

UOGAS - A GET AWAY SPECIAL SATELLITE WITH ORBIT-RAISING
CAPABILITY

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ABSTRACT

The low cost of satellite deployment from Shuttle GAS canister makes it an attractive launch option. However, the low deployment altitude severely constrains lifetime so the UoGAS (University of Surrey Get Away Special) spacecraft will incorporate a propulsion system. Lifetime extension methods are considered and a start-of-mission orbit-raising manoeuvre is selected. An orbit dynamics simulation method (taking into account the atmospheric drag) is discussed and results presented. Mission profiles, propulsion systems (including colloid, resistojet and cold gas thrusters, as well as a novel hybrid bipropellant combination) and stabilisation options are discussed: an aerodynamically stabilised vehicle with a resistojet propulsion system is suggested and some design aspects of the satellite systems are discussed.

INTRODUCTION

Although satellite launch from an Get Away Special canister on board the US Space Shuttle is a very cheap way of getting to orbit, vehicles launched in this manner suffer from comparatively short lifetimes owing to orbit decay caused by atmospheric drag. For many missions it is desirable to have a longer orbital lifetime, and research at the University of Surrey has been initiated into achieving this.

Two methods of lifetime extension exist - continuous operation of a propulsive device to compensate for the drag, and raising the orbit at the start of mission to an altitude where atmospheric density (and hence drag) is less severe. Both methods require a propulsion system which must not only deliver the required impulse, but must also make minimal impact on the mass, volume and power budgets of the vehicle. In addition, compliance with the stringent STS safety rules is necessary.

This report describes the preliminary results of the UoGAS (University of Surrey Get Away Special) study, with particular attention to the propulsion system and the demands it places on the attitude control of the spacecraft.

THE PROBLEM

Before considering the possible devices for the propulsion system, it is necessary to define the requirements it must fulfil.

An object in LEO undergoes a slight decelerative force due to the drag of the rarified upper atmosphere. This drag dissipates the satellite's orbital energy with the result that the orbit contracts. This contraction brings the satellite to lower altitudes where atmospheric density is greater so orbital decay accelerates.

Predictions of orbital lifetime are notoriously difficult to make (c.f. Skylab) as atmospheric density varies not only with altitude, but also with solar activity. The two small satellites that have so far been deployed from STS-GAS canisters, Nusat and GLOMR, were inserted into approximately 350 km orbits. As a result of unexpectedly low solar activity, the satellites had lifetimes of about 600 days : a more realistic lifetime for a 65kg satellite sized to fit a standard GAScan is one year or less, too short for most applications. For a three year plus lifetime an initial altitude of 500km is required (see fig.1).

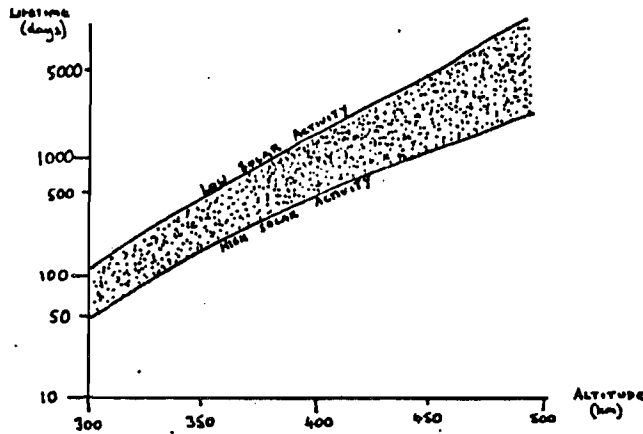


Fig.1 GAS Satellite Lifetime

Operational altitude of shuttle missions varies from flight to flight, 350km being somewhat higher than average to date. A nominal deployment altitude of 300km is therefore assumed in this paper. The ideal (Hohmann) delta-v required to transfer from 300km to 500km is 114m/s, although in practice the manoeuvre will be non-impulsive due to low engine thrust and there will be a drag penalty.

The alternative to raising the orbit for lifetime extension, is to remain at low altitude and fire a thruster intermittently or continuously to maintain the orbit. In continuous drag compensation thrusting, the thrust is set equal to the drag force so that the satellite remains in the same orbit. In semi-continuous drag compensation, the thruster is fired at intervals to keep the satellite within a specified altitude band, firing when the lower bound is reached until the orbit is raised to the upper bound, when the motor is turned off and the satellite slowly descends again to repeat the process.

Because continuous drag compensation keeps the satellite at a low altitude where drag is higher, one expects intuitively that using a given quantity of propellant initially to raise the orbit will give a longer lifetime than using that propellant to cancel the drag. This has been borne out by computer simulations, so attention will focus on orbit-raising.

ORBIT DYNAMICS

As indicated above, the dynamics of the UoGAS craft are non-ideal. It is likely that a low-thrust propulsion system will be used, and for a portion of the transfer at least, the spacecraft will suffer an atmospheric drag force equal in magnitude to a significant fraction of the engine thrust, a condition the author refers to as 'in drag'. In order to accurately compute the propellant requirements, therefore, an iterative approach has been adopted.

For a satellite in orbit about the earth at altitude h km, the orbital velocity is given by

$$v = \sqrt{u/a} \quad (1)$$

Where u is the gravitational constant of the earth, $3.99E-14$ and a is the semi-major axis of the orbit given by

$$a = h+r_0 \quad (2)$$

r_0 being the earth's radius, 6370km

The drag force experienced by the satellite is given by

$$D = \frac{1}{2} S C_d \rho v^2 \quad (3)$$

S being the area presented by the spacecraft normal to the relative flow of the atmosphere (the atmosphere is assumed to be unmoving, so this is the direction of the spacecraft velocity)

C_d is the coefficient of drag, taken as 2.2 (a typical spacecraft value)

ρ is the atmospheric density, assumed to be constant throughout the orbit

The orbit period is given by

$$P = 2\pi(a/v) \quad (4)$$

so the velocity increment (Δv) per orbit is

$$dV = P (CT-D)/M \quad (5)$$

where T is the engine thrust and C the engine duty cycle factor (we assume for the moment continuous thrust so $C=1$)

M is the spacecraft mass (it is assumed that the mass does not significantly alter during one orbit)

King-Hele (ref. 1) developed the following formula for low-thrust coplanar orbit transfer, valid for situations where the orbit altitude does not change significantly

$$dV = V_c(1-n^{-0.5}) \quad (6)$$

rearranging, we have

$$n = (1-dV/V_c)^{-2} \quad (7)$$

where n is the ratio of semi major axes of the orbit after and before transfer, dV is the velocity increment and V_c is the orbital velocity in the initial circular orbit (= V in eqn.1 above.)

So the new height h after one orbit of thrusting is

$$h = na - r_0 \quad (8)$$

during which time the spacecraft mass changes by

$$dm = C T P / (Isp g) \quad (9)$$

where Isp the the delivered specific impulse of the propulsion system (in seconds) and g is terrestrial acceleration due to gravity, 9.81 m/s

so the new spacecraft mass is

$$m = m - dm \quad (10)$$

Successive application of these formulae is made by computer after the initial conditions are set, until target orbit is reached or propellant is exhausted.

Extensive computer simulations were made using the above method to determine the effects of varying atmospheric density, engine thrust, deployment and target altitudes and specific impulse.

The atmospheric density, as mentioned before, is a strong function of both solar activity and altitude. For the purpose of these calculations, data was taken at 50km intervals from the CIRA 1972 reference atmosphere (see, for example, ref.2) and an empirical (although not necessarily best) fit was made.

The formulae used to approximate the density at an altitude H km (for the altitude range 200 to 500 km) are

$$F = (H-199)/300 \quad (11)$$

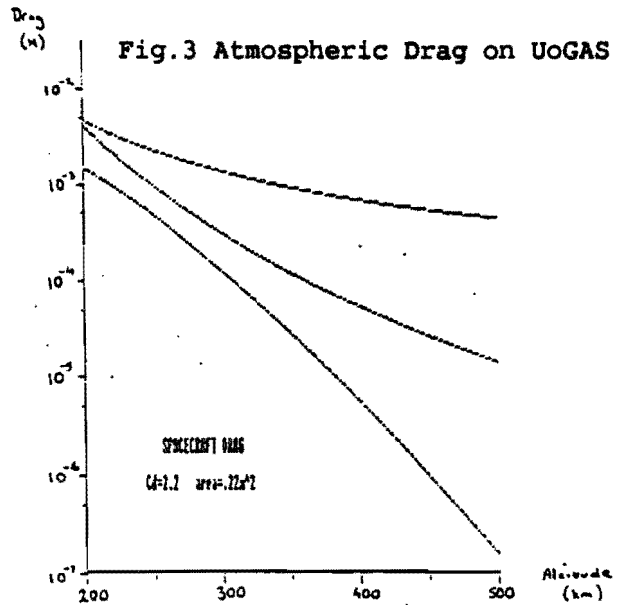
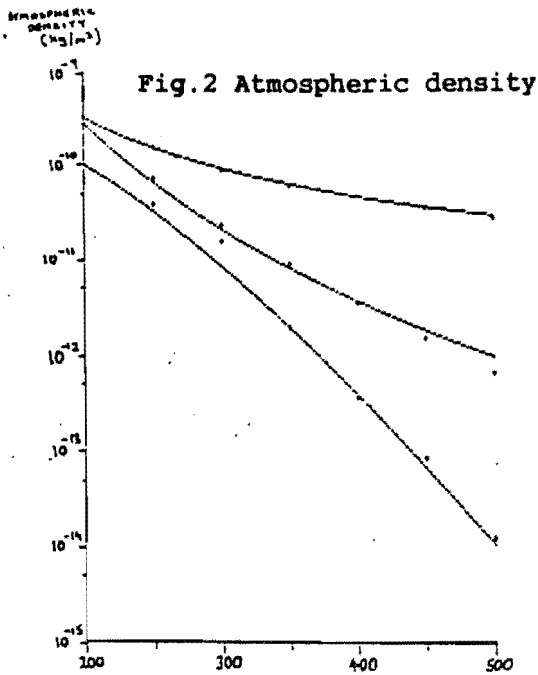
$$SH = A + Bx F^{0.5} + Cx F \quad (12)$$

$$\rho = D \times \text{EXP}((200-H)/SH) \quad (13)$$

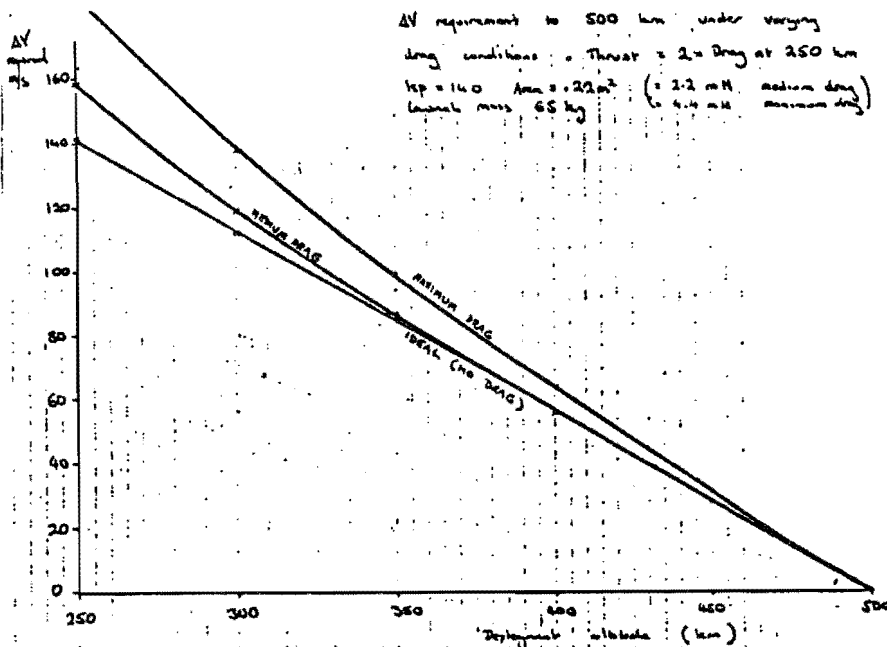
Where for low solar activity $A=55$ $B=-27$ $C=5$ $D=1.0E-10$
for mean solar activity $A=26.5$ $B=14$ $C=13$ $D=2.8E-10$
and for high solar activity $A=50$ $B=20$ $C=60$ $D=3.2E-10$

High solar activity is expected for the next few years.

Figure 2 shows the fit of the curves defined by the above equations to points taken from (ref.2) and figure 3 shows the corresponding drag forces undergone by the spacecraft.



The simulations indicate that to be assured of reaching the target orbit for a wide range of starting conditions, a nominal delta-v of 140 metres per second is required (see figures 4 and 5.) Including a contingency to allow for a pointing error and other losses (see later) a figure of 160 m/s will be taken as a requirement for the propulsion system. An orbit-averaged thrust (CT) of 2.2mN will be sufficient for all except the worst deployment conditions i.e. below 300km and solar maximum (the CIRA maximum density is perhaps a little pessimistic in any case.)



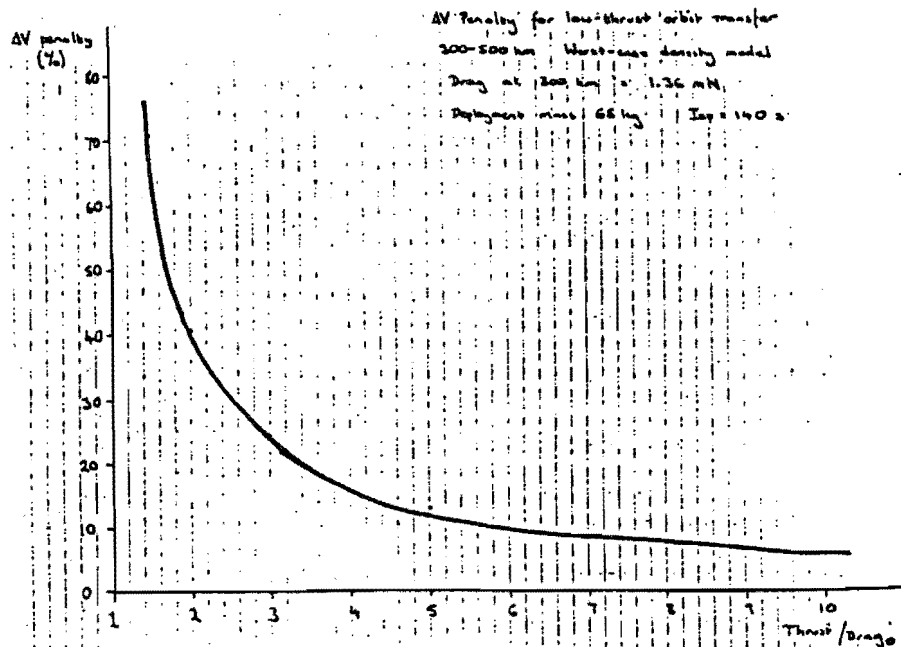


Fig.5 Delta-V penalty for low thrust/drag

MISSION DESIGN

Having ascertained the delta-v requirements, we must now consider how to generate this impulse. Since an impulse must be directed along the spacecraft's velocity vector in order to raise the orbit, the attitude control system and the propulsion system are strongly coupled. Possible orbit-raising schemes are described below.

Spin Stabilisation (1)

The satellite is spun up after deployment with the spin axis lying in the orbit plane. Two motors (one at either end) are carried and are fired at opposite parts of the orbit when the spin (and motor) axis lies along or near to the spacecraft velocity vector. For high-thrust systems this allows a simple Hohmann (ideal, impulsive) orbit transfer to be made, and the spin stabilisation is useful for evening out any thrust misalignments. For low-thrust systems, many small Hohmann transfers will be made until the target orbit is reached.

Spin Stabilisation (2)

The satellite is again spun up with the spin axis in the orbit plane. Only one motor is carried and is fired when the spin axis is aligned with the velocity vector. The new orbit after the burn will be slightly eccentric and the line of the apsides will rotate in the orbit plane as a result of the Earth's oblateness. When the spin axis is aligned with the velocity vector at apogee (after about 14 days in an equatorial orbit) another burn is made to raise the perigee. The process is repeated as necessary. This method has the advantage that only one

motor is required, but the very low duty cycle means that if thrust is low, a prohibitively large drag penalty will be incurred.

Spin Stabilisation (3)

This time the spin axis is parallel to the orbit normal and a thruster is carried which fires normal to the spin axis. The thruster is fired once per revolution when the thrust vector will point along the orbit.

Magnetic Stabilisation

In polar orbits it may be possible to make the spacecraft track along the lines of force of the geomagnetic field by using a set of magnetorquing coils and appropriate algorithms. (Note that it is not possible to effectively track the field using only a permanent magnet - an experiment was carried out on UoSAT-2 to verify just this: it is necessary to commutate the magnetorquers to damp out oscillations.) For at least two instants in the orbit (and possibly more) the spacecraft will be aligned with its velocity vector: only one motor should be required.

Aerodynamic Stabilisation

By locating the spacecraft centre of pressure behind its centre of mass a 'weathercock' stability is produced, tending to point the satellite 'head first' along the orbit. The spacecraft is therefore always aligned with its velocity vector, enabling continuous operation of a low-thrust propulsion system. The last option is that currently favoured for UoGAS. The volume allowed by the GAS canister means that the correct moment of inertia combination for the spin axis to be the axis of symmetry cannot be obtained easily without significantly affecting the spacecraft design (e.g. by the incorporation of booms with tip-masses to increase the transverse moment of inertia.) As indicated later, the propulsion system selected is likely to be a low-thrust one so thruster duty cycle should be as high as possible: the aerodynamic stabilisation option is the only one which permits continuous thrusting for all inclinations.

PROPULSION SYSTEM

The propulsion system must not demand too great a fraction of the available mass volume and power, plus it must not violate the stringent (and occasionally indeterminate) safety rules imposed by a man-rated launch system such as Shuttle.

Conventional monopropellant (e.g. hydrazine, hydrogen peroxide) and bipropellant (hydrazine/dinitrogen tetroxide) systems are therefore rejected on the grounds of safety. Other liquid propellants (LOX/hydrocarbons) are rejected on storability and complexity grounds.

A possibly interesting alternative is a gaseous H_2/O_2 motor, using propellants generated electrolytically from liquid water stored on-

board. Such a system would be attractive on the grounds of safety and performance, but development attempts to date have proven unsuccessful.

Solid rocket motors are reasonably safe, but are even so rejected for safety reasons. Ion thrusters cannot be used as, to generate the required level of thrust, they would consume far too much electrical power.

The options left then are cold gas propulsion systems and low-power electrical thrusters.

Cold gas systems are very simple and therefore reliable. However the maximum specific impulse that can be obtained is low, and the best propellants, propane and ammonia (ref.3), may be ruled out by the safety regulations and by the volume constraint.

Resistojets are a step up from cold gas systems in that the specific impulse is increased by electrically heating the gas. Specific impulse is determined by the amount of heat injected per unit mass of gas, and this flexibility is useful.

The colloid thruster (ref.4) is a device that received much attention during the early 1970's. Essentially a conducting fluid is passed to an emitter (a needle or a slit) where in the presence of an intense electrostatic field its meniscus cannot be sustained and tiny multimolecular droplets are evolved which are then accelerated by the electric field. These devices offer the highest power-to-weight ratio of any high performance electric thruster and still give an excellent specific impulse (1000-1300 seconds.) The propellant used by these devices is glycerol (doped with sodium iodide to make it conducting) - about as innocuous as you can get. The particles emitted are positively charged, so beam neutralisation is accomplished simply by a hot-wire cathode.

Plasma thrusters of a size and power small enough for use on a small satellite are being developed for use on the West German AMSAT Phase 3D spacecraft. The propellant to be used is water, with available specific impulse adjustable in the range 100-300 seconds. The thruster is being developed at the University of Stuttgart, but the author has no further data at present.

A new hybrid bipropellant combination was devised by the author in 1987 (ref.5) which, it was felt, might have a chance of satisfying the safety rules. Like other bipropellants, the two components are kept safely apart until firing, but unlike conventional bipropellants, neither component is particularly inflammable, corrosive or toxic. The combination is based on a chemists' 'party trick' - glycerol dripped onto potassium permanganate will spontaneously (after a few seconds) burn vigorously. Initial experiments by the author suggested that the combination would offer reasonable performance. More recently thermodynamic calculations based on the methods of (ref.6) indicate that a specific impulse of about 100 seconds is likely, although the mass utilisation (ejected mass divided by propellant mass) is low -

about 0.45. It was felt that this level of performance did not merit the effort of solving the complicated mechanical design of the propulsion system and incurring the development risk. However, the combination (and higher-performance, but non-hypergolic variations - e.g. heavy liquid hydrocarbons and perchlorate oxidisers) might find useful application elsewhere.

The colloid thruster would clearly be an attractive choice, but there are several obstacles to its implementation. First is that some of the emitted beam might impinge on the spacecraft structure (with consequent sputtering and back-emission possibly resulting in degradation of the thruster.) The emission sites (e.g. the needle tips) need to be made from platinum-iridium to minimise erosion, so fabrication of most designs is difficult and expensive. Although colloid thrusters were ready for flight test in the 1970's the work has since been abandoned, so it may be difficult to reactivate the technology.

Resistojet

We are therefore left with the resistojets, and this device will be described in more detail here. In general, the propellant is (in order to occupy a reasonably small volume) a vapourising liquid. Vapour from the liquid enters a chamber called the plenum where it is heated electrically by several hundred degrees centigrade. The hot gas is then expanded through a nozzle to generate thrust.

The fluid drawn from the propellant tank may be liquid or vapour, so before entering the thruster it should be dried. One way of achieving this is to pass the fluid down a long pipe coiled around the tank: this is the method selected by Fleeter (ref.6) who designed the propulsion system for PACSAT, a similar satellite to UoGAS. PACSAT was intended to carry a packet communications payload from shuttle orbit to some 800km, using a freon-fuelled resistojets and magnetic stabilisation (although the details of the magnetic stabilisation were not described.)

An alternative is to have an intermediate chamber, the preplenum, to which fluid is only admitted when the preplenum pressure is below a certain value. This reduces the pressure excursions undergone in the plenum as well as ensuring gaseous propellant feed (ref.7).

The electrical heating of the propellant causes the pressure drop (due to friction) from inlet to nozzle throat to increase with the result that the plenum pressure is lower and hence the mass flow rate drops. The exhaust velocity of the propellant (and hence specific impulse) increases, but the overall thrust drops as the input power increases. For example an ammonia resistojets system described in (ref.8) has a nominal thrust of 2.2mN at 140s for an input power of 8W. If the power is cut off, the specific impulse falls to about 95 seconds, but the thrust increases to about 2.9mN.

The temperature of the gas is limited to about 2000K by the materials that can be used. At such high temperatures, many of the propellants will tend to decompose, hence lowering the mean molecular mass of the

exhaust products and raising the specific impulse. However, the energy absorbed in dissociating the propellant molecules increases the power requirement significantly (fig.5 in ref.9), so dissociation should be discouraged unless very high temperatures are used.

The main cause of inefficiency in the resistojet is heat loss, mainly due to radiation, so to minimise this loss resistojet thrusters are small. This reduction in scale tends to introduce some viscous losses owing to the low Reynolds number of the flow and the nozzle throat is more easily blocked by debris, but the radiative loss is the more significant one.

The nozzle itself is frequently conical for ease of fabrication, as for nozzles of this size the advantage (if any) to be obtained from a more paraboloid nozzle is minimal. Expansion ratio will depend on the propellant used and its temperature. For maximum performance the propellant should be cooled during its expansion to the point where it will condense just beyond the exit plane of the nozzle (ref.3). The thruster in (ref.6) using Freon 114 heated to 700K had a nozzle with a 15 degree half-angle and an area ratio of 100 with a throat diameter of 0.635 mm.

The gas in the plenum will be typically at 1000K or more (the limit depends on the performance needed and the stability of the propellant: for example many freons decompose above 1000K.) Hence it is common for the heater and nozzle to be fabricated from metals such as rhenium or tungsten, although these metals are not without their problems. For devices operating at moderate temperatures (say below 1000K) high quality stainless steels such as Inconel can be used. The insulation used to limit losses to the spacecraft body and to deep space depends on the exact configuration, but metal foil is normally used. One study (ref.10) identified fibrous block materials as being adequate for most applications, with refractory paper offering better performance if suitably constructed. Molybdenum foil offered the best insulation, but it was noted that the performance was sensitive to design details so the configuration for maximum performance for a given thruster requires experimental optimisation.

The choice of propellant is, for this application, one of the trickier design areas. This particular application is more volume-constrained than mass-constrained, so the density of the stored propellant should be as high as possible. Shuttle safety rules require that pressure vessels be able to withstand four times the vapour pressure of their contents at 105 degrees centigrade, so the propellant should have as low a vapour pressure as possible. In order to maximise specific impulse the fluid used should have a low molecular weight. A low latent heat of vapourisation at the storage temperature is desirable, as is a low freezing point to minimise danger of propellant freezing in the tank or feed lines. The fluid used should be as unreactive as possible to minimise materials compatibility problems. Finally, the propellant should be non-toxic and non-inflammable to minimise the risks to ground personnel and to the launch vehicle and its crew.

STS safety rules are vague on what can and cannot be allowed - before the Challenger disaster it was expected that GAS satellites might be able to use ammonia, an excellent resistojet propellant, but now it is not certain whether GAS satellites with any kind of propulsion system will be allowed to fly. A new programme, CSCP (Complex Self-Contained Payloads) allows GAS-type payloads with hazardous materials (for example on missions like DSI's Chemical Release Observation satellites, which will carry hydrazine) to be flown, although this entails far more complicated (and hence expensive) safety qualification procedures.

Although they suffer from fairly high storage pressures and comparatively low densities, propane and ammonia are good propellants, but may be restricted by safety rules. Various non-toxic and non-flammable refrigerants (Freons 114 and 21 especially) therefore become attractive, especially on the grounds of high density and storage pressure. However, they cannot be used at very high temperatures as decomposition products (e.g. carbon) might clog the nozzle throat. This fact, plus their high molecular weight gives them fairly low performance (e.g. Freon 114 at 700K gives a specific impulse of about 90 seconds). Another difficulty has arisen more recently as a result of environmental concerns: chlorofluorocarbons of this type destroy the ozone layer. Their use is accordingly being limited, so availability may become a problem. New Freons (e.g. F-134a) without the damaging chlorine in their compositions, are being investigated.

Water is another candidate propellant, and has a low molecular mass and a reasonably high density. It is (as is well-known) non-toxic and non-flammable and presents few materials compatibility problems. Its low vapour pressure means that the storage vessel need not be strong and massive, but makes propellant feed difficult. This would probably necessitate the incorporation of a bladder and a pressurant in the propellant tank, which would make the propulsion system more complicated. Potential freezing is also a problem. The main objection to water is its very high latent heat of vapourisation, which imposes a large power requirement on the propulsion system. Water is therefore an unattractive propellant on performance criteria, except when very high temperatures/input powers are used.

STABILISATION SYSTEM

Aerodynamic stabilisation is rarely used on spacecraft, for obvious reasons. It was used on Cosmos 149 and 320, two experimental meteorological spacecraft in 267x327km orbits, achieving pointing accuracies better than 5 degrees (ref.10) and may be used on the forthcoming TSS-2 (Tethered Satellite System) mission, at about 120km altitude. Thus although the method has been used in practice, it has not been proposed for use up to the 500km height intended for UoGAS.

In principle aerodynamic stabilisation is straightforward - a body with its centre of mass significantly ahead of its centre of pressure will tend to point 'into wind'. Although the atmospheric density at altitude is very low, as discussed earlier, the large orbital velocity gives a

significant dynamic pressure which can cause appreciable forces and torques. One of the important and valuable features of this form of aerodynamic stabilisation is that the satellite is held in an attitude presenting the minimum area to the the incident flow - i.e. drag is minimised.

The spacecraft configuration must be designed so that the aerodynamic torques are greater than the other (disturbing) torques acting on the vehicle. Magnetic torques should not be significant, unless magnetorquers are energised, and solar radiation pressure is an order of magnitude lower than the aerodynamic pressure up to 500km, except during low solar activity, when the ceiling effective stabilisation descends to about 400km. The unwanted torques generated by the propulsion system are estimated to be small, as the thrust vector should always lie within a couple of degrees of nominal. Gravity gradient torque is, as might be expected, the main disturbance torque. This torque is minimised by making the moments of inertia of the satellite as similar as possible. The volume available in the GAS canister almost makes satisfaction of this requirement inevitable.

In order to have a centre of pressure behind the centre of mass, the configuration in figure 6 is proposed. The basic shape is an octagonal cylinder, sized to fit inside the standard GAS canister (fig. 7), an octagon being a reasonable compromise between ease of fabrication (favouring perhaps a cuboid) and volume utilisation (favouring a circular cylinder). On the eight long sides of the vehicle are slightly smaller flaps, hinged at the bottom which fold out through 180 degrees to form a skirt - doubling the length of the vehicle after deployment, but keeping the centre of mass in largely the same position.

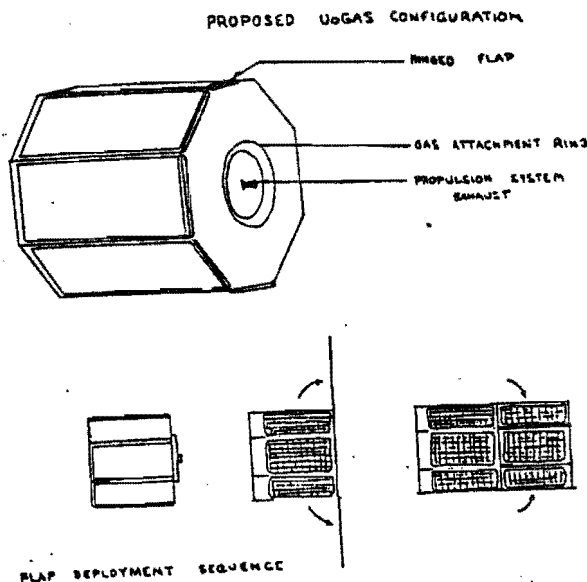


Fig.6 Proposed UoGAS configuration

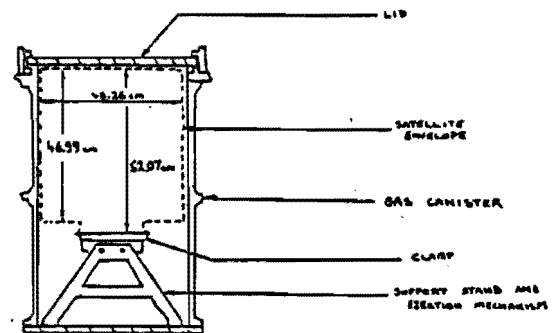


Fig.7 GAS Satellite Envelope

The aerodynamic torque is found by considering the atmosphere as of uniform density throughout the orbit, and at rest with respect to the centre of the earth (i.e. no upper atmospheric winds and no atmospheric rotation.) As the mean free path of the species at this height is large compared with the dimensions of the spacecraft, free-molecular flow is assumed (i.e. molecules do not interact with one another) and the flow is also assumed to be hyperthermal (at this altitude the most probable thermal speed of the molecules is about one seventh of the orbital velocity of the spacecraft.)

For accurate modelling of aerodynamic forces and torques it is necessary to model the re-emission of molecules from the spacecraft surface (refs.11,12). The re-emitted momentum flux depends on a large number of parameters, including angle of attack, speed ratio, surface composition and temperature etc. As yet, no completely satisfactory model has been formulated. In this study therefore, a simplified approach has been adopted, where the force on a surface element dS is

$$dF = \frac{1}{2} \rho dS v^2 C_d$$

with symbols having their usual meanings and dF acting in the direction of the incident momentum flux. C_d is again taken as 2.2. The surface elements considered are simply the flat sides of the satellite. This expression calculates the drag force acting on the sides of the satellite, from which moments can be calculated - for the configuration above, the resultant moments are shown in figure 8. The re-emission effects tend to produce a lift force (typically less than half of the drag force) which would further increase these moments (and hence the stability of the vehicle.) So while this method is far from accurate, it is useful in that it gives a lower bound to the torques produced. In any case, the great uncertainty in estimating atmospheric density largely negates any advantage in improving gas-surface interaction models.

At low altitudes the stable attitude will be along the orbit, collinear with the velocity vector. However, as noted in (ref.13), the gravity gradient torque falls off as the inverse cube of distance, whereas atmospheric density falls off exponentially. Thus at high altitude the magnitude of the two torques becomes similar and the stable attitude becomes one with a pitch angle (θ_{eqm}) between 0 and 90 degrees, the exact angle being determined by the relative magnitudes of the two torques. A limit of 26 degrees has been set on this angle, corresponding to a 10 per cent loss in effective thrust ($\cos 26^\circ = 0.9$.) The stability of an aerodynamically stabilised vehicle of this configuration is shown in figure 9. (The gravity gradient torques were

calculated for uniform mass distribution, with the main body of the satellite being of mass 60kg and the 'skirt' 5kg.)

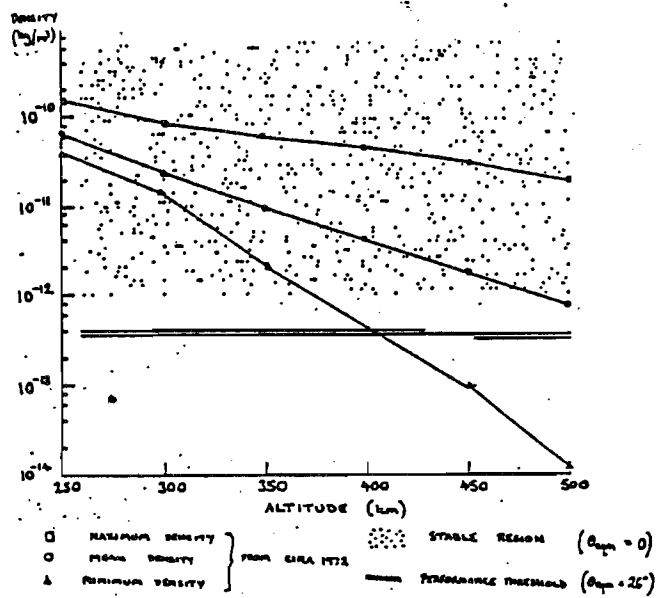
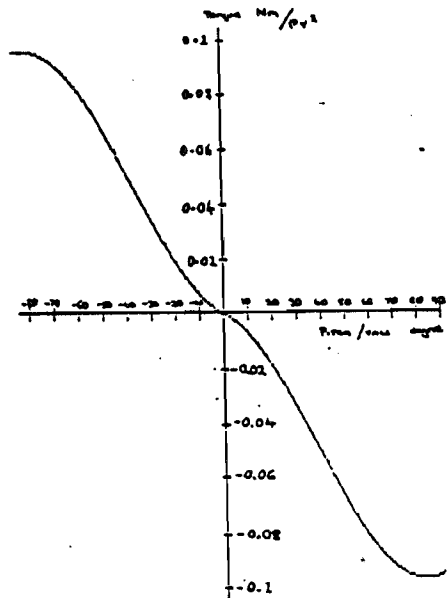


Fig. 8 Aerodynamic Torque Characteristic Fig. 9 Attitude Control Performance Map

It is seen that adequate performance is obtained to 500km in all cases, except during low solar activity when a ceiling of 400km is imposed.

However, the existence of a position of stability is no guarantee of effective attitude control. First, the required attitude and attitude motion must be acquired (GAS payloads are not in general allowed to impose demands on the shuttle orbiter's orientation) and in addition, anomalies may cause the craft to be disturbed from this stable position. The aerodynamic torque is non-dissipative, so any initial position or motion error will not be damped down. Hence large amplitude librations (much like on gravity gradient stabilised satellites) can build up and would persist. For attitude acquisition and delibration, therefore, a secondary attitude control system is required, and following successful experience on UoSATs 1 and 2, magnetorquers are suggested. These would be wound around the edges of the spacecraft structure, or around the edges of the solar panels. The attitude control algorithms developed and under development (ref.14) at UoSAT involve the commutation of the 3-axis magnetorquer coils under on-board control using data from a high-resolution (12/14 bit) 3-axis fluxgate magnetometer.

Simulations of the attitude motion of the satellite (using the moment coefficients in fig.8 above and the medium atmospheric density profile) are shown in figures 10,11,12 below. The numbers at the top of the graphs indicate altitude, initial pitch angle, initial pitch rate, magnetometer activation threshold (torquer fires if scaled rate plus angle exceeds threshold) and torque exerted by magnetorquer. As can be readily seen, an initial displacement results in undamped libration. Activation of the torquer (actual torquer firings are shown at the

bottom of the figure) reduces the amplitude of the libration. If smaller thresholds and torques are used, the residual libration will be smaller, although +/- 5 or 10 degrees should be quite acceptable.

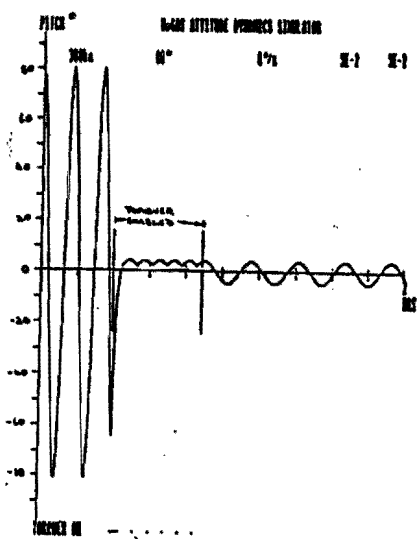


Fig. 10 300km

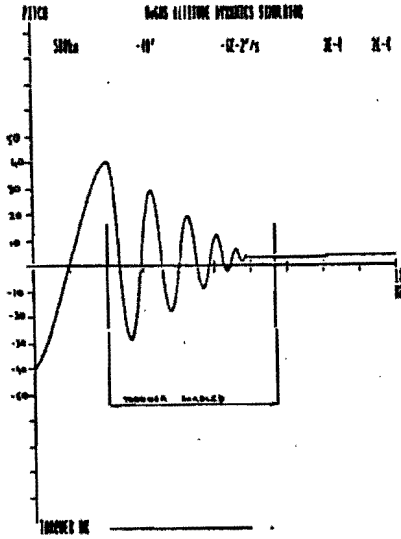


Fig. 11 500km

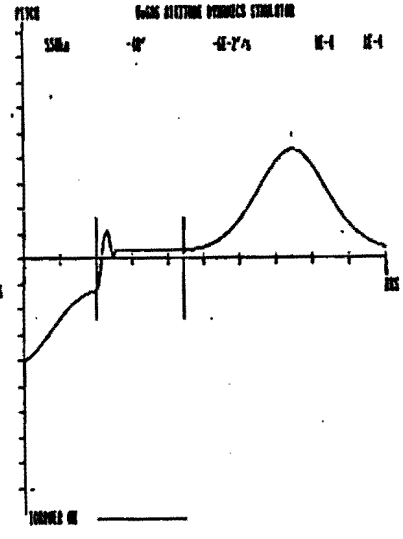


Fig. 12 550km

Attitude Dynamics Simulation

Note that in figure 12, with an extreme altitude of 550km, librations occur, as predicted, about a stable position corresponding to a pitch offset of about 25 degrees. At higher altitudes still, the equilibrium pitch angle tends rapidly to 90 degrees (gravity gradient stabilisation.) Note also that since the restoring aerodynamic torque falls off with altitude, the libration period increases as the satellite climbs. The yaw motion of the satellite is virtually identical, except that the stable position has no offset at high altitude.

The motion about the roll (motor) axis of the vehicle has not been investigated as yet. It is assumed for the moment that the spacecraft will not need to be stabilised about this axis, but if a particular pointing is desired, it should be possible to achieve this, at least for a short period, with the magnetorquers.

Note that a magnetorquer cannot generate a torque about the local line of force of the geomagnetic field. However, the field is not a perfect dipole, so it should be possible to generate at least a small torque about any of the three principal spacecraft axes at virtually any time.

Advanced low-cost sun and earth sensors are under development at the University of Surrey and it is possible these could be included for fine attitude determination. One other option under consideration as a combined attitude determination/science instrument is a free-molecular pressure probe, used as a pitot probe to measure the dynamic pressure on the satellite. The signal from such a device will vary with angle of attack, therefore providing directly useful attitude information. For example the thruster could be commanded to fire only when the pressure probe indicates that the angle of attack is within a certain value. A free-molecular pressure probe (ref.15) consists of a pressure sensor in a gauge volume. The gauge volume is connected to the environment either by a small orifice or a long tube, the pressure with angle function depending on the precise geometry. In varying the angle of attack from 0 to 60 degrees the measured pressure on a typical probe will drop by 50 per cent. The overall accuracy of the instrument will depend on the sensitivity of the pressure sensor used, which will have to measure pressures down to about 10^{-4} Pa or less.

Attitude determination will be carried out on-board, perhaps using Kalman filtering techniques to improve accuracy.

STRUCTURE

The shell of the satellite will probably be constructed, as in previous UoSAT spacecraft, with aluminium honeycomb sandwich. Solar panels are laid on kapton onto the honeycomb, which is usually 6mm thick. Around the edges of the panels magnetorquers are wound. For additional strength, these panels might be mounted on rails running down the corners of the `cylinder`. The top panel will probably also be made of aluminium honeycomb. The configuration at the bottom (thruster end) will probably have to be made somewhat stronger as this will have to transmit the loads from the launcher.

Conveniently, standard UoSAT module boxes (as designed for UoSATs C, D and E) will fit into the available envelope - see figure 13. Room at the corners is tight, so the corners may be bevelled off. The boxes are 330mm square, with a standard depth of 26mm, although deeper boxes are available. Hence, seven such boxes can be easily accommodated in the configuration shown in fig.14.

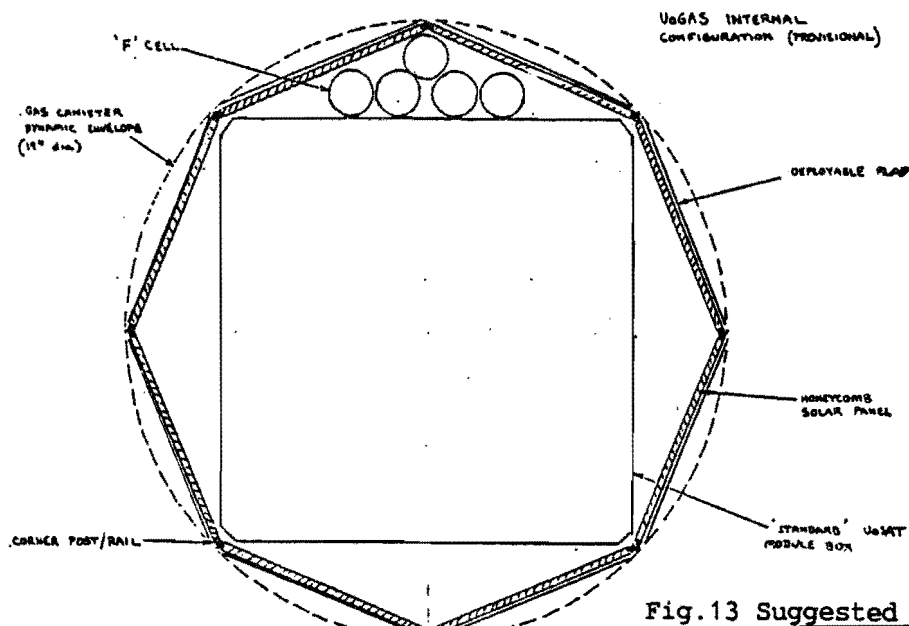


Fig.13 Suggested internal fit

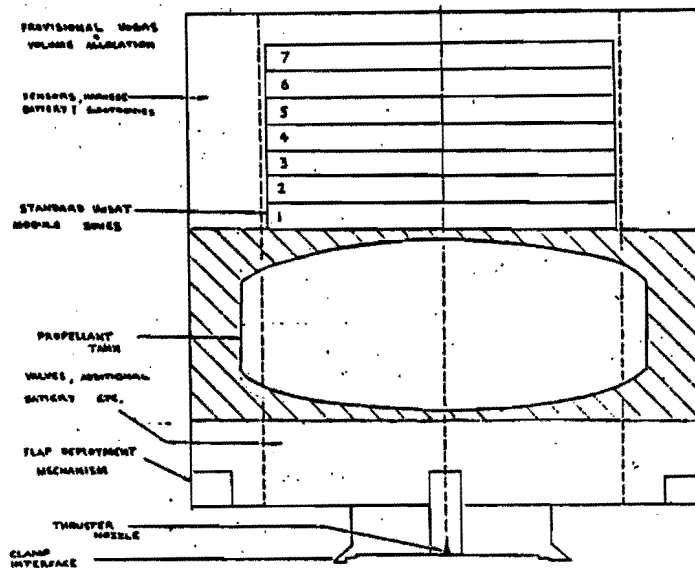


Fig.14 Suggested Configuration

The propellant tank design will depend on the propellant used. Stainless steel is an obvious choice, but this option may have to be ruled out on magnetic compatibility grounds. Even if the spacecraft were degaussed before launch, as soon as the magnetorquers were fired the tank would become magnetised. This might generate unwanted magnetic torques, but more importantly, the measurements from the attitude determination magnetometer would become swamped by the field from the tank. A material such as aluminium is therefore recommended (titanium is an attractive choice, but is expensive.) The internal shape of the tank will be a domed cylinder or an ellipsoid, with a volume of between 12 and 15 litres (depending on the pressure rating, this could be extended somewhat to a little less than 20 litres). It is proposed that the 'outside' of the tank be octagonal and used as a principal structural support, to which the rails and perhaps panels could be attached. It would seem logical to machine the tank in two halves out of solid aluminium, with the two halves bolted together. Appropriate channels would be drilled for cables and propellant tubes, and cut-outs made to reduce mass (within the stiffness and pressure-loading constraints of course.)

The space underneath the tank is reserved for the thruster and its supporting valves and plumbing. The lifetime of satellites of this class is frequently battery-constrained, so an additional battery could be carried here to sustain the extra loads and cycling imposed by the propulsion system. The primary battery could be carried in one of the module boxes, or in one of the gaps between the module box and the satellite wall (see figure 13 above.) If space for electronics becomes tight, it is possible to fit a UoSAT-2 module box in this gap (these boxes are 235x176x30mm and a number are left over from previous missions,) although this would necessitate a cut-out in the propellant tank.

The aerostabilisation flaps will be marginally smaller than the solar panels, to prevent interference with each other and the GAS canister. They do not carry large loads so they can be made quite thin and light. As we shall see later, power is restricted, so there would be some

advantage in covering the flaps with solar cells. This imposes the constraint that the flaps be stiff enough to resist thermal bending which could cause damage to the solar cells. Although heavier than a honeycomb, a couple of millimetres thickness of solid metal should offer adequate stiffness, while being thin enough not to interfere with the GAS envelope. In order to prevent damage to the solar cells during launch, an absorptive blanket would be placed between the flaps and the sides of the vehicle.

Interesting experiments in aerodynamic stabilisation of spacecraft could be conducted if the flaps could be retractable - sweeping the flaps forward into a frustum would increase the restoring torque at low angles of attack, but increases the overall spacecraft drag. (This mode of operation could be of use at higher altitudes where the drag is less of a worry.) However, a suitable motorised system would add much complexity to the system. Reliability concerns would therefore favour a one-shot mechanism such as a spring. One-shot operation would also avoid complications incurred in lubricating the hinge. Before deployment the flaps could be held flush with the walls of the vehicle by wires to be severed with a pyrotechnic bolt-cutter. An interesting low-cost alternative is that used to hold down the Nusat antenna before deployment (ref.16): a nylon fishing line is used to hold down the antenna and release was effected by passing a large current down a nichrome wire to melt the nylon. If this had failed, atomic oxygen impingement would have rapidly degraded the nylon, ensuring eventual deployment. Note that flap deployment on UoGAS is vital - failure to unfold the flaps (which obscure the solar panels) would result in loss of power and hence loss of the mission, so this sort of redundancy is highly desirable.

The skirt behind the main body of the satellite has a number of implications on the spacecraft. The main effects will be on the propulsion system: the expansion of the exhaust from the thruster nozzle will be limited by the skirt, so a residual `atmosphere` will be generated in the skirt region. This will lead to some loss in efficiency of the thruster nozzle itself, but this should be compensated by the small additional thrust generated at the exit plane of the skirt. There will, however, be some viscous losses, but these should not amount to more than a few percent. The impact on the thermal control of the satellite has yet to be investigated, as has the possibility of propellant deposition.

The satellite is designed to fit inside a standard GAS canister, and would also require an ejection mechanism (spring with pusher plate, with the vehicle held down with a marmon clamp) and a motorised lid (FDMDA - Full Diameter Motorised Door Assembly.) In order to simplify ground handling, it is proposed that the propellant fill/vent ports be fitted at the top surface, so that propellant loading can be accomplished if necessary with the vehicle in the canister. Hardpoints for ground handling, and a power/checkout umbilical socket could also be fitted on the top of the vehicle. A large square area (see figure 15) is left, and this will be available for payload. A solar panel

could be fitted here, or other large area devices, such as dust detectors, could be carried.

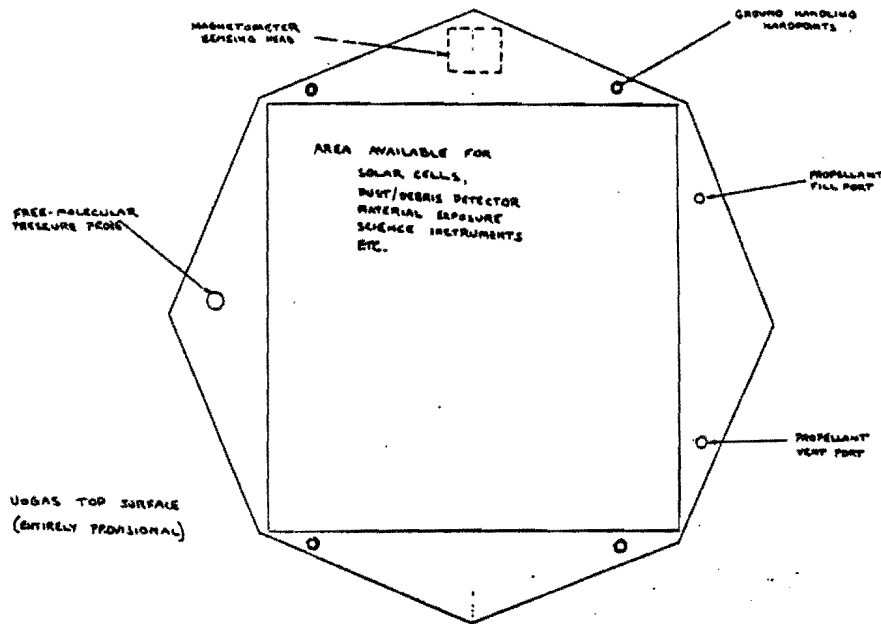


Fig.15 Possible top surface detail

POWER

A major constraint on small satellite operations is the available electrical power. This constraint is particularly pertinent in this application where as much electrical power as possible should be devoted to the propulsion system. The use of higher electrical power levels allows increase in specific impulse and/or thrust. Operation at a higher thrust has the advantage of allowing a wider nozzle throat which would be less likely to be blocked and would also suffer less from viscous losses.

In the early stages of the study it was assumed that a spacecraft power budget of about 20W would be appropriate. However, consideration of the stabilisation mode suggested that power may be a problem as the long axis of the spacecraft (parallel to which the solar panels are mounted) does not spend much time normal to the sun vector. Computer simulations were therefore run to determine the available power for various orbital inclinations. Power profiles for inclinations of 28.5, 57, and 98 degrees are shown in figures 16,17,18. The profiles are based on the long sides (and flaps) of the vehicle having a 75 per cent population of 17 per cent efficient Gallium Arsenide cells - i.e. the maximum power that can be practicably raised. If the flaps do not have cells, the figures should be divided by two. The use of less efficient, but cheaper, silicon cells (say 14% efficiency) would result in a drop of about a fifth. The profiles assume a 300km orbit (for 500km figures should be slightly better owing to the lower fraction of time spent in eclipse.)

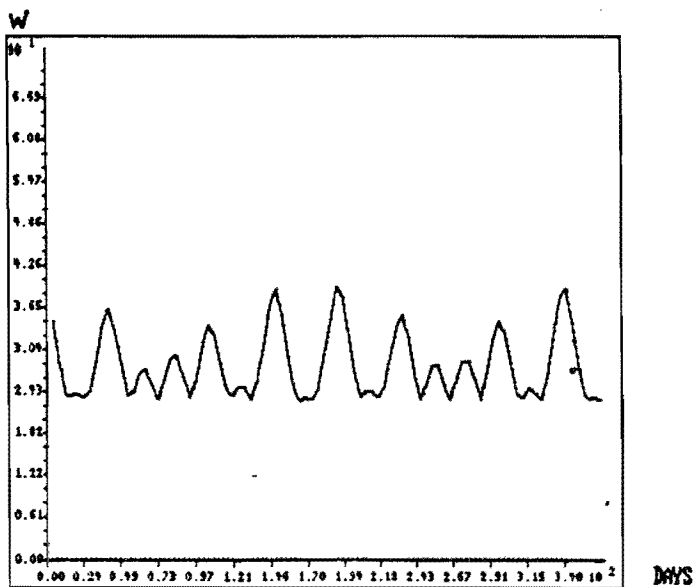


Fig.16 UoGAS Power Profile
 Inclination - 28.5 degrees
 Average Power - 28W

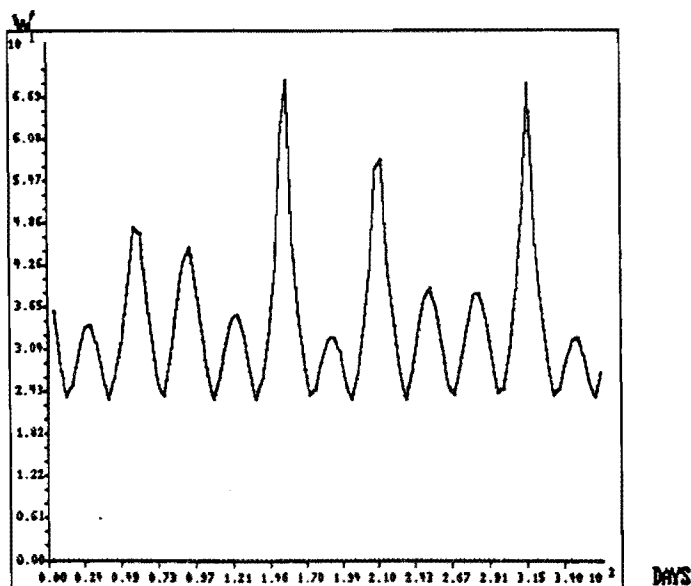


Fig.17 UoGAS Power Profile
 Inclination - 57 degrees
 Average Power - 34W

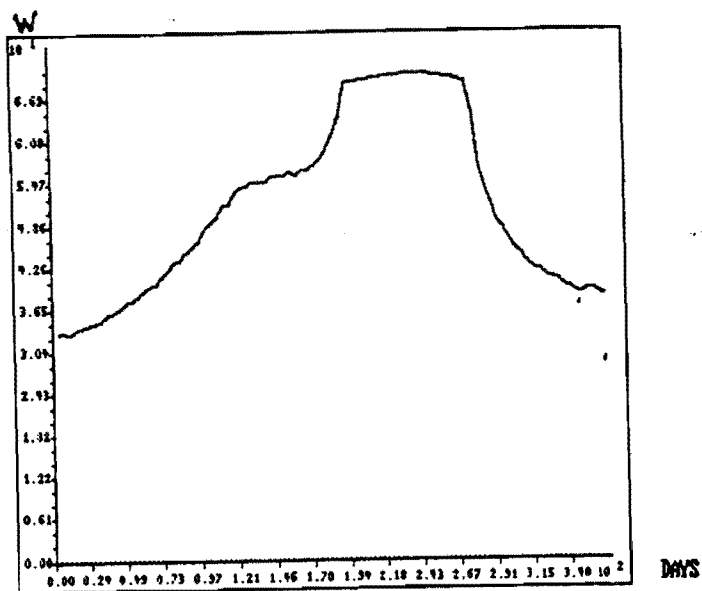


Fig.18 UoGAS Power Profile
 Inclination - 98 degrees
 Average Power - 51W

The advantage of a high-inclination orbit is manifest - a greatly increased average power is provided in polar orbit, with extended periods of high power (corresponding to seasons of dawn-dusk orbits.) A high inclination orbit would also be attractive from a communications point of view - a command station in the UK would be hardly ever have line-of-sight communications with a craft in a 28.5 degree orbit.

STATUS

The UoGAS mission is at present only a paper study. Also there are great uncertainties in the Shuttle programme - not only have shuttle flights yet to resume at the time of writing (August 1988) but the frequency of flights remains in doubt. GAS payloads are not carried on all flights and there is currently a considerable backlog of payloads awaiting flight. Further, the Vandenberg launch facility is not in use, so for the time being 98 degree orbits are not available from shuttle, and the highest inclination available from KSC is 57 degrees and these flights are infrequent. The various safety questions that could strongly influence choice of propellant also remain unresolved.

Despite the low cost of a shuttle launch, it would therefore seem prudent to investigate possible alternative launch methods. There are several small satellite launch vehicles under development, such as Conestoga, ILV, Littleo etc. and a satellite such as UoGAS could readily be accommodated as a secondary payload on larger boosters - sizing the satellite for GAS compatibility allows a very wide range of launch options.

One further idea worth considering is an ISKRA-type deployment. ISKRAS 2 and 3 were small satellites built by students at the Moscow Aviation Institute and launched (by hand) from the airlock of the Salyut 7 space station. A similar type of deployment for UoGAS from MIR would put the satellite into an orbit of reasonable inclination and suitable altitude. However, there are still potential safety problems, as well as perhaps political ones.

The study so far has focussed on the critical areas of attitude control and propulsion. It is assumed that other functions (Telecommand, Telemetry, On-Board Data Handling etc.) can be performed by existing hardware. The first UoGAS mission would probably be regarded more as a test of the propulsion and other systems, but there would appear to be room for some payload.

CONCLUSIONS

The UoGAS study has identified a technically exciting mission, involving release of a satellite into low orbit (300km.) Lifetime extension would be accomplished by raising the orbit to 500km using an electrothermal thruster (resistojet.) The choice of propellant awaits clarification of STS safety and propellant availability issues. Initial

studies show that aerodynamic stabilisation, augmented by magnetorquers, should be able to fill mission requirements.

While the spacecraft would use a number of technologies and components derived from previous and current UoSAT missions, it would stimulate developments in several areas of small satellite technology including propulsion systems and would demonstrate the feasibility of aerodynamic attitude stabilisation.

NOTE

As indicated above, this paper summarises the results of the UoGAS study undertaken by the author at the UoSAT Spacecraft Engineering Research Unit at the University of Surrey. The information presented herein is subject to change and does not indicate formal UoSAT policy.

REFERENCES

1. D G King-Hele 'The Enlargement of Elliptic Satellite Orbits by Continuous Micro-Thrust' Proceedings of the XIVth International Astronautical Congress 1963 Vol.4 pp.155-170 Gauthier-Villiers 1965
2. J R Wertz (ed) Spacecraft Attitude Determination and Control D Reidel 1978 p.820
3. H M Briscoe 'The Choice of Propellant for a Cold Gas Propulsion System for a Satellite' J. Brit. Interplan. Soc. 20 1965-66 pp.72-78
4. A G Bailey 'Colloid Thrusters - A New Space Propulsion System' Electronics and Power 12 July 1973 pp.281-283
5. R D Lorenz Unpublished Communication 26 August 1987
6. M J Harper 'A Short Method for the Calculation of Specific Impulse' J. Brit. Interplan. Soc. 20 1965-66 pp.43-45
7. R Fleeter 'Propulsion System for a Get Away Special Satellite - Pacsat' AIAA 86-1693
8. H Ferguson and J Sovey 'Performance Tests of a 1/2-millipound (2.2mN) Ammonia Resistojet Thrustor System' NASA Technical Note TN-D 4249 1967
9. I Edwards and R E W Jansson 'Performance of Low-Thrust Ammonia Resistojets' AASU Report 307 Department of Aeronautics and Astronautics, University of Southampton 1971
10. NSSDC 'Compendium of Meteorological Satellites and Instruments' NASA-TM-X-69986

11. J Stark 'The Aerodynamic Modelling of Spacecraft for Precise Orbit Determination' in Space Flight Dynamics (Proceedings of 2nd International Symposium on Spacecraft Flight Dynamics, Darmstadt FRG October 1986) ESA SP-255

12. R M Fredo and M H Kaplan 'Procedure for Obtaining Aerodynamic Properties of Spacecraft' J. Spacecraft and Rockets 18 No.4 July/August 1981 p.367 (AIAA 81-4214)

13. R D Lorenz 'Aerodynamic Attitude Stabilisation for a Small, Low-Orbiting Manoeuvrable Satellite' IAF paper ST-88-07 (to be presented at XXXIXth Congress of The International Astronautical Federation, Bangalore, India, October 1988)

14. M S Hodgart and P S Wright 'Attitude determination, Control and Stabilisation of UoSAT-2' J. Inst. Electronic and Radio Engineers 57 No.5 (Supplement) September/October 1987 pp.S151-S162

15. K R Enkenhus, E L Harris and G N Patterson 'Pressure Probes in Free Molecule Flow' UTIA Report No.62 Institute of Aerophysics, University of Toronto 1959

16. C Williams 'NUSAT-1's Layered Protection Software Design' AMSAT-NA Technical Journal 1 No.2 Winter 1987/88 p.32

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