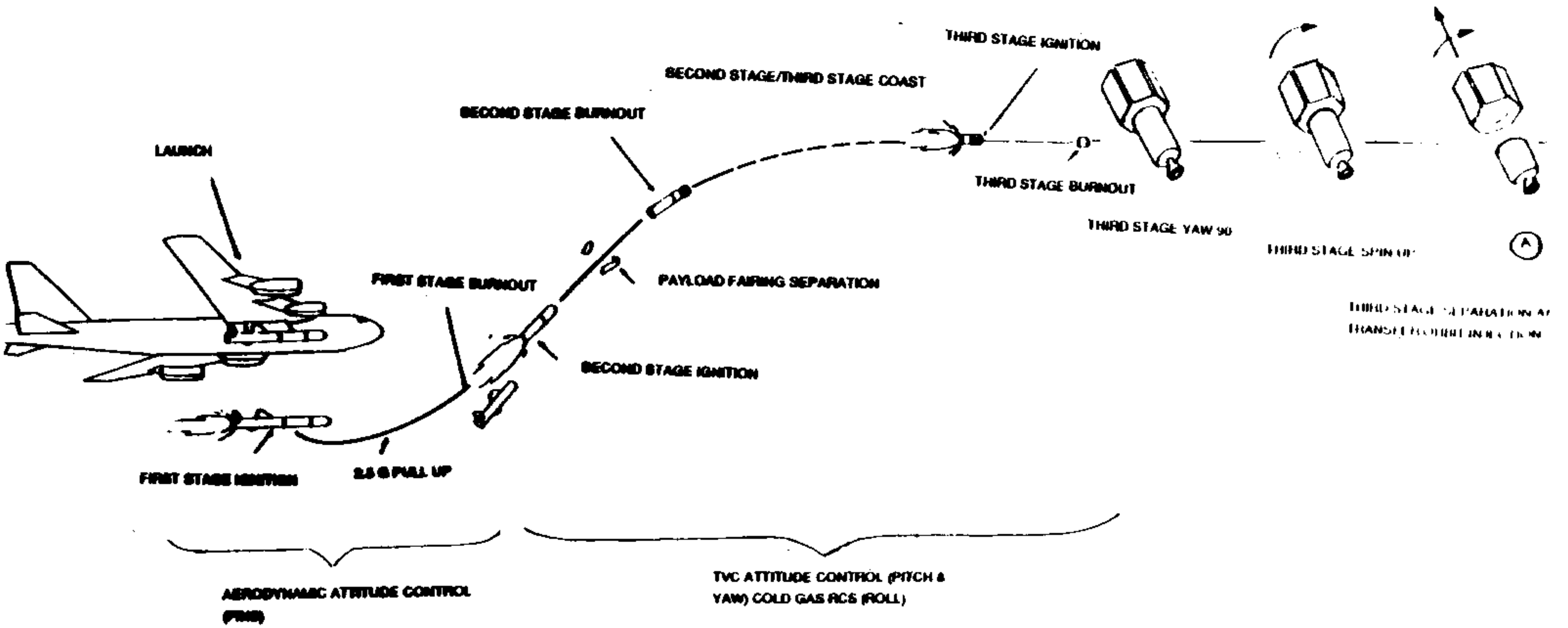
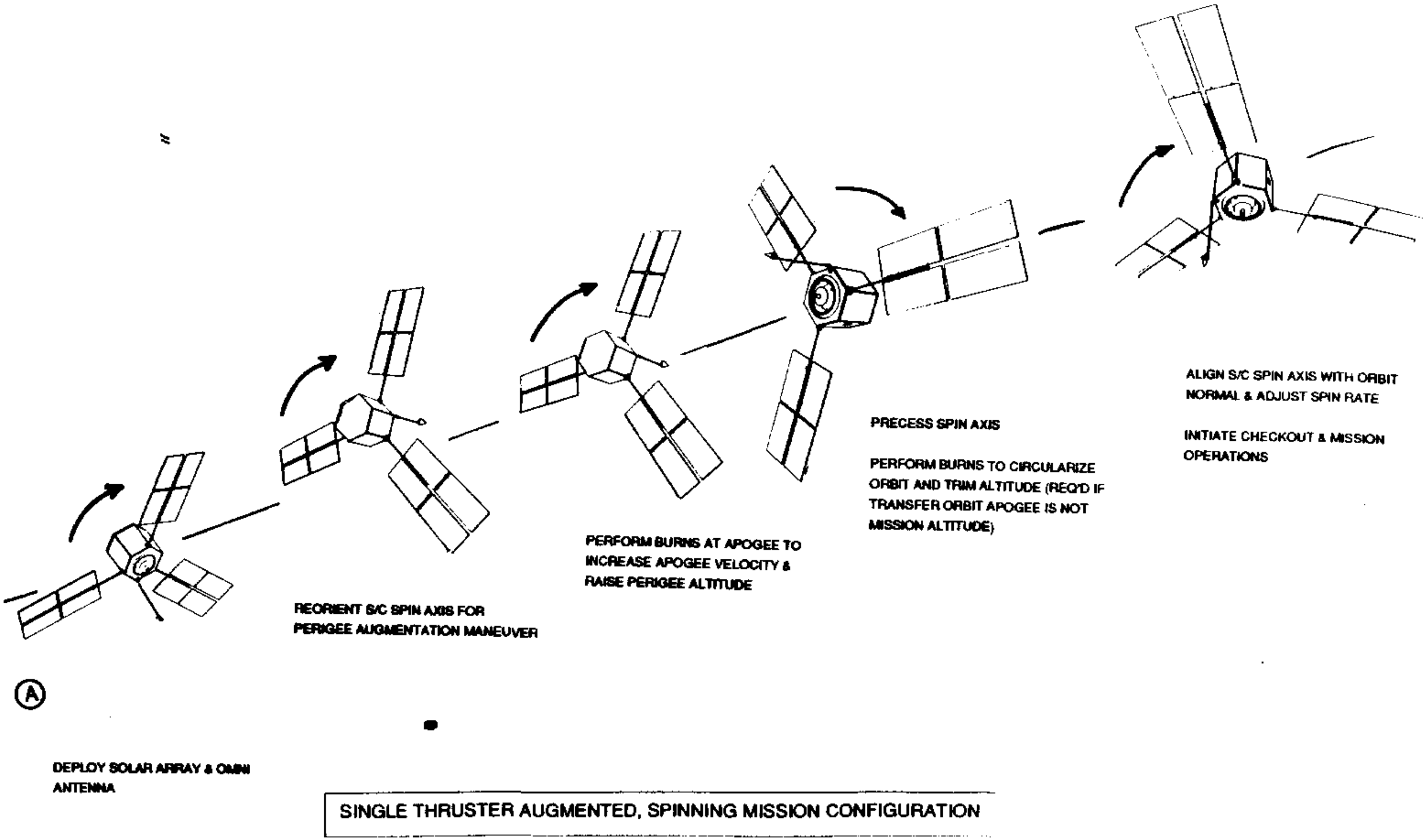


Figure 1. LAUNCH SEQUENCE



DEPLOY SOLAR ARRAY & OMNI ANTENNA
REORIENT S/C SPIN AXIS FOR PERIGEE AUGMENTATION MANEUVER
PERFORM BURNS AT APOGEE TO INCREASE APOGEE VELOCITY & RAISE PERIGEE ALTITUDE
PRECESS SPIN AXIS
PERFORM BURNS TO CIRCULARIZE ORBIT AND TRIM ALTITUDE (REQ'D IF TRANSFER ORBIT APOGEE IS NOT MISSION ALTITUDE)
ALIGN S/C SPIN AXIS WITH ORBIT NORMAL & ADJUST SPIN RATE
INITIATE CHECKOUT & MISSION OPERATIONS
SINGLE THRUSTER AUGMENTED, SPINNING MISSION CONFIGURATION

Figure 2.

FIGURE 3 - ADACS SENSORS AND ACTUATORS FOR DIFFERENT MISSION PLANS

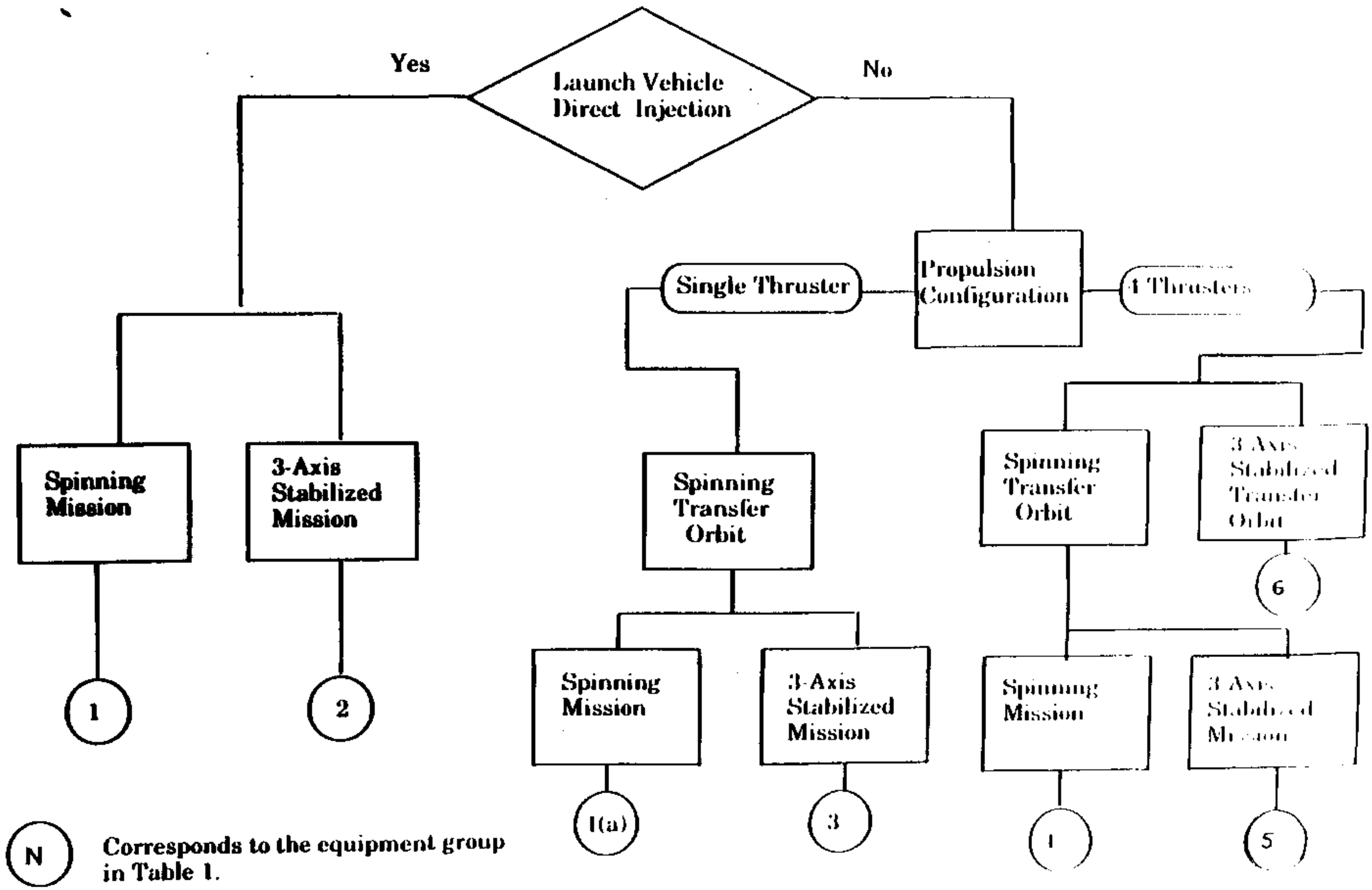
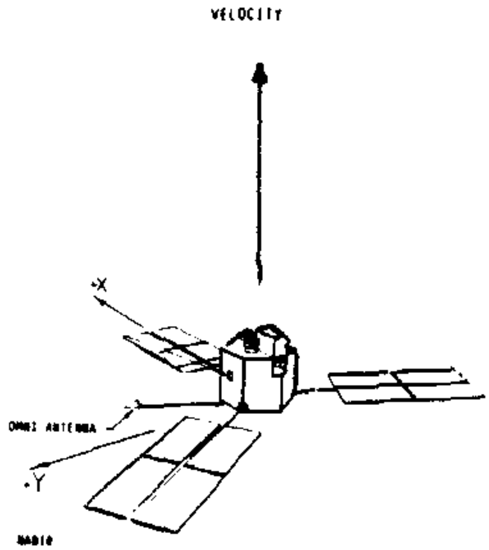
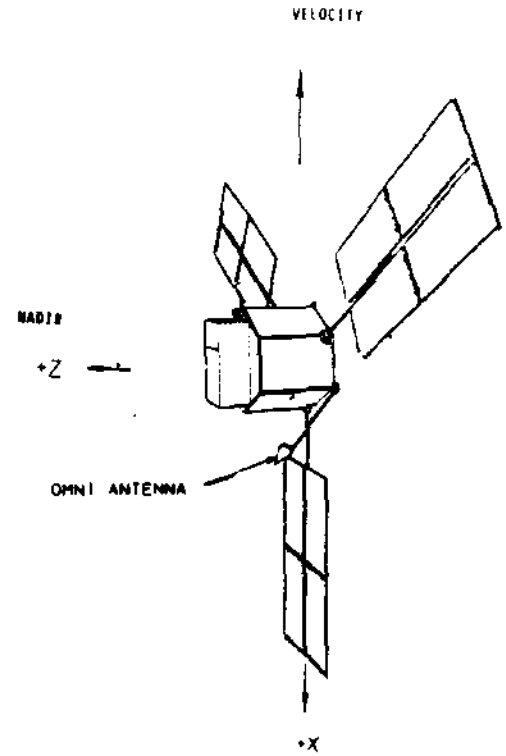


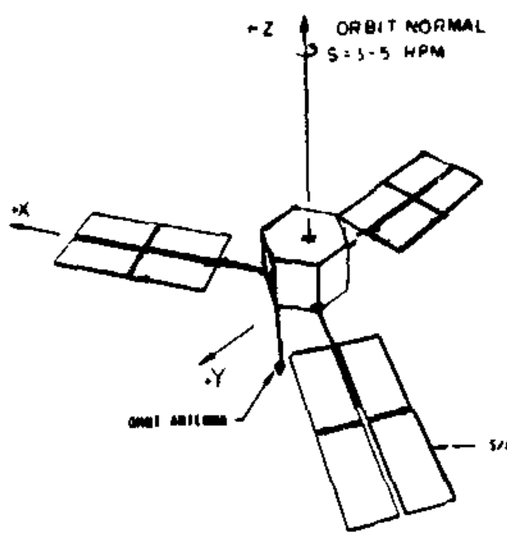
FIGURE 4
ORBITAL CONFIGURATIONS
FOR MISSIONS ONE, TWO AND THREE



MISSION 1.



MISSION 2.



MISSION 3.

TABLE 1 - ADACS HARDWARE

Remarks
1) Altitude knowledge required with one CES. 2 CES's required for an elliptical orbit.
2) 1 Thruster Configuration.
3) 4 Thruster Configuration.
4) Spinning Transfer Orbit
5) 3-Axis Transfer Orbit.
6) CES used as HSA in static mode.
7) Launch Vehicle Direct Injection.

Equipment Group (See Fig. 3)	Sun Sensor Assembly (SSA)	Horizon Sensor Assembly (HSA)	Magnetic Torquers Pitch and Roll-Yaw	Magnetometer	Nutation Damper	Conical Earth Scanner (CES)	Momentum Wheel Assembly (MWA)	Spin Stabilized Mission	3-Axis Stabilized Mission	Remarks 1	2	3	4	5	6	7
1	X	X	X	O	X			X								X
1(a)	X	X	X	O	X			X			X		X			
2			X	O	X	X	X		X							X
3	X	X	X	O		X	X		X	X	X		X		X	
4	X	X						X				X	X			
5	X	X				X	X		X	X		X	X		X	
6						X	X		X	X	X			X		

Note: X = Required for the indicated ADACS sensors / actuators group shown in Figure 3.
O = Optional equipment - may be replaced by an Earth magnetic field model and ephemeris, depending upon the accuracy / autonomy required.

TABLE 2 - MISSION PAYLOAD SUMMARY

	Meteorology	Navigation	Communication	Surveillance
Payload	AVHRR	Doppler System	Multi-Channel EHF	Various
Orbit	850 km Circular Sun- Sync.	1000 km Circular Polar	Geosynchronous Molniya Low Earth Orbit	Various
Orientation	Nadir Pointing	Nadir Pointing	Nadir Pointing	Nadir Pointing
Attitude Control	3-Axis Bias Momentum	3-Axis Bias Momentum	3-Axis Bias Momentum or Spinner	3-Axis Bias Momentum

TABLE 3
CASE STUDIES - ORBIT PARAMETERS AND MASS PROPERTIES

Parameter	Mission 1	Mission 2	Mission 3
Orbit Altitude (km)	193 x 1600	750	500 x 1000
Inclination (deg)	90	90	90
Total S/C BOL Weight (kg)	289	312	333
Stabilization	3-Axis	3-axis	Spinning
Inertias BOL (kg-m ^{2})			
Ixx	329	329	349
Iyy	328	328	328
Izz	314	325	749
Maximum Frontal Area (m^2)			
Core	1.1	0.578	0.572 [1]
Core + Arrays	7.1	4.0	2.85
Maximum Center of Mass /Center of Pressure Offset (m)	0.221 [2]	0.353	0.353
Maximum Dynamic Pressure (N/m ^{2})	0.043	1.2×10^{-6}	6.2×10^{-4}

Notes:

1) Mission 3 frontal areas are spin-averaged.
2) This offset is equivalent to a 0.75 deg. solar array error with the arrays feathered through Perigee (ie. frontal area = 1.1 m^2)

TABLE 4
WORST CASE ENVIRONMENTAL DISTURBANCE SUMMARY

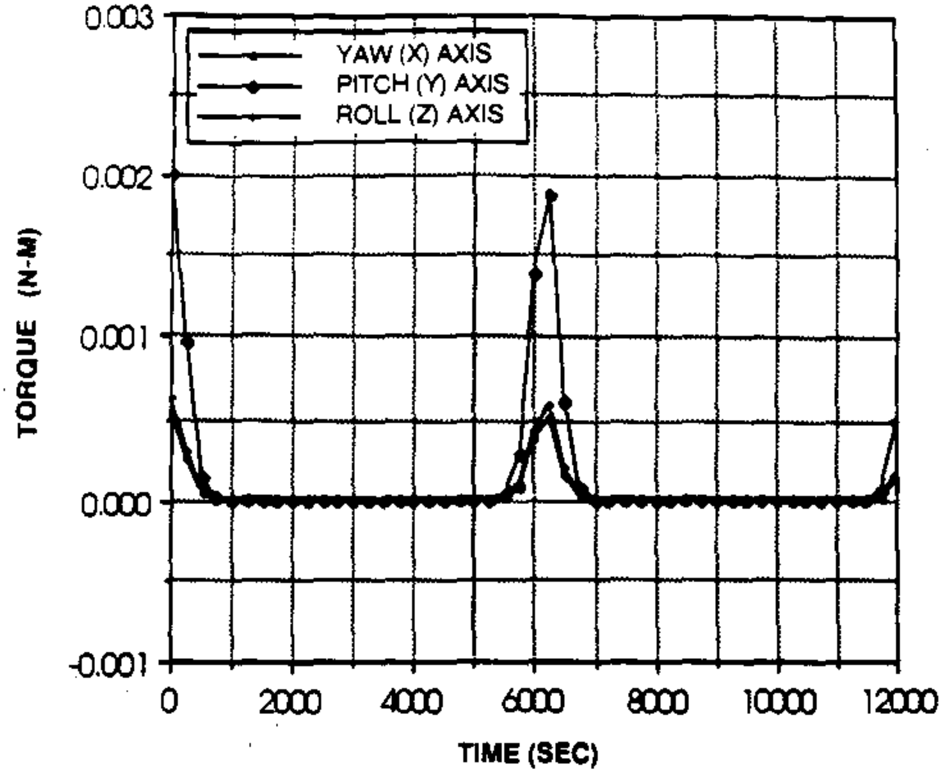
	Peak Torque (μ-N-m)			Maximum Cyclic Momentum Zero-Peak (milli-N-m-s)			SecularMomentum Growth (milli-N-m-sec/orbit)		
	X	Y	Z	X	Y	Z	X	Y	Z
Mission 1 [1]	590	660	2100	8	0.002	8	200	270	910
Mission 2	6.2	27	0.002	7	0.001	7	1.5	170	1.5
Mission 3 [2]	0.6	0.6	0.7	0.1	0.1	0.1	2.3	2.3	4.0

Notes:

1) Calculated with a 0.75 deg. worst case combination solar array error.

2) For mission 3, the Z-axis is the spin (orbit normal) axis with a nominal 3 RPM rate.

FIGURE 5
MISSION ONE DISTURBANCES



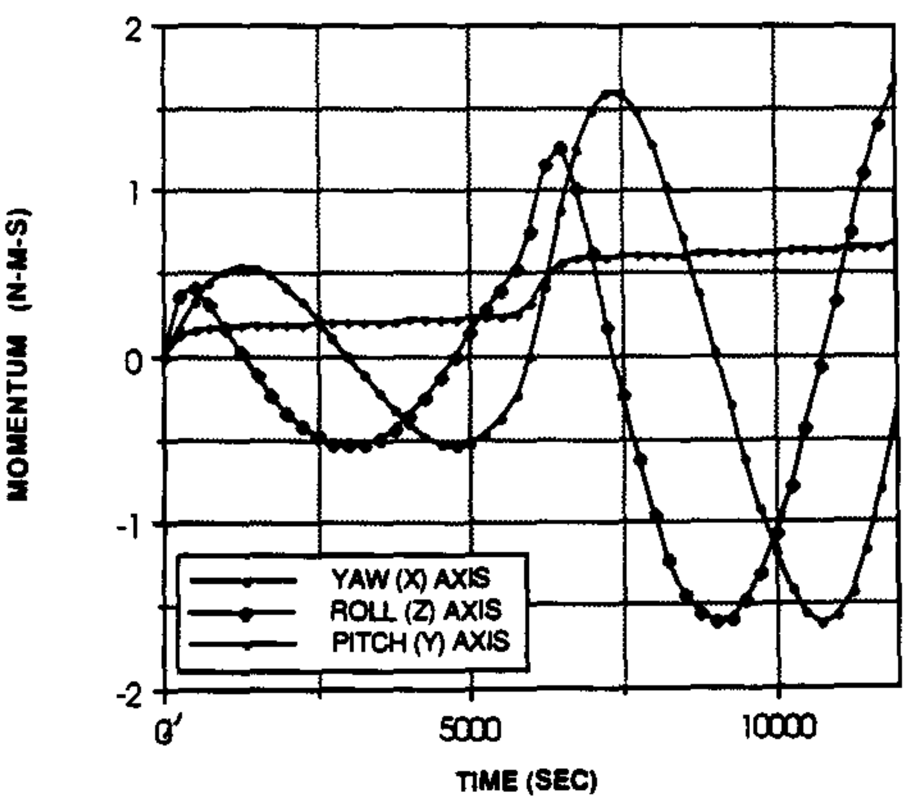
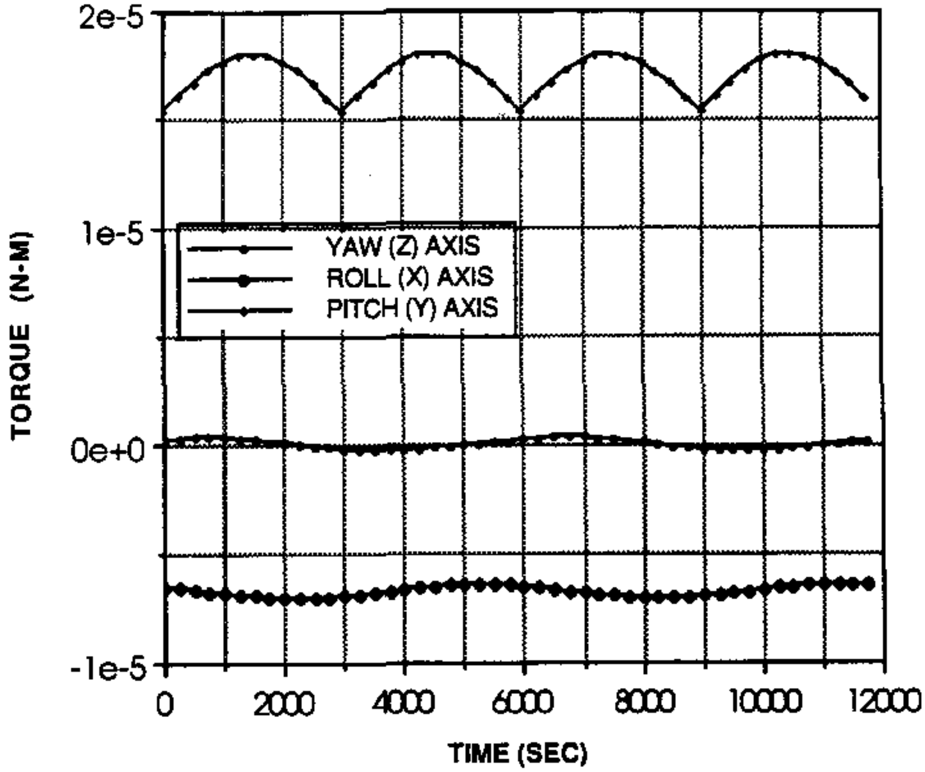


FIGURE 6
MISSION TWO DISTURBANCES



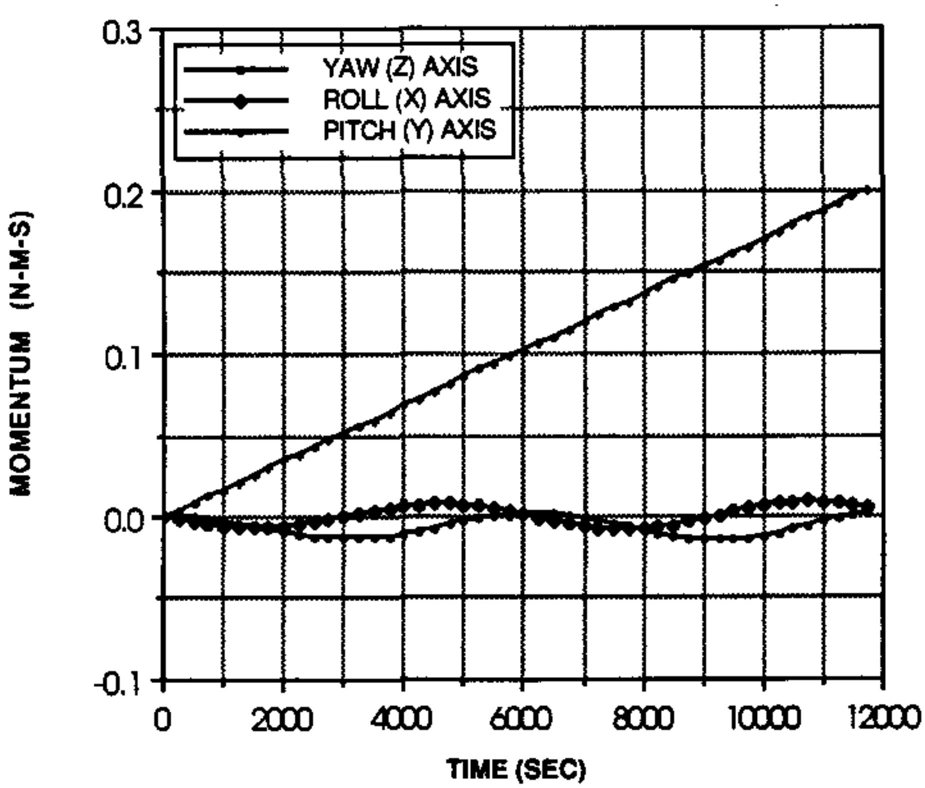
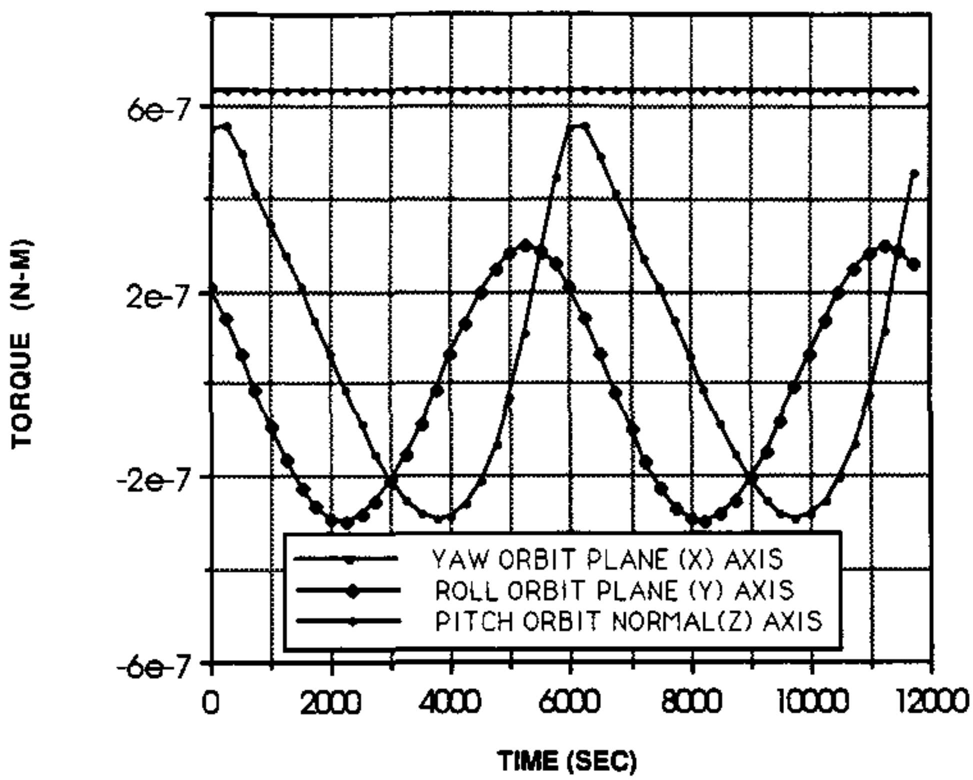


FIGURE 7
MISSION THREE DISTURBANCES



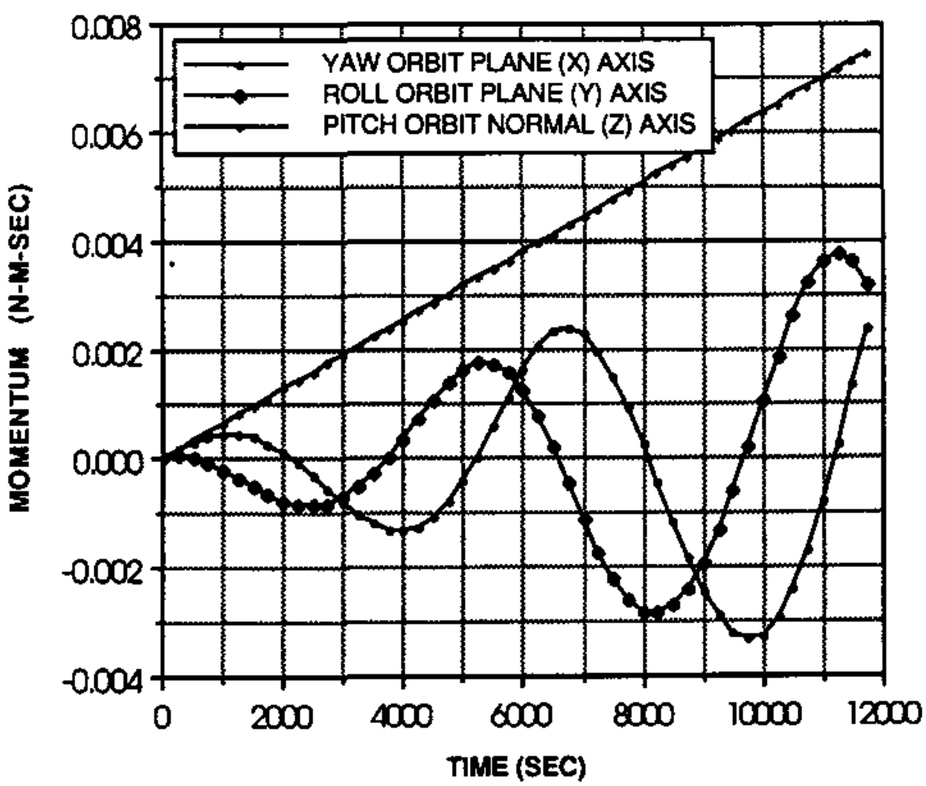


FIGURE 8 MISSION 1 ROLL-YAW CONTROL

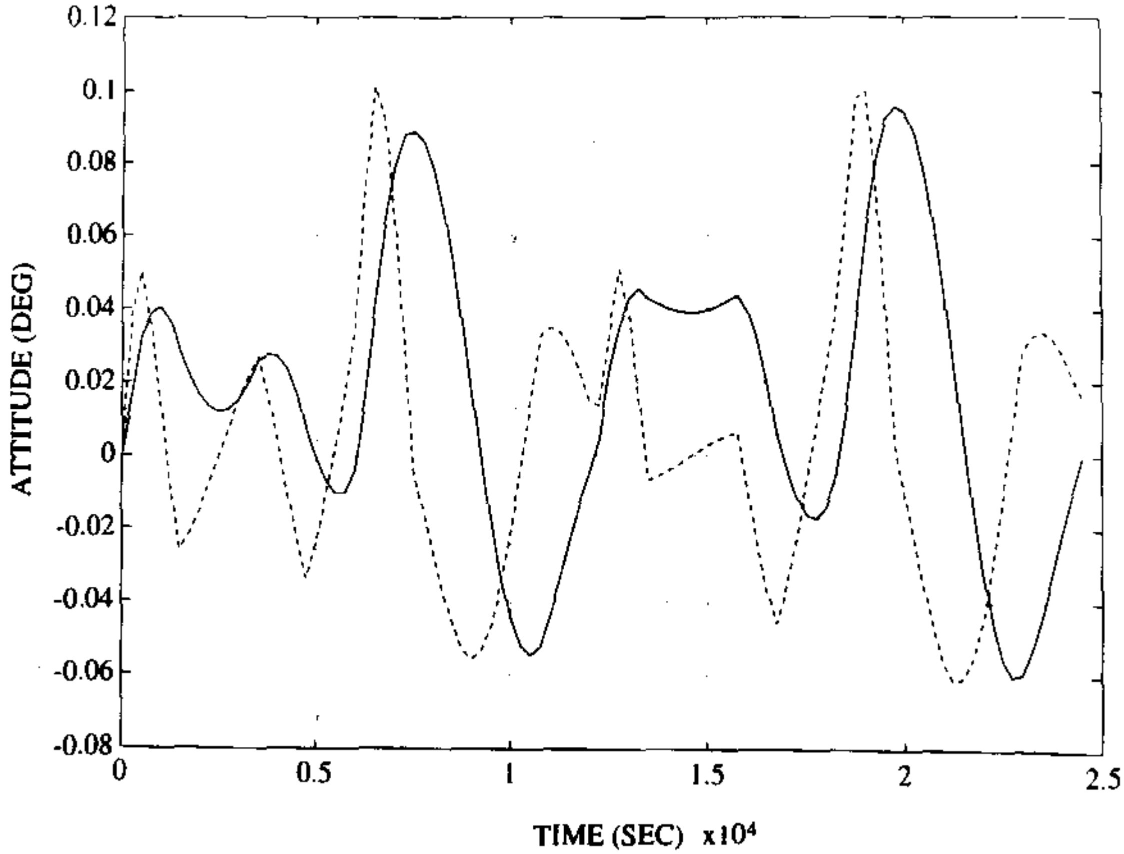


FIGURE 9 - MISSION 2 ROLL-YAW CONTROL

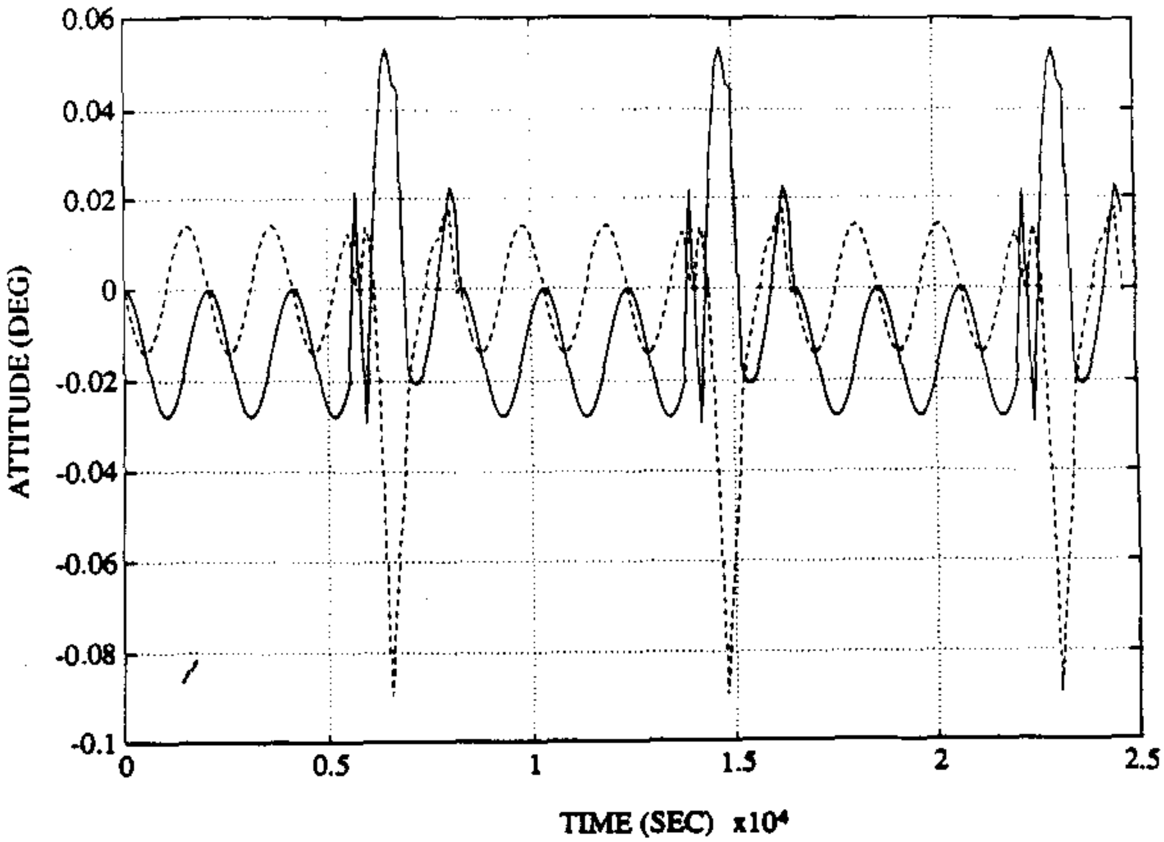


FIGURE 10 - MISSION 3 ROLL-YAW CONTROL