ATTITUDE CONTROL OF A SMALL SPACECRAFT IN AN ELLIPTICAL ORBIT

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A small research satellite is being developed by the Royal Aerospace Establishment, UK, for the purpose of in-orbit technology research. The satellite is planned for launch on an Ariane 4 mission into the severe environment of a geostationary transfer orbit in which it will remain. This paper addresses the challenges which have arisen in designing an attitude control system for a satellite in this unusual orbit. The paper includes discussion of how recovery from an initial tumbling state is to be performed, why spin stabilisation has been selected, how magnetorquers can be used to control the attitude and finally the effect of aerodynamic disturbance torques.

INTRODUCTION

The Space Technology Research Vehicle (STRV-1) is a small research satellite being designed and developed at the Royal Aerospace Establishment (RAE), Farnborough. The satellite will be used to carry out in-orbit investigations into the performance of a range of new space technologies which show a good potential for improving the lifetime, reliability and overall cost-effectiveness of future military and commercial space missions. The orbit chosen for STRV-1 is a geostationary transfer orbit (GTO); apogee 36000 km, perigee 200 km and inclination 8°, see Fig. 1. The GTO orbit has been selected for STRV because it encompasses a wide range of space environments - principally:

1. A high radiation dose for research into the performance of solar cells and microelectronics.
2. Plasma conditions at apogee which are suitable for electrostatic charging experiments.
3. Atomic oxygen is present at the low perigee altitudes which enables erosion rate experiments to be performed.

This highly elliptical orbit together with the type of launch chosen, i.e. as a piggyback passenger on an Ariane 4 mission, poses some new challenges for the attitude control of small spacecraft. The main characteristics of the STRV mission are given in Table 1, and Fig. 2 shows a sketch of the expected satellite shape and dimensions.

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Spin stabilization has been chosen to maintain the attitude of the satellite for four main reasons:-

i) it maintains its orientation in inertial space (desirable for power raising)

ii) it is effective at all altitudes

iii) it can assist with the thermal balance

iv) it has been widely used and there are many examples to assist with the design.

Table 2 shows the characteristics of the two types of actuator chosen for STRV-1. Magnetorquers have been chosen as the main actuators because of their simplicity - although they are only useful at low altitudes. Cold gas jets have been selected for the spin-up manoeuvre. The cold gas (xenon) on board the satellite is in fact intended for use as part of an experiment but a small quantity can be used for attitude control without disturbing the experiment.

Many similar satellite missions have been studied for this work including Explorer 45[1], Oscar 13[2] and AMPTE[3].

INITIAL CONDITIONS AND ACQUISITION

The launch envisaged for STRV-1 is as an auxillary passenger on an Ariane IV vehicle. The initial conditions for the satellite may be summarised as follows:-

i) a small component of angular momentum about an axis is the xy plane will exist due to mis-alignment between the separation force and the centre of mass.

ii) no spin is present about the z axis; neither the launcher nor the separation mechanism can provide this due to the particular method of launch

iii) no particular orientation of the initial angular momentum vector is guaranteed

The expected configuration of STRV-1 is shown in Fig. 2, the z-axis being both the axis of greatest moment of inertia and the intended spin axis. The reason for this is that any energy dissipation will cause the satellite motion to drift towards rotation about this axis, (e.g. Explorer 1, ATS 5). This process can be exploited by introducing energy dissipation devices (nutation dampers) and hence pure rotation about the z axis will be ensured. By this method it is possible to recover the satellite from its initial tumbling state.

Once the satellite is released from the launch vehicle, the gas jets can be operated to induce a 5 rpm rotation about the z-axis and the rotation about the axis in the xy plane will be reduced by the action of the nutation damper.

When the nutation has been reduced to zero it will be necessary to precess the z (spin) axis into the plane perpendicular to the sun line with an error of no more than $\pm 20^\circ$, see Fig. 3. This allows at least 94% of the maximum power to be generated whilst only a small solar flux impinges on the $\pm$ z faces. During the mission lifetime it will be necessary to precess the spin axis by about 1° per day to maintain its position in the desired plane as the Earth rotates around the sun, in addition to the manoeuvres necessary to correct any motions induced by disturbance torques.

Recovery from Initial Tumble

Fig. 4 shows the situation after the spin up manoeuvre has been carried out. The spin up angular momentum L_z is added to the initial angular momentum vector L_i, creating a total angular momentum vector L_T. Since this vector is not along a principal axis of moment of inertia the vehicle will nutate with a nutation angle, θ, given by

$$\tan\theta = \frac{L_i}{L_z} \tag{1}$$

The nutation damper needs to be carefully designed to provide rapid damping over a very wide range of nutation angles i.e. $0<\theta<60°$. In this cae a design goal is to achieve a residual nutation of 5° within one orbit period of 10 hours 30 minutes starting from 60°. There are a variety of different types of nutation damper, each of which has its own advantages and disadvantages[4]. Simplicity of the design is one of the most important criteria in selecting a damper. Serious contenders for selection were ball-in-tube dampers, tube-with-end pot dampers, and viscous fluid ring dampers. The partially filled axial fluid ring damper was chosen for the following reasons:

i) it does not have a natural frequency and is thus suitable for a variety of nutation rates

ii) it provides rapid nutation damping, compared to most other types

iii) it is mechanically very simple.

Note: if a fluid loop damper has an offset between its centre and the spin axis then it does have a natural frequency.

Precession of the Spin Axis

The magnetorquer coil design is of great importance and it must be optimized to achieve the required precession rate whilst minimizing the mass and power requirements. Precession is most efficiently achieved when the Earth's magnetic field strength is large. Fig. 5 shows how the field strength varies with eccentric anomaly around the orbit. The useful region is between $E = \pm 75°$ which relates to a 2 hour period with a mean flux (Bm) of approximately 10000 nT.

The required mean rate of precession is

$$\dot{\phi} = \frac{\Delta\phi}{Tp} \tag{2}$$

where Tp is the duration of the perigee pass in seconds, ϕ is the precession angle and $\Delta\phi$ is the total precession required in time Tp. The required coil magnetic moment is

$$M = \frac{\dot{\phi}\, L_z}{Bm} \tag{3}$$

To design the coil we must first derive the equations which relate coil mass and power to the magnetic moment.

Let

$c = 4\ell$ (circumference of coil side length ℓ)
σ = resistivity of wire
ρ = density of wire
a = cross sectional areas at wire
r = resistance per turn
R = coil resistance

The power consumed by the coil, P, is

$$
\begin{aligned}
P &= I^2 R \\
&= N I^2 r \\
&= \frac{N I^2 c \sigma}{a} \qquad (4) \\
&= \frac{4 N I^2 \ell \sigma}{a}
\end{aligned}
$$

but

$$I = M/NA$$

so

$$P = \frac{4M^2 \ell \sigma}{NA^2 a} \quad or \quad P = \frac{4M^2 \sigma}{N\ell^3 a} \qquad (5)$$

and the mass of the coil is given by

$$m_c = Nca\rho \quad or \quad m_c = 4N\ell a\rho \qquad (6)$$

As can be seen from equations (5) and (6) there are different constraints on N, ℓ and a, according to whether it is coil mass or coil power which is to be minimized. To minimize the power these parameters have to be large but to reduce the mass they have to be small. Clearly mass can be traded for power depending on the availability of each within the spacecraft budget.

IMPLEMENTATION OF CONTROL LAWS

The spacecraft attitude will be determined using Earth sensors and sun sensors. Once this has been done it will be necessary to implement control laws to perform two types of manoeuvre. These two types are: "Acquisition Manoeuvres" in the early part of the mission ($\Delta\phi$ up to $90°$ in the worst case), and "Corrective Manoeuvres" ($\Delta\phi$ a few degrees to correct disturbance torques).

If a UK ground station is selcted, these operations have to be performed out of ground contact since magnetorquers are used at perigee. There are three potential ways of issuing implementing the necessary control:-

i) hardwired closed loop control

ii) open loop control using time tagged commands

iii) on-board computer/software closed loop control

Of these three suggested, method iii) is the most flexible and the sophistication of the software need not be great. This method also allows for the attitude criteria to be altered during the mission lifetime should an experiment require it.

Other points to consider when selecting a control method include

i) visibility of STRV-1 from the UK ground station in the first 24 hrs is relatively poor; hence the satellite will probably be unattended during this critical stage

ii) sun sensors will be in-operative during eclipse which will most probably coincide with perigee (initially). A computer could record the sun angle before the perigee pass and predict the magnetorquer ON/OFF times.

It would be difficult to develop hardwired closed loop circuitry which has sufficient flexibility. The above arguments favour the use of computer based control for implementation of both the time-tagged and the softrware closed-loop control methods.

ENVIRONMENTAL DISTURBANCE TORQUES

On a spacecraft the size and shape of STRV-1, both solar radiation and gravity gradient torques can be presumed to be negligible. Detailed analysis from SSS-1[5] confirms this assumption. The main disturbance to attitude will come from aerodynamic torques caused by the offset between the centre of pressure vector and the centre of mass of the satellite when passing through the residual atmosphere at perigee.

Analysis of Aeroynamic Torques

Work carried out in this aerodynamic torque study was a continuation of work carried out by Van der Ha[6]. In his paper he developed the accommodation coefficients model suggested by Schaaf et al[7] to give the torque induced on a box-liike satellite. These formulae were coded in FORTRAN and the STRV-1 mission and configuration parameters were used to generate a graph showing how the magnitude of the induced aerodynamic torque varies throughout a typical perigee pass - see Figure 6.

The atmospheric parameters used for the analysis have a large effect upon the magnitude of the torques obtained, and since an exact launch date is unknown it was decided to assume a worst case. This consisted of a low perigee height (h_p = 190 km) and a high value of solar activity (10.7 cm solar flux, F = 260). Using these values with Jacchia[8] gives an atmospheric density at perigee of 4.95×10^{-10} kg m^{-3} and a density scale height of 34230 m.

The density relating to other points in the orbit was calculated using:

$$\rho(E) = \rho_p \exp\{-\beta(1-\cos E)\} \tag{7}$$

where E = eccentric anomaly, $\beta = H_p/ae$, ρ = air density, a = semi-major axis of the orbit and e = eccentricity of orbit

Suitable limits for the aerodynamic torque analysis can be found by considering only regions where the density is ≥ 1% of the value at perigee (ρ_p).

So if $\rho(0) = \rho_p = 4.95 \times 10^{-10}$ kg m^{-3} and $\rho(E) = 0.01\ \rho(0)$ kg m^{-3} then the limits on E are $0 \pm 8^{\circ}$ and using Kepler's equation it can be seen that this relates to a period of just 7.58 minutes (<1.2% of total orbit period).

The maximum torque which occured in the initial studies had a value of 6.7×10^{-4} Nm, and it was found that the spin axis orientation would be changed by about 7.2° - assuming a spin rate of 5 rpm and the worst angle of attack and centre of mass offset.

Analysis of possible luni-solar pertubations[9] has indicated that under normal launch conditions the perigee height could, at times, be 40km below the initial perigee height over a possible one year mission. An investigation into the effects of such changes was carried out and the results are shown in Fig. 7. At 140 km the worst case torque is about 0.005 Nm and the associated deviation would be about 40° per pergiee pass (at 5 rpm spin rate), thus rendering suitable attitude control quite difficult.

As a further safeguard for the attitdue control of the spacecraft it has been proposed to have a 2 stage spin up. The first spin would be carried out using the gas jets up to a speed of 5 rpm. Any nutation would then be damped out and the magnetorquers operated to attain an attitude in the plane perpendicular to the sun.

When this correct attitude has been achieved a second spin up manoeuvre can be carried out either by operating a spin plane coil or, if there is enough cold gas, by opening the gas jets again. This would increase the spin rate to 15 rpm. The advantages of this strategy over one where the spin rate is initially 15 rpm is that the spacecraft will exhibit less gyroscopic stiffness in the initital acquisition phase, where larger manoeuvres are more likely to be needed.

CONCLUSIONS

STRV-1 will be tumbling about an axis other than the desired z-axis immediately after separation from the launch vehicle. The task during the acquisition phase of the mission is to recover from this state and re-orientate the spacecraft z (spin) axis to lie in a plane perpendicular to the sun line. The first stage is to spin up the satellite about the z-axis and rapidly damp the resulting nutation. Of the various types of nutation damper available, the axial fluid loop damper appears to be the most suitable since it gives rapid energy dissipation, does not have a natural frequency and is mechanically very simple.

The precession of the spin axis to its desired position appears to be achievable using an axial spin-axis magnetorquer which is activated at perigee. The mass and power consumed by a magnetorquer of a strength sufficient to produce a 90° precession per perigee pass appears to be compatible with the spacecraft budgets. Mass and power can be directly traded-off between each other depending on which is to be conserved.

The use of a computer based system to control STRV-1 is favoured because

i) it gives flexibility to the mission in general

ii) ground contact is not possible during the first 2 orbits of the acquisition phase

iii) problems with lack of sun sensor data during eclipse can be overcome

iv) both time-tagged and software closed loop control can be used.

Analysis has been carried out on the effect of aerodynamic torques on the satellite and it has been suggested that two different spin rates be used in the mission. The first of about 5 rpm, will be used while nutation is damped out and for the large acqusition manoeuvres to be carried out. The second of 15 rpm will be used to give the spacecraft more gyroscopic rigidity so improving its resistance to aerodynamic torques at perigee.

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Table 1

STRV-1 SPACECRAFT SUMMARY

Purpose	Technology Research
Mass	50 kg
Orbit	36000 km × 200 km (GTO), i = 7°
Stabilisation	Spin (– 15 rpm)
Attitude control	Magnetorquers/gas jets
Solar arrays	Body-mounted 35W BOL
Structure	Composites: Carbon PEEK
Computer	MIL-STD 1750 Silicon-on-sapphire
Communications	1 kb/s, 2.2 GHz
Launch	Ariane ASAP
Ground Station	RAE Lashm, UK

Table 2

ATTITUDE CONTROL ACTUATORS

1 Magnetorquers	Simple
	Easily controlled
	Mass/power can be traded off
	Consumes power
	Torque about a field line is not possible
	Only efficient at low altitudes
2 Cold Gas Jets	Usable at all altitudes
	Propellant required
	Complex 'plumbing'
	Careful control required

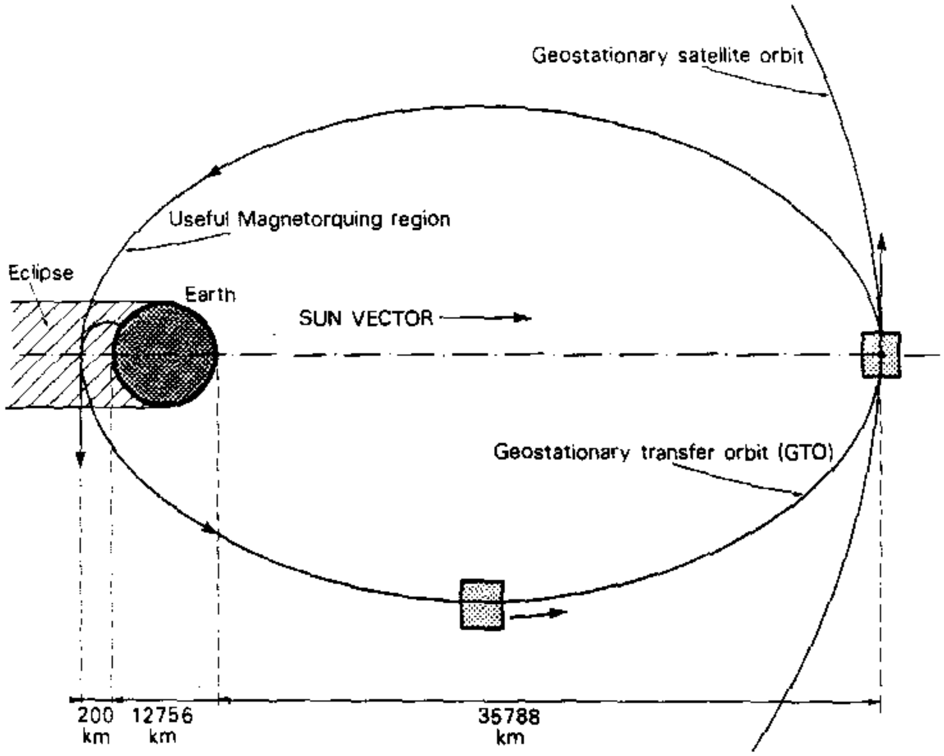

Fig. 1 The STRV orbit - Ariane 4 geostationary transfer orbit.

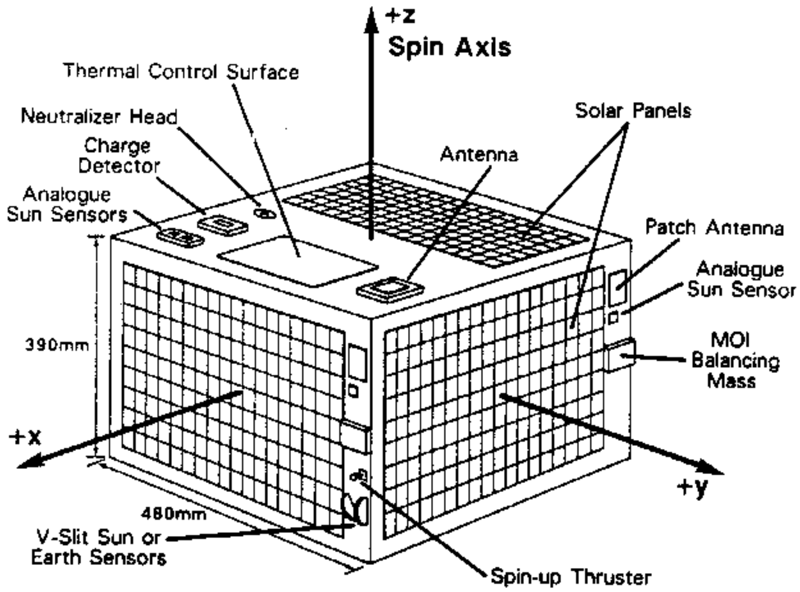

Fig. 2 STRV 1 general configuration.

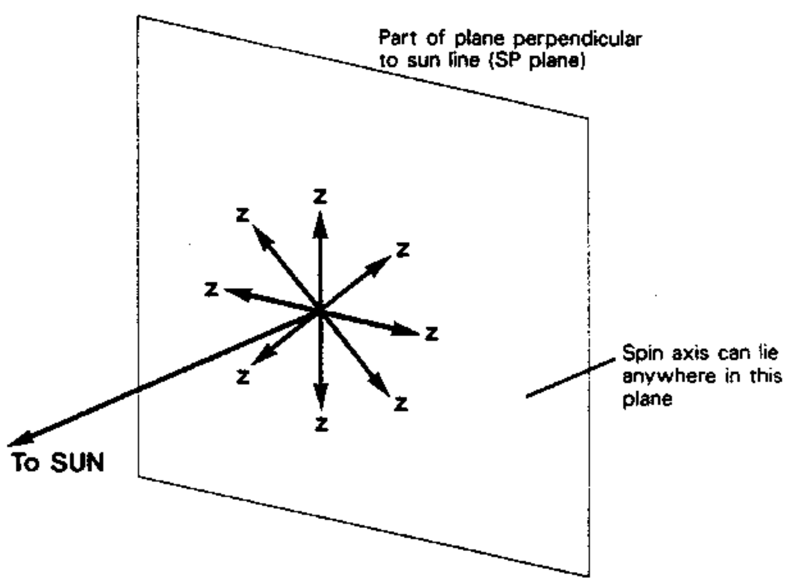

Fig. 3 The 'sun-perpendicular' plane.

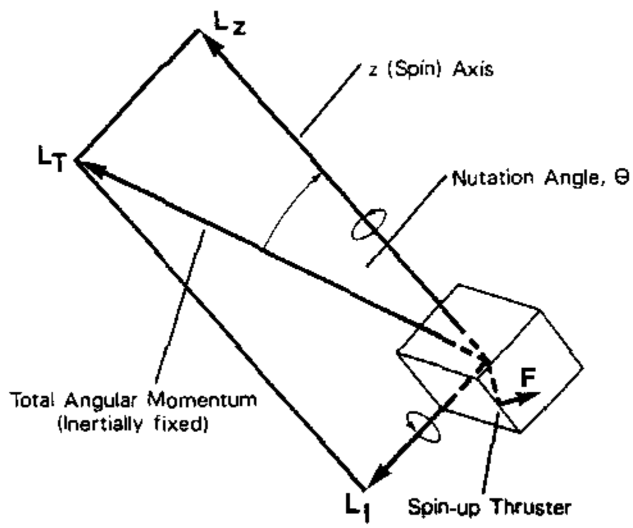

Fig. 4 Nutation angular momentum diagram.

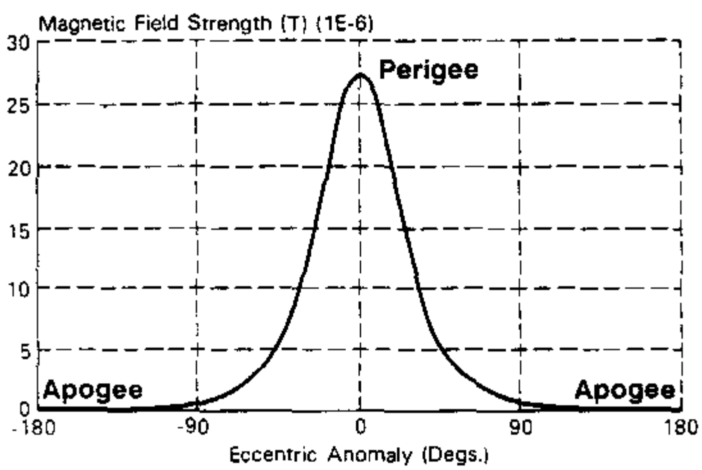

Fig. 5 Variation in magnetic field magnitude in GTO.

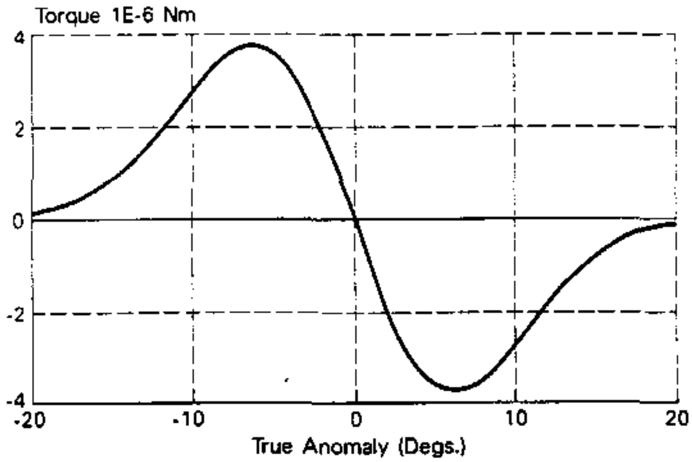

Fig.6 Typical variation of aerodynamic torques through a perigee pass.

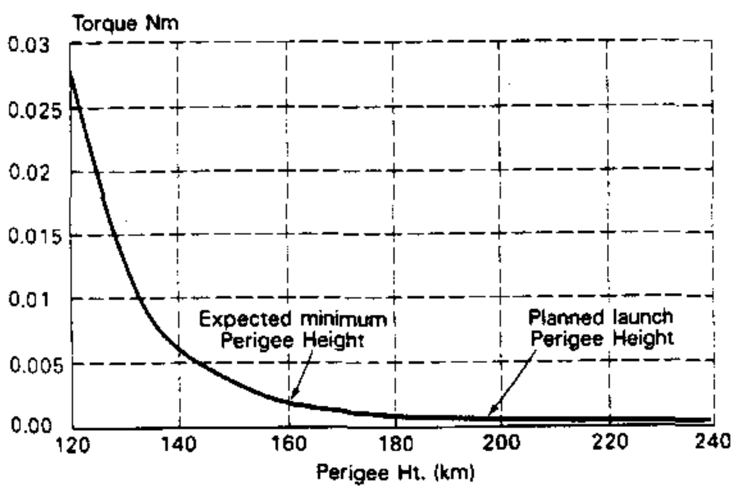

Fig. 7 Effect of perigee height changes on the maximum predicted torque values.