The Solution of Some Problems in the Design of Microsatellite Constellations for Space Plasma Physics

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Abstract

In modern space plasma physics high spatial and temporal resolution in particles and fields measurements has brought new discoveries. Further progress can be achieved if time and space variations can be separated by multi-point measurements. Spatial derivatives of the H-field can also be formed from multi-point measurements and be used to determine the otherwise unobtainable current density **j** through the equation rot **H** =**j**. A constellation of microsatellites can provide such multi-point measurements of particles and fields. Whether for resolving space/time variations or for forming spatial derivatives the satellites in the constellation need station-keeping manoeuvres to maintain or change the inter-satellite configuration.

The satellites therefore need to be equipped with a propulsion system combining simplicity, low cost, high specific impulse and low volume. The use of propane seems to offer a good trade-off between these criteria.

Accurate maintenance of the configuration of the constellation requires good orbit determination. Space-based GPS (SGPS) is possible to use in low earth orbit.

Space plasma physics satellites usually spin, both to keep wire booms taut but also make particle sensors scan in various directions. Two different orientations of the spin axis are common;

Sun-pointing or perpendicular to the orbital plane (cartwheel). The sun-pointing attitude provides power from the sun with the least solar generator area but requires more than two thrusters to allow velocity impulses in the orbital plane and perpendicular to it. The cartwheel orientation provides simple thruster locations while requiring a large solar generator area because of the arbitrary solar aspect angle.

Simple on-board control laws for magnetic actuators can be used for both orientations. For the cartwheel orientation the classical "sign-B-dot" law can be used in high inclination orbits to automatically control the satellites' spin axes. A simple algorithm for sun-pointing control is also possible.

With an instrument payload mass of 8-10 kg each microsatellite weighs 60-80 kg. Two such satellites can brought together to an elliptical orbit with apogee near 10000 km using a solid rocket motor. The launch mass of the two satellites and the kick stage is 250-300 kg allowing a piggyback launch on medium-sized launch vehicles.

1 Introduction

A constellation of satellites can provide multi-point measurements of particles and fields that can separate time and space variations of these plasma parameters. The studies envisaged to be conducted with such satellite constellations require an apogee in excess of 8000 km and inclinations in the range 80° - 110°. It is also important that the separations between the satellites are actively controlled in the range a few kilometres to several hundred 100 kilometres. Repeated manoeuvres to vary such distances need to be made and in the present paper the total velocity impulse that each satellite needs to be able to perform is Δv =160 m/s. The smallest payload needed for obtaining reasonably complete set of plasma measurements is estimated to weigh 8-10 kg and consume about 14 Watts of power.

These requirements are such that microsatellites could be envisaged to fulfil such a mission. However, it is not a trivial task to build microsatellites that are equipped with a propulsion system for frequent orbital manoeuvres combining simplicity, low cost, high specific impulse and low volume as well as adequate on-board navigation systems and autonomous attitude control. This paper explores these design challenges and indicates possible solutions in the form of a design sketch for a two-satellite constellation, which we call the **Constellation Mission** .

As a starting point for the design of the satellites, Swedish Space Corporation's (SSC) Astrid-2 microsatellite has been used. The satellite, designed for plasma physics measurements is depicted in Figure 1 and has the following main characteristics:

Astrid-2	Platform	Payload
Baseline box dimensions (mm)	400 x 400 x 300	
Mass (kg)	21.2	9.4
Average power consumption (W)	16	14
Solar array power (W, EOL)	88	
Battery capacity (Ah)	2.3	
Orbit	h= 1000 km, i=83°	
Orientation	Sun-pointing spin axis	
Mass memory	80 Mbit	
Wire boom length (m)	3.2	
Max. eclipse length (min)	33	
Propulsion system	None	
Precision of timing reference	<1 s with regard to UT	

To adapt Astrid-2 to the requirements imposed by a the Constellation Mission , the following main modifications are required:

- The satellites need to use Space-based GPS (SGPS) receivers for navigation and on-board time-tagging.
- An on-board propulsion system is required on each S/C
- A kick motor is required to raise the apogee height of the two or more satellites in the constellation that are launched together on the same launch vehicle.
- If a cart-wheel mode is required instead of sun-pointing, the solar panel area must be significantly increased

In addition to these major changes there are modifications needed to the present subsystems of the Astrid-2 satellite design For example a larger mass memory is required on each satellite. The 80 Mbit memory on Astrid-2 needs to be increased to 1 Gbit. Also, due to the longer maximum eclipse in an elliptical orbit with a >8000 km apogee a larger battery capacity is required. A different wire boom system for supporting E-field measurements is required. The current wire-boom system on Astrid-2 is limited to approximately 10 m wire length while up to 40 meters length is needed at >8000 km altitude. Also, the Astrid-2 wire booms are wrapped around the satellite body, a design principle which is not suitable for the Constellation Mission. However, traditional wire boom deployment systems are quite heavy and the development of a lightweight deployment system is needed for the Constellation Mission.

The main modifications listed above will be addressed in this paper. In addition, the hitherto unpublished autonomous sun-pointing attitude control laws developed for Astrid-2 are described because they are useful also for the Constellation Mission.

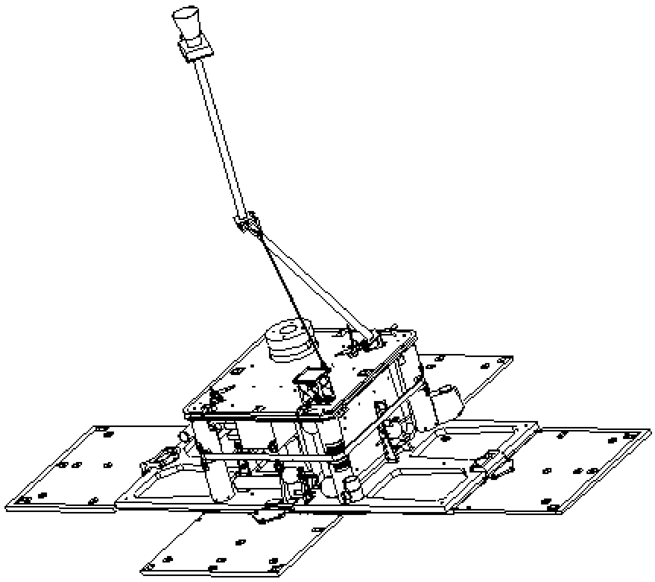

Figure 1 The Astrid-2 microsatellite

2 Timing and navigation

A constellation of microsatellites can provide such multi-point measurements of particles and fields. Whether for resolving space/time variations or for forming spatial derivatives the satellites in the constellation need station-keeping manoeuvres to maintain or change the inter-satellite configuration. Each satellite also needs a stable time reference so that measurements at from different satellites can be correctly correlated in time. Space-based GPS (SGPS) is possible to use in low earth orbit and gives a post-processed position accuracy of the order 50 meters. However, the Constellation Mission requires an apogee in excess of 8000 km. At present SGPS does not operate at such an altitude. Therefore the satellites need to be equipped with ultra-stable oscillators for time reference updated with GPS timing at perigee.

Sunil Bisnath at the Department of Geodesy and Geomatics Engineering of at the University of New Brunswick. describes SGPS time synchronisation in the following way (at his Web-site: **http://gauss.gge.unb.ca/grads/sunil/sgps.htm**): *At the time of GPS signal acquisition by a GPS receiver, the crystal oscillator (usually a TXCO-based receiver clock) is synchronised with GPS time by means of pseudorange measurements and the GPS navigation message. A spacecraft bus architecture can be used to supply GPS-derived time to synchronise all spacecraft subsystem clocks. In this manner, daily clock updates from ground stations will not be required and precise time-tagging of downloaded satellite data will be possible. Once receiving GPS signals, a GPS receiver is capable of a conservative GPS time clock synchronisation accuracy of approximately 300 nanoseconds with SA engaged. This accuracy is quite sufficient for many satellite missions. Synchronisation [with the onboard local clock] is usually accomplished through the use of a feedback loop and an appropriate filter. A computer tracks the difference between GPS time (or UTC) and the local clock, and uses these data to steer the local clock by means of e.g., a phase micro-stepper.*

Since GPS position data is not available at all points along the orbit, GPS state vectors near perigee need to be propagated to the high altitude parts of the orbit by integration of the equations of motion in order to achieve the required position determination accuracy of 1 km (1σ).

3 Propulsion

The satellites need to be equipped with a propulsion system combining simplicity, low cost, high specific impulse and low volume. Three types of on-board propulsions systems have been considered:

- A mono-propellant hydrazine system operating in blow-down mode with initial pressure 20 bars
- A nitrogen cold gas system with initial pressure 240 bars
- A propane cold gas system with fluid propane stored in a temperature-regulated tank

The ΔV-requirement for each spacecraft is assumed to be 160 m/s. The table below shows the approximate propulsion system masses for the different systems assuming a spacecraft dry mass of 46 kg (sun-pointing spinner).

Propulsion system	Typical specific impulse (s)	Propellant mass (kg)	Prop. system dry mass (kg)	Prop. system total (kg)	Tank $\varnothing_i$ (m)
Hydrazine (N_2H_4)	200	3.9	4.5	8.4	0.24
GN_2	70	12.0	17.5	29.5	0.44
Propane (C_3H_8)	61	14.1	3.7	17.8	0.39

A hydrazine system has the least mass but has also the highest cost (approximately 0.5 million US dollars per spacecraft). The safety regulations are also higher than for the other two systems. A gaseous nitrogen system presents the simplest solution but has a severe mass penalty. The main contributor to the dry mass is the tank which has a mass of approximately 22 kg (7 mm thick Grade 5 Ti, 240 bars with Safety Factor = 2). The suggested baseline is the propane system which represents a compromise between the other two systems. This type of system will be discussed at some depth below.

In a propane cold gas system liquid propane is stored in a tank (Figure 2). The tank volume is initially filled to approximately 80% with liquid propane and the remaining volume with propane gas. The pressure of this gas is strongly temperature-dependent and is determined by the vapour pressure diagram for propane (Figure 3).

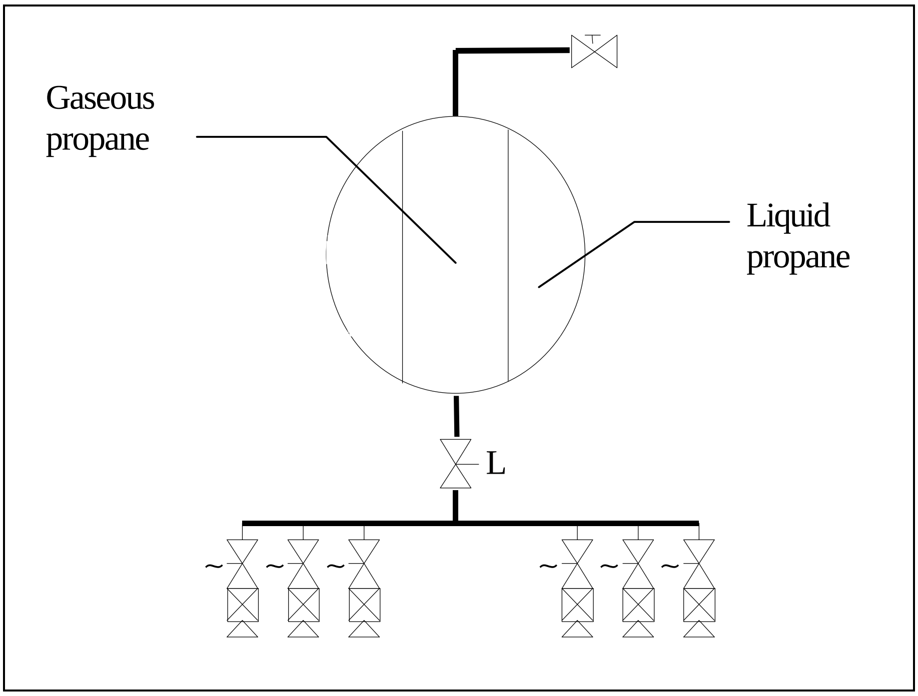

Figure 2 Propane system

The thrust provided by the system is proportional to the tank pressure, but a high tank pressure requires a heavier tank and a higher fluid temperature. A suitable compromise is a tank pressure of 10 bars, giving a fluid temperature of +28° C.

As gas is taken out of the tank, new gas must be generated. This requires a significant amount of vaporisation heat: 470 kJ/kg. To generate 6 N of thrust, approximately 9 grams must be evaporated each second, requiring 470*9 = 4.2 kW of heat. In practice it is impossible to compensate for this heat loss, so the effect is that the temperature of the propellant drops quite rapidly: 0.1-0.5 K/s depending on how full the tank is.

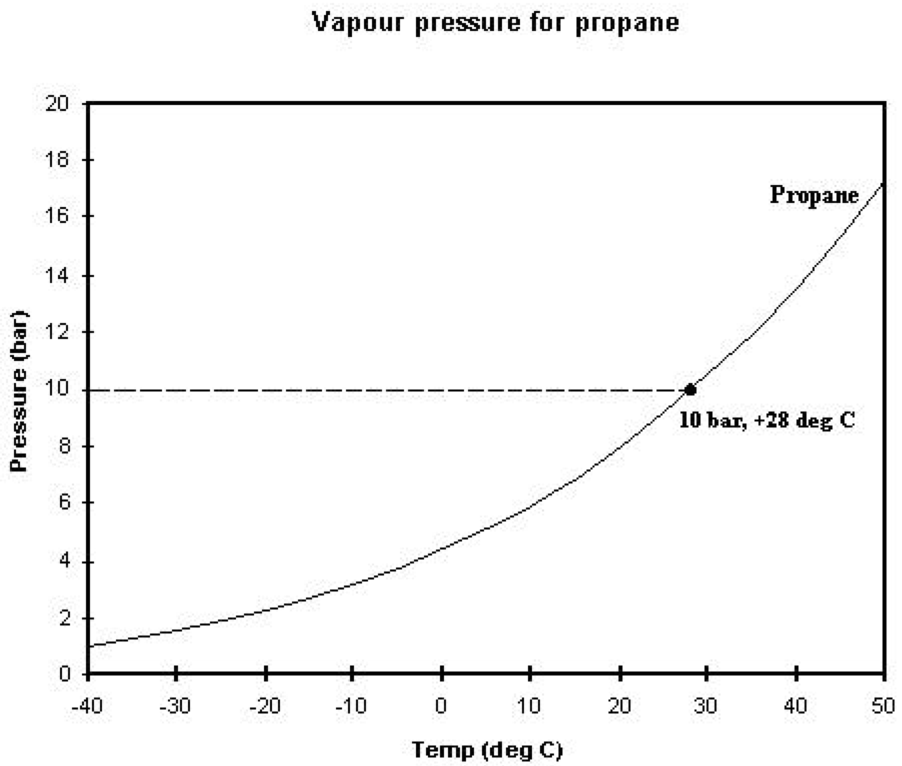

Figure 3 Vapour pressure curve for propane

Assuming that the propellant pressure (and thereby the thrust) is allowed to drop by 50% during a ΔV firing, the vapour pressure diagram gives the following information:

Pre-firing propellant pressure and temperature: 10 bars, +28°C
Post-firing propellant pressure and temperature: 5 bars, + 4°C

As can be seen, the propellant temperature drop is 24 K, which is quite significant. Assuming that a 6 N (@ 10 bars) thruster is used, the table below shows the propellant cooling rate, the amount of ΔV that can be generated during each firing and the time it takes to re-heat the propellant after each firing (assuming a 10 W tank heater).

Fluid remaining in tank	Rate of propellant cooling in tank during thruster firing	ΔV generated during firing (m/s)	Time to re-heat propellant to +28 °C (hrs)
100% (14 kg)	0.13 °/s	18.5 m/s	23.5
50 % (7 kg)	0.25 °/s	10.6 m/s	11.7
25 % (3.5 kg)	0.50 °/s	5.8 m/s	5.9

Although adequate for the envisaged ΔV manoeuvres, a cartwheel stack would require active nutation damping (using thrusters) before and after the firing of the kick motor. Further analysis will show if the nutation damping will consume more propellant than can be supplied by the propane system. If so, at least one of the S/C will probably have to be equipped with a hydrazine system.

4 Attitude control laws

In a satellite constellation the satellites need to be as autonomous as possible in order to reduce the need for support from the ground. This is particularly true in the case of attitude control. These satellites are spin-stabilised and their spin axes need to point in the same direction (to easier correlate measurements). Autonomous on-board control systems that can keep the spin axes pointed in the same direction are needed. Space plasma physics satellites usually spin, both to keep wire booms taut but also make particle sensors scan in various directions. Two different orientations of the spin axis are common; Perpendicular to the orbital plane (cartwheel) or Sun-pointing.

Simple on-board control algorithms using magnetic actuators can be used for both orientations. For the cartwheel orientation (spin axis aligned to the orbit normal) the classical "sign-B-dot" law can be used in high inclination orbits to automatically control the satellites' spin axis. In this algorithm the projection of the Earth's magnetic field along the spin axis is measured. The sign of the time derivative of this measurement is used to determine the polarity of the magnetic dipole moment of a magnetic torque coil with its dipole moment aligned with the spin axis. By applying this control law the spin axis aligns with the orbit normal[1].

A simple control law algorithm for automatically pointing the spin axis in the direction of the sun using a sun-sensor, a magnetometer and magnetic actuators is also possible. A Swedish patent application (no.) for this algorithm has been submitted. The principles of the control law are described in Appendix 1.

5 Spacecraft configurations

In the cartwheel mode, the angle between the spin axis and sun direction changes as the orbital plane changes its orientation in inertial co-ordinates because of nodal regression caused by the J_2 term in the Earth's gravitational field model. On a spinning spacecraft with many appendages it is very difficult to have sun-tracking solar panels. Fixed solar panels are therefore assumed here. To generate the same power, the solar panels on a on a satellite spinning in the cart-wheel mode need to have three times larger area compared to those of on a sun-pointing spacecraft. This is a serious drawback of a cartwheel mode of orientation, which drives the overall S/C size which in turn has an impact on the mass budget. The implications are shown below.

In this analysis we assume that the satellites in the constellation are launched together into orbit stacked on some kind of adapter. We also assume that this stack will be dropped off in a high-inclination circular parking orbit (taken to be circular at 804 km) as would be the case when being launched as a piggy-back payload on a sun-synchronous mission. Thiokol's catalogue of solid rocket motors has been used to identify suitable kick motors to raise the apogee from 804 to ≥8000 km. In particular, the capacity of the motor to raise the apogee to 9484 km has been evaluated. In this orbit the satellite makes precisely seven sidereal revolutions per day. It is found that the smallest motor that can be used is the STAR 17A which has approximately 114 kg of propellant. As this motor is slightly over-dimensioned with respect to the needs and a mass margin is available which can be used to incorporate e.g. traditional, rather heavy, wire booms systems.

5.1 Sun-pointing satellite

If sun-pointing spacecraft are considered acceptable, the main modification to Astrid-2 is to increase the base area (and volume) in order to accommodate the propane tank and mass memory. This involves increasing the main box dimensions from approximately 0.4x0.4x0.3 on Astrid-2 to 0.55x0.55x0.4 for the Constellation Mission. The approximate spacecraft dimensions including the kick-stage and its excess capacity are shown in Figure 4. The satellites are small enough so that they can be mounted side by side on kick stage. In this way the stack can be made oblate so that active nutation damping will not be required in connection with the kick-motor firing. As can be seen, there is a 50 kg mass margin to the 804x9484 km orbit providing 7 sidereal revolutions per day.

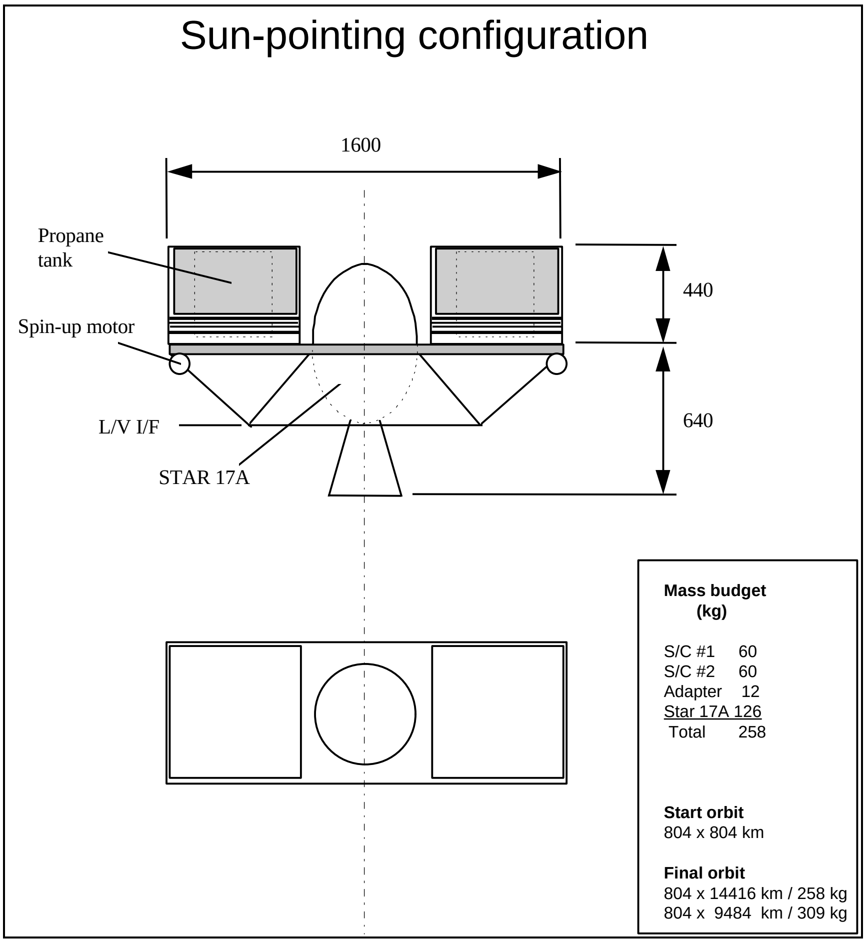

Figure 4 Sun-pointing spacecraft

5.2 Cartwheel configurations

If a cartwheel configuration is selected, the solar panel area needs to be increased by a factor of three compared to a sun-pointing spinner. The satellite mass increases to about 80 kg. As shown in Figure 5, this increases the overall dimensions of the spacecraft beyond those of the sun-pointing version in order to accommodate the larger solar panel area. The increase in diameter also means that the satellites cannot be mounted side by side, but rather have to be mounted on top of each other. Figure 5 also shows that the STAR 17 motor which is one step smaller in size than the STAR 17A has insufficient capacity to raise even a stripped spacecraft stack (Propulsion on only one satellite and only 6000 km apogee - Case 2 in Figure 5). Using a STAR 17A motor (Case 1 in Figure 5), the mass margin to the 804x9484 km orbit is 10 kg. Active nutation damping during the kick-motor phase will be required, so it may be necessary to equip one of the S/C with a hydrazine propulsion system.

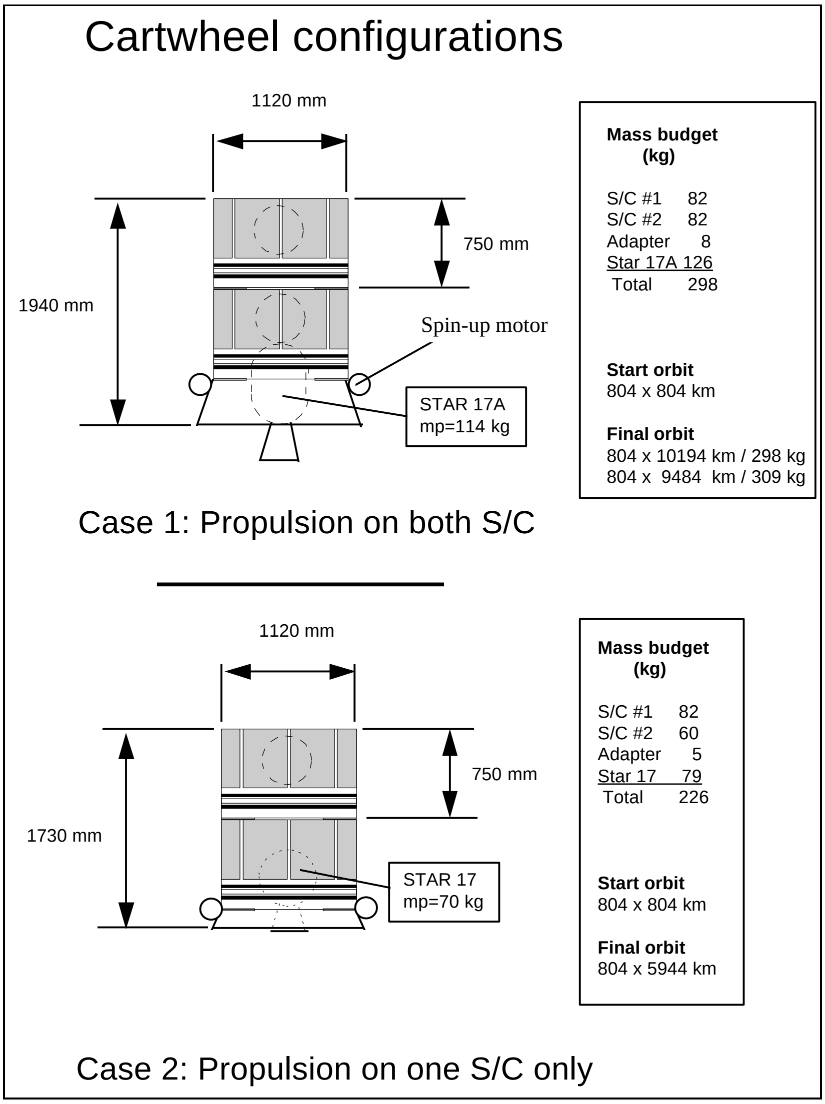

Figure 5 Cart-wheel configurations

6 Summary

The Constellation Mission is possible to carry out with microsatellites (The University of Surrey, a pioneer in microsatellite engineering, defines a microsatellite as weighing less that 100 kg) because the minimum required satellite mass is about 60 kg for the sun-pointing version and about 80 kg for the cartwheel version. The mass and power allocation to the payload is 8 kg and 14W respectively. The overall mass of the stack of two satellites plus a kick motor to move them to an elliptical orbit with an apogee above 8000 km is approximately 300 kg. This reasonable launch mass opens possibility of using piggyback opportunities on major launch vehicles such as the Ariane 5.

7 References

[1] Hodgart M.S., Attitude control and dynamics of UoSAT angular motion , The Radio and Electronic Engineer, Vol. 52, No. 8/9, pp 379-384, August/September 1982.

Appendix 1 An On-board Algorithm for Automatic Sun-pointing of a Spinning Satellite

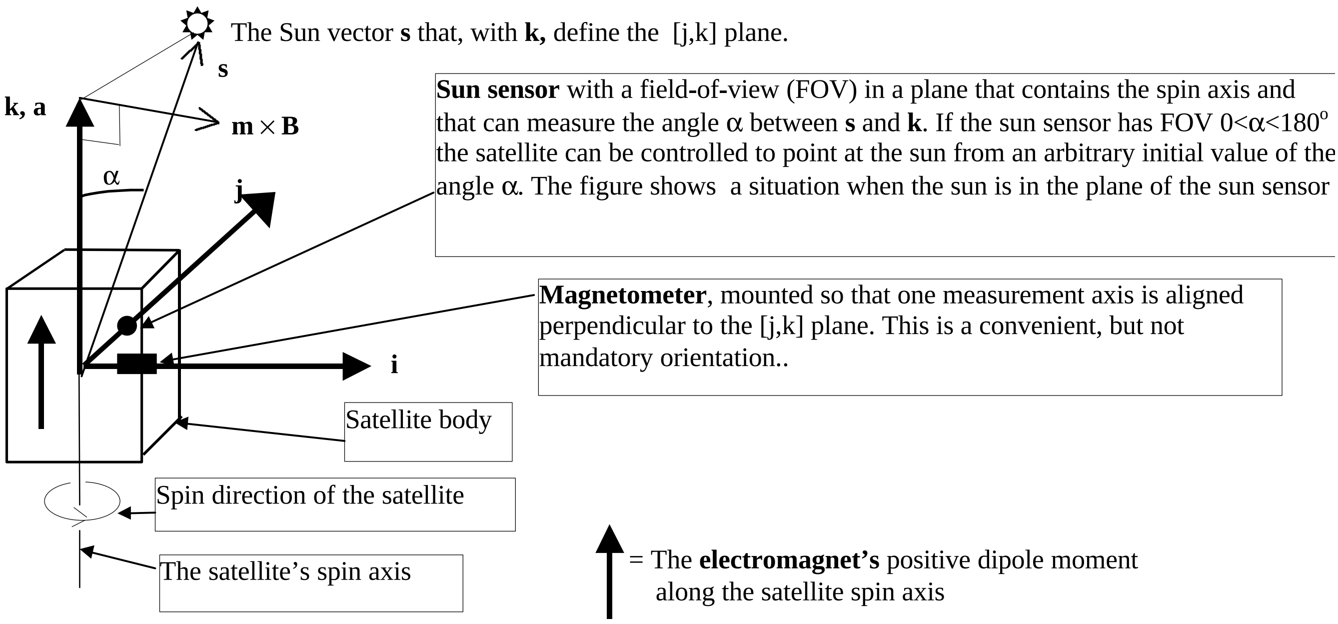

Figure 6 Orientation of sensors and actuators

Earlier methods to point the spin axis of a satellite in the direction of the sun have been based on a two-stage process: 1) the determination (on board the satellite or on the ground) of the orientation of the spin axis in inertial coordinates using e.g. a sun sensor and a magnetometer; 2) computation onboard the satellite (or on the ground) of the suitable current direction through the electromagnet and the instants of reversing this direction so that the interaction of the electromagnet with the Earth's magnetic field would turn the spin axis towards the sun. The disadvantage with this traditional method is that it requires the explicit computation of the orientation of the spin axis in inertial coordinates. The method described here requires that neither orientation of the spin axis or the sun in inertial coordinates need to be determined or known. The present method uses a sun sensor, a magnetometer and an electromagnet, the magnetic dipole moment of which is parallel or anti-parallel to the spin axis and can change sign when the current through the electromagnet is reversed. The angle between the spin axis and the direction of the sun need to be measured onboard the satellite. <u>The direction of the current through the electromagnet shall be such that the magnetic dipole moment of the electromagnet along the spin axis has the same sign as the component of the earth's magnetic field along an axis perpendicular to the plane defined by the spin axis and the direction towards the sun.</u> Only a simple logic, implemented in hardware or software, is needed onboard the satellite. The sun sensor provides an indication when the sun passes through the plane in which its field-of-view is defined and this indication defines this plane. The sun sensor also measures the angle between the spin axis and the direction to the sun. The following quantities are used below in the derivation of this control law:

a	the spin axis orientation (unit vector). **a** is parallel to the unit vector **k**.
B	the Earth's magnetic field (vector)
B	the scalar magnitude of the Earth's magnetic field
b	a unit vector parallel to the earth's magnetic field vector with components b_x , b_y, b_z where the axes x,y,z correspond to unit vectors **i,j,k.**
i,j,k	are unit vectors in a Cartesian coordinate system with the origin on the satellite's spin axis. The [j,k] plane is defined by the spin axis and the direction to the sun (unit vector **s**).
L	the satellite's angular momentum (vector)
L	the scalar magnitude of the satellite's angular momentum
M	the magnetic dipole moment vector of the electromagnet

m a unit vector parallel to the magnetic dipole moment vector of the electromagnet

s direction to the sun (unit vector)

T the torque vector acting on the electromagnet due to its interaction with the Earth's magnetic field

T the scalar magnitude of the torque vector acting on the electromagnet due to its interaction with the Earth's magnetic field

t a unit vector parallel to the torque vector acting on the electromagnet due to its interaction with the Earth's magnetic field

u the polarity of the dipole moment of the electromagnet

The control law is based on that:

- the sun sensor is designed such that it has a field-of-view in a plane directed out from the spin axis (and parallel to the spin axis) and provides an indication when this field of view passes the [j,k] plane. (See Figure 6)
- the sun sensor measures the angle α between the direction to the sun and the satellite's spin axis when its field-of-view passes the [j,k] plane. (The sun angle α is larger than zero by definition.)
- the magnetometer is designed such that it can determine the sign of the component of the earth's magnetic field perpendicular to the plane defined by the spin axis and the direction to the sun (the [j,k] plane).
- the satellite is equipped with an electromagnet the magnetic dipole moment of which, **M** (unit vector: **m**), is parallel or anti-parallel to the spin axis can change sign if the current through the electromagnet is reversed, i.e.

$$\mathbf{m} = u\,\mathbf{k} \qquad \textit{(1)}$$

Where u = { the polarity of the dipole moment of the electromagnet } = ± 1.

A unit vector parallel to the Earth's magnetic field is called **b** and has the components b_x , b_y, b_z where the axes x,y,z correspond to the unit vectors **i,j,k**. The scalar magnitude of the Earth's magnetic field is called B.

From elementary electrodynamics we know that the electromagnet is subject to a torque **T** the direction of which (unit vector **t**) is the vector product of magnetic dipole moment vector and the magnetic field vector of the Earth:

$$\mathbf{T} = T\,\mathbf{t} = B\,\mathbf{m} \times \mathbf{b} = u\,B\,(0,0,1) \times (b_x , b_y, b_z) = \\ = u\,B\,(-b_y, b_x, 0) \qquad \textit{(2)}$$

The spin motion of the satellite is assumed to take lace without nutation or other deviations from a pure rotation. The angular momentum, **L**, of the satellite is then assumed to be:

$$\mathbf{L} = L\mathbf{a} \qquad \textit{(4)}$$

Classical mechanics states that angular momentum should be conserved, i.e:

$$d\mathbf{L}/dt \equiv \mathbf{T} \qquad \textit{(4)}$$

T is always perpendicular to **L** since **L•T**= =BL**a•**(**m** ×**b**)={**a** and **m** are parallel}= =BL**a•**(**a** ×**b**)≡0. So, the scalar magnitude of the angular momentum is not changed by the torque of the electromagnet, only its direction. Thus, (4) can be rewritten:

$$L\,d\mathbf{a}/dt \equiv \mathbf{T} \qquad \textit{(5)}$$

One can visualize the time derivative of the spin axis vector as a vector originating from the tip of **a** with the scalar magnitude T/L, i.e d**a**/dt can be said to make **a** turn around an axis defined by **a** ×d**a**/dt. If this vector has a component parallel to the normal of the plane defined by **a** and **s** the vector **a** tends to align with **s** and α is reduced. The expression for the normal is **a** × **s.** The condition that **a** ×d**a**/dt and **a** × **s** must be as parallel as possible can be expressed by stating that they must not have any component that is antiparallel (a vector **p** is antiparallel to the vector **q** if **p** = -**q**), i.e.:

$$(\mathbf{a} \times d\mathbf{a}/dt) \bullet (\mathbf{a} \times \mathbf{s}) > 0 \qquad \textit{(6)}$$

In this equation

$$\mathbf{a} \times \mathbf{s} = (0,0,1) \times (0, \sin\alpha, \cos\alpha) = \\ = (-\sin\alpha, 0, 0) \qquad \textit{(7)}$$

and

$$\mathbf{a} \times d\mathbf{a}/dt = \mathbf{a} \times (\mathbf{T}/L) = \\ = (0,0,1) \times u\,(B/L)(-b_y, b_x, 0) = \\ = u\,(B/L)\,(-b_x, b_y, 0) \qquad \textit{(8)}$$

Inserting (7) and (8) in (6) we obtain

$$(\mathbf{a} \times d\mathbf{a}/dt) \bullet (\mathbf{a} \times \mathbf{s}) = \\ = (-\sin\alpha, 0, 0) \bullet u\,(B/L)\,(-b_x, b_y, 0) = \\ = \sin\alpha\,(B/L)\,u\,b_x > 0 \qquad \textit{(9)}$$

Since B>0, L>0 and α>0 are always true statements this expression can only remain >0 ***if, and only if, u and*** b_x have the same sign.

Therefore the control law can be formulated:

The passage of the field-of-view through the [j,k] plane is used to determine the sign of the Earth's magnetic field in a direction, **i** , perpendicular to the [j,k] plane (**i,j,k** are unit vectors in a Cartesian coordinate system with the origin on the spin axis of the satellite). The direction of the current through the electromagnet shall be such that its magnetic dipole moment along the satellite's spin axis ($\mathbf{a} \equiv \mathbf{k}$) has the same sign as the component of the Earth's magnetic field along the axis **i**. If the magnitude of this component is very small the current through the electromagnet is turned off.

The control law remains in effect until the measured value of α has decreased to the desired value, e.g. the smallest value when the suns sensor still provides a sufficiently accurate indication of when its field-of-view passes the [j,k] plane (when $\mathbf{s} \equiv \mathbf{a}$ the [j,k] plane is not defined). This control law is implemented in logic onboard the satellite than can read out the sun sensor and the magnetometer and control the current through the electromagnet.