AMSAT Phase IV Propulsion and Attitude Control Systems Conceptual Design

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Amateur Radio Satellite Corporation (AMSAT) is developing a satellite designated Phase IV for operation in geosynchronous orbit. Conceptual development of the primary propulsion system and the attitude and velocity control system (AVCS) is described with an approach outlined for hardware selection. A launch opportunity in Ariane will provide insertion into a transfer orbit and spin stabilization. The primary propulsion system, which will provide the orbit insertion burn only, will be designed around a Messerschmitt-Bolkow-Blohm (MBB) 400 newton engine. Two subsystems will comprise the AVCS, a momentum storage system for countering small external perturbations to maintain attitude control and a reaction control system (RCS) for satellite maneuvering and momentum system despinning. Methods for sizing momentum system components, thrusters and propellant tanks are presented, and preliminary mass property data is generated in examples to demonstrate how parameters of these two systems are interrelated.

<u>**Introduction**</u>. The Phase IV AMSAT satellite is being designed to be boosted into a Geosynchronous Transfer Orbit (GTO) and spin stabilized as a secondary payload on an Ariane launch vehicle. Primary propulsion for the apogee impulse to insert the satellite into Geosynchronous Equatorial Orbit (GEO) will be provided by a 400 Newton engine manufactured by Messerschmitt-Bolkow-Blohm (MBB). Two other systems will provide station keeping and attitude control services for the on-orbit life of the satellite. These are a momentum storage system for fine attitude control and a thruster Reaction Control System (RCS) for course attitude control and station keeping. This paper presents preliminary design considerations for and interactions between these systems.

Small angular corrections about the three primary axes will be made as a reaction to torque inputs to three rotating masses with spin axes parallel to these three axes. By using momentum wheels, many external attitude perturbations can be compensated for without the consumption of RCS propellant, extending satellite service life. The RCS will have four basic functions, insertion burn trim, spacecraft despin following the insertion burn, station keeping and periodic course attitude control for momentum wheel despinning. The momentum wheels and RCS thrusters are interactive, and each will influence the design of the other. One objective of this paper is to present relationships that will lead to criteria and boundary conditions for selecting hardware. Consideration of all maneuvers and events which involve these two systems provides a framework for developing these relationships.

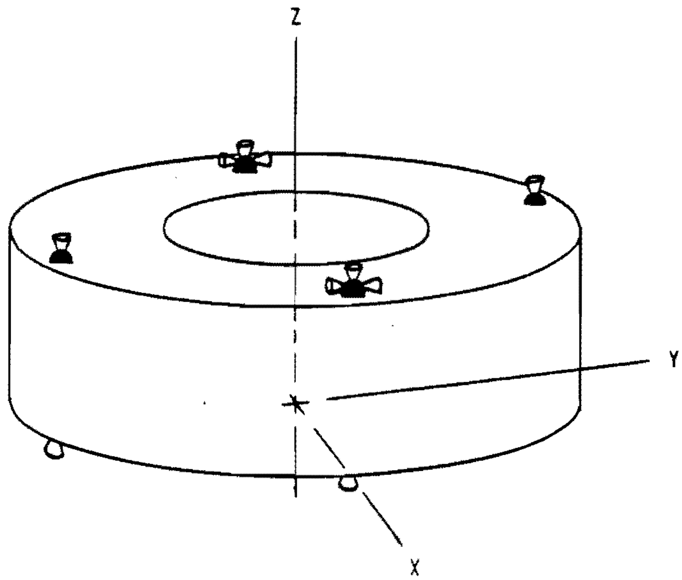

Figure 1. Phase IV Satellite Axis and AVCS Thruster Orientations.

Baseline Spacecraft. The Phase IV spacecraft is in a preliminary design phase. It will fill a cylindrical volume 2.26 meters in diameter and one meter high as constrained by the space allocated within the launch vehicle. Figure 1 shows the general shape, principal axes and orientation of RCS thrusters. The momentum storage system and RCS tanks/plumbing will be internally mounted. Only a few simple parameters that represent the total mass of the spacecraft are needed for this analysis. Assuming that the total mass at MBB burnout is 247 Kg and that the mass is uniformly distributed, the appropriate mass properties can be estimated. Actual properties will be computed and updated regularly as the design matures, and this analysis will be updated as well. Given this spacecraft model, the moments of inertia are:

$$I_{xx} = 73.76\ \text{ft}\cdot\text{lb}\cdot\text{sec}^2$$

$$I_{yy} = 73.76\ \text{ft}\cdot\text{lb}\cdot\text{sec}^2$$

$$I_{zz} = 116.98\ \text{ft}\cdot\text{lb}\cdot\text{sec}^2$$

GEO Insertion Burn. the transfer orbit will be controlled by the Ariane. Published general performance data for the GTO when launched from Kourou, French Guiana lists the following parameters:[1]

Perigee altitude = 200 Km (107.9 n.mi.)
Apogee altitude = 35,789 Km (19312 n.mi.)
Inclination = 7.0 degrees

The velocity vector diagram which solves the problem of circularizing the orbit and inserting the spacecraft into GEO is drawn in Figure 2.[2]

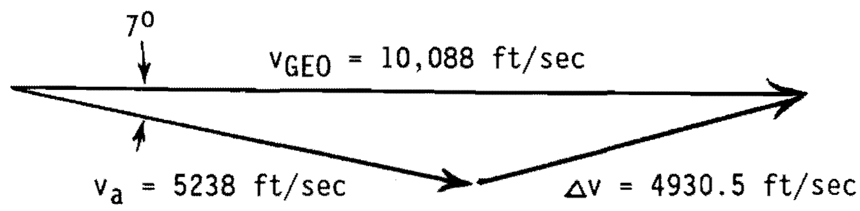

Figure 2. Velocity Vector Diagram for GEO Orbit Insertion.

Thus a Δv of 4930.5 ft/sec (1502.8 m/sec) must be provided by the MBB engine. The quantity of propellant required is found by use of the rocket equation.[3]

$$\Delta v = I_{sp} g_c \ln(m_o/m_b) \tag{1}$$

Assuming an initial mass (m_o) for the spacecraft of 400 Kg, and a specific impulse value of 319 seconds, the mass at burnout (m_b) is found to be 247 Kg. Propellant consumed will be 153 Kg (337.3 lb_m) or 38.25 % of the initial mass.

<u>Delta v Trim</u>. At burnout of the MBB engine some orbital drift may be desirable to move the satellite to a predetermined location over the equator. At that point the RCS thrusters will be required to adjust the velocity to that required for GEO. Some off-nominal velocity is to be expected due to main propulsion I_{sp} dispersions, propellant depletion uncertainty or any of several other conditions which could cause Δv uncertainty. At this point thruster selection is practically unconstrained, but propellant consumed is a function of I_{sp} rather than thrust. Very small thrusters will be selected for calculation purposes (compatibility with the momentum system will be shown to dictate small thrusters).

Assume:	Thrust of each	= 0.2 lb
	I_{sp} of each	= 200 $lb_f \cdot sec/lb_m$
	Correction required	= 1.0 % of Δv

Equation 1 gives the RCS system propellant needed to provide the required Δv of 49.3 ft/sec.

$$m_b = 245.115 \text{ Kg}$$

$$\Delta m = 1.885 \text{ Kg } (4.156 \text{ lb}_m)$$

<u>Spacecraft Despin Maneuver</u>. One thruster pair that will produce torque about the z axis in the proper direction will be required to despin the spacecraft after the MBB insertion burn. If 0.2 pound thrusters are used as in the model described above, they will produce a torque of 17.794 in·lb for the despin maneuver. Assuming that the spin rate is 30 rpm, the time and propellant required to reduce it to zero can be determined from a

torque relationship and the thruster specifications.[4]

$$\text{Torque} = I_{zz}\dot{\omega}_z = I_{zz}(\Delta\omega/\Delta t)$$

$$\text{At 30 rpm,}\ \omega_z = 3.14\ \text{rad/sec}$$

$$\Delta t = (116.98 \times 3.14)/\text{torque} = 247.7\ \text{sec}$$

Choosing the same value for I_{sp} as before, 200 seconds,

$$\text{Mass flow rate, } \dot{m} = \text{Thrust}/I_{sp} = 0.001\ lb_m/\text{sec}$$

per thruster.

$$\text{Propellant consumed, } \Delta m = \dot{m}(\Delta t) = 0.495\ lb_m = 1.09\ \text{Kg}$$

for the spacecraft despin maneuver.

East-West Station Keeping. An additional task for the attitude control system is maintaining the position of the Phase IV satellite in geosynchronous orbit. Only east-west positioning errors will be corrected. The spacecraft design and normal on-orbit orientation (z-axis perpendicular to the plane of the equator) require that a 90 degree rotation be made to position the satellite such that the four thrusters on the -z face can fire in either an easterly or a westerly direction. Assuming no spin stabilization, attitude control during thrusting will depend upon balanced thrust. Any imbalance must be corrected by modulating thrust.

Duration of thrusting and propellant consumption will depend upon how rapidly a correction is to be made. For the purpose of an example calculation, it is assumed that a position drift rate of one degree in four days will be satisfactory. One impulse will be required to begin the drift and another to stop it. In order to drift toward the east, the first firing must be in a westerly direction followed by an easterly thrusting when the satellite has drifted into position. The effect is to transfer into an orbit with a shorter period such that each subsequent time the satellite is at apogee, it will be over the equator a little east of the previous point.

Orbital period is a function of a, the semi-major axis.[2]

$$T_p = 2\pi\sqrt{a^3/\mu}$$

For a circular orbit with a period of one sidereal day (23 hr 56 min 4 sec),[2]

$$a = r = 22{,}752\ \text{n. mi.}$$

To move toward the east a drift orbit should be entered with a period of

$$T_{pd} = 23.9344(1439/1440) = 23.9178\ \text{hrs.}$$

or

$$a_d = 22741.8\ \text{n. mi.}$$

Delta v for each transfer is the difference between the baseline circular orbit velocity and the velocity at apogee in the drift orbit.[2]

$$v_{circ} = \sqrt{\mu/r} = 10{,}087.6 \text{ ft/sec}$$

$$v_{ad} = \sqrt{\mu(2/r - 1/a)} = 10{,}084.8 \text{ ft/sec}$$

$$\Delta v = 2.8 \text{ ft/sec}$$

Assuming a specific impulse of 200 seconds, Equation 1 gives the propellant required

$$m_1/m_2 = 1.000435$$

For a 245 Kg satellite,

$$\Delta m = 0.10876 \text{ Kg} = 0.240 \text{ lb.}$$

Using the same mass flow rate as above with four 0.2 lb thrusters firing together, the duration of the firing must be 60 seconds. An equal Δv (another 0.24 lb of propellant) will be required to stop the drift when the satellite reaches the desired position. Thus, 0.48 lb of propellant will be required for each such correction.

Satellites in geosynchronous orbit will drift toward one of two null points. The drift correction that will be required depends upon the desired position in the orbit. Seven feet per second per year is a reasonable drift correction, and the rocket equation gives propellant required. For ten years, an I_{sp} of 200 seconds and a 245 Kg satellite,

$$\Delta m = 2.65 \text{ Kg} = 5.84 \text{ lb.}$$

Baseline Momentum System. Honeywell data is available on momentum wheels used in several small satellite designs. Since these have been developed and are in production, the data will provide a good baseline.[5]

Table 1. Momentum Wheel Performance Data and Specifications

Application	Angular Momentum (ft-lb-sec)	Wheel Speed (RPM)	Weight (lb)	Outside Diameter (inches)	Output Torque (oz-in)
DSCS III	1.4	2525	5.2	9.25	6.5
TDRSS	17.0	3100	19.0	12.6	8.0
HEAO	30.0	2000	30.0	14.1	17.0

The system would be configured with each momentum wheel mounted with its spin axis parallel to a principal spacecraft axis. Data for three existing wheels are listed in Table 1. For the purpose of demonstrating calculations, the data for the wheel used on the TDRSS will be used.

At rated angular momentum and speed (17 ft·lb·sec, 3100 rpm) the moment of inertia can be calculated:

$$I_W = \text{Mom.}/\dot{\omega}_W = 0.0524 \text{ ft}\cdot\text{lb}\cdot\text{sec}^2$$

__Spacecraft-Momentum Wheel Interaction__. Perturbing torques on the spacecraft will be compensated for by torque inputs to the reaction wheels. Torques in opposite directions will cancel, and the momentum wheels will be able to compensate for all external torques within the saturation limits. Momentum wheels are designed and operated such that some nominal spin rate, say 85% of rated, is the unsaturated rate, and the wheel becomes saturated at +10% and - 10% of that value. When either saturation limit is reached, the wheel is torqued to the nominal value while the spacecraft is given a torque of equal magnitude by the RCS thrusters. In the baseline system 3100 rpm represents 100%, and the wheel is unsaturated at 2635 rpm and saturated at 2325 and 2945 rpm.

Total system momentum is conserved in any maneuver, and the sum of the momentum for the spacecraft (subscript s) and for the momentum wheel (subscript w) will remain constant.[4]

$$I_S \omega_S + I_W \omega_W = \text{constant} \tag{2}$$

Since the values of I are essentially constant, the torque given up by one system and received by the other will be equal in magnitude (opposite in direction).

$$I_S \dot{\omega}_S = -I_W \dot{\omega}_W \tag{3}$$

The maximum accumulated torque on the spacecraft that can be stored by a wheel at its rated value can be determined by using equation 2. For the model previously described,

$$I_S \omega_S = 17 \text{ ft}\cdot\text{lb}\cdot\text{sec}$$

The difference between the wheel at saturation and nominal spin corresponds to a spacecraft rotation about the z axis of

$$\omega_S = (I_W/I_S)\ \omega_W = 0.0145 \text{ rad/sec}$$

Table 1 gives the minimum output torque on the motor which drives a wheel as eight ounce inches. Choosing 0.2 lb thrusters, thruster generated torque may be compared with motor torque. If a pair of thrusters fire to generate a moment, and they are located 1.13 meters from centerline,

$$\text{Torque} = 0.2(2.26)(39.37) = 17.794 \text{ in}\cdot\text{lb}$$

or 35.6 times the motor torque. It isn't necessary that the two torque values match. They can effectively be equalized by thrusting such that the areas under the torque-time plots are equal. This would be accomplished by running the torque motor continuously during the maneuver while the thrusters are pulsed at the proper interval to give the correct thrust-time result.

To determine the quantity of propellant consumed in the despin maneuver, let I_{sp} = 200 seconds

$$\text{Mass flow rate} = 0.2/200 = 0.001\ \text{lb}_m/\text{sec per thruster}$$

The time required through which the external torque must be applied can be determined from equation 3,

$$\Delta t = (I_s \cdot \Delta\omega)/\text{torque} = 1.144\ \text{sec}$$

The propellant consumed is

$$\Delta m = (0.002)(1.144) = 0.00228\ \text{lb}_m = 0.00504\ \text{Kg}$$

for each despin maneuver.

An analysis of how much accumulated torque is to be expected over the life of the spacecraft would indicate how many despin maneuvers would be needed and how much propellant would be required for this purpose. All three axes must be considered. A characteristic of pressure blow-down systems is a decay of thrust and I_{sp} over the life of the program. This will be partially offset by a very small reduction in the moments of inertia due to propellant consumption. However, the result will be a gradual increase in the propellant required for each despin event as system pressure decays.

<u>Total Propellant Requirement Analysis</u>. Propellant consumed during the life of the satellite can be calculated when information about drag and external perturbing forces is known. For purposes of a preliminary analysis, it will be assumed that over the ten year life the equivalent of 500 z-axis momentum wheel despin events will be required and 10 orbital position correction maneuvers will be required in addition to a total drift correction equal to 70 ft/sec. This provides a mission baseline for assessing total propellant requirements allowing specific impulse to vary to show the sensitivity of propellant required to specific impulse in the AVCS. All other values are as stated or assumed in the preceding sections. Results are listed in Table 2.

Actual quantities of propellant and the associated tank sizes required will be refined as the AMSAT Phase IV satellite design progresses. The selection of the type of AVCS thrusters is an open issue at this writing, but the computations show general trends. A relatively simple and inexpensive cold gas system in the lower range of I_{sp} values of Table 2 would require two to three times the propellant of a more expensive system in the higher range of values. Momentum system mass is a significant portion of total spacecraft mass and a more favorable selection than the example described may well be available. The methodology presented is intended to show the interaction of design features and to describe a model to be used for hardware selection as the satellite design matures.

Table 2. Example Satellite Propellant Consumption Requirements for Various Events or Maneuvers.

AVCS I_{sp} (sec) =	75	100	200	300
Mass (Kg)				
(Initial Mass)	400	400	400	400
Insertion Burn	153	153	153	153
AVCS Propellant Required	27.3	20.6	10.3	6.9
Allowance for Δv Trim	5.0	3.8	1.9	1.3
Satellite Despin	2.9	2.2	1.1	.7
Orbital Position Correction	5.7	4.3	2.2	1.4
Drift Correction	7.0	5.3	2.6	1.8
Momentum Wheel Despin	6.7	5.0	2.5	1.7
(Dry Satellite Mass)	219.7	226.4	236.7	240.1

REFERENCES

1. Arianespace. Ariane 4 User's Manual, Issue 1, Revision 4. Evry Cedex, France. 1988.

2. BATE, R. R., D. D. MUELLER and J. R. WHITE. Fundamentals of Astrodynamics. New York: Dover Publications, 4 May 1984.

3. HILL, P. G. and C. R. PETERSON. Mechanics and Thermodynamics of Propulsion. Reading, MA: Addison-Wesley, 1970.

4. Space Handbook, Revision 11. Air University, Maxwell Air Force Base, Alabama. August 1977.

5. Honeywell. "Momentum Wheel and Reaction Wheel Assemblies." Brochure S61-1720-05-01. May 1988.

NOTATION

a	orbit semi-major axis
g_c	Earth gravitational constant, 32.174 $(lb_m \cdot ft)/(lb_f \cdot sec^2)$
I	moment of inertia
I_{sp}	specific impulse
m	mass
$\dot{m}$	mass (propellant) flow rate
r	radius, orbital
t	time
T_p	period of an orbit
v	velocity
μ	Earth gravitational parameter, $1.407654 \times 10^{16}\ ft^3/sec^2$
ω	angular velocity
$\dot{\omega}$	rate of change of angular velocity

Subscripts

a	apogee
b	burnout
d	drift orbit
GEO	geosynchronous equatorial orbit
s	spacecraft
w	momentum wheel
x, xx	x axis reference
y, yy	y axis reference
z, zz	z axis reference
0	initial condition
1	first increment
2	second increment