A Multi-Spacecraft Magnetospheric Monitoring Mission Study

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Abstract. We describe the results of an ivestigation of the feasibility of creating a system for monitoring the global magnetospheric response to solar wind behaviour. The study was performed by a group of 20 MSc (postgraduate) level students. Our goals were to investigate three mission scenarios using increasing numbers of spacecraft. The scenarios were selected to illustrate general issues associated with creating constellations to make global scale measurements of the magnetosphere.

1. Introduction

There is a growing interest in using large numbers of spacecraft to collect simultaneous in-situ measurements from points distributed throughout the magnetosphere, or at least from within key magnetospheric regions, to study the global behaviour of the magnetosphere. Although there are many science issues which can be addressed by global scale multi-point measurements, issues such as cost and orbital dynamics will typically constrain the focus of a single mission design to address a single scientific problem well, and perhaps to take on other issues as secondary goals.

This paper describes a study of how to carry out a relatively ambitious constellation mission, using existing technologies. The aim was to investigate the problems and estimate the costs of such a mission. The study was also intended to investigate possible economies of scale as a function of constellation size, and so it considers three stages in the development of a constellation. The use of existing technologies should produce a worst case (and interesting benchmark) cost estimate for the concept. Suitable use of rapidly advancing technologies will undoubtedly allow significant cost savings in the future. The philosophy of the study thus contrasts with that of most "real-world" proposals which start with a cost cap and seek to achieve a mission within that constraint, sometimes by using relatively risky new technology.

Each constellation configuration was selected for study according to specific scientific objectives, but the orbits have not been meticulously optimized for addressing these objectives. Instead, the configurations were intended to illustrate a range of issues common to this class of missions, in the hope that the lessons learned might be applicable to a range of more focussed mission designs.

The detailed work was performed by a group of 20 postgraduate students in an intensive, six week interval. The students worked from a single briefing document, and apart from weekly interactions with a staff member, they worked independently to produce a technical report. The report was presented to and subsequently reviewed by a panel composed of members outside of University College London.

2. Framework for the Study

Our working mission specifications include specific constellation configurations and a minimal payload in order to address predefined science goals at a minimum cost. The study goal was the definition of several critical mission components, including spacecraft, payload, launch, mission operations, data recovery, and spacecraft position-finding. Data validation, dissemination and archiving issues were left for future studies.

2.1 Science Goals

The mission science questions (in priority order) are:

1. What is the shape and dynamic evolution of the magnetospheric outer boundary regions (bowshock, magnetopause, cusp) under variable solar wind conditions?
2. What are the global magnetospheric magnetic field and magnetospheric plasma flow variations for a range of magnetospheric activity levels?
3. What is the large-scale evolution of the nightside magnetic field and plasma during substorms?

2.2 Constellation Configuration

The project specification required that three scenarios were studied. Each scenario required that full-time solar wind monitoring was available, preferably from the region immediately upstream of the bowshock. Each scenario also required spacecraft in orbits whose apogees fell (at a given epoch) into specified Local Time (LT) regions. Each LT region was to be populated by at least one spacecraft in an equatorial orbit, and at least one spacecraft in a highly inclined orbit. We were required to specify orbital characteristics such as apogee, perigee and inclination, the number of spacecraft to place in each orbit and how to phase them relative to one another and to spacecraft on other orbits. These decisions are of course scientifically movtivated, but also affect communications and launch strategy. The LT coverage to be provided by the magnetospheric survey spacecraft was specified for a given epoch as follows:

Scenario 1:	Cover LT Sectors: 00, 06, 12, 18.
Scenario 2:	As Scenario 1, plus 03, 09, 15, 21 LT
Scenario 3:	As Scenario 2, plus 22, 23, 01, 02 LT

The layout of the equatorial orbits chosen is illustrated in Figures 1 to 3. The orbital coverage of Scenario 1 gives a basic ability to address the first two science goals. Scenario 2 gives better spatial resolution and coverage. The symmetry of Sceneria 1 and 2 minimizes the seasonal dependence of regional coverage associated with the annual rotation of the constellation about the Earth, and maintains global coverage throughout the year. Scenario 3 enhances resolution in a single LT quadrant, and is intended to better support the third science goal. The use of three cases, each building on the last, was intended to reveal any economies of scale associated with building up constellations using a collection of standard orbits. In all three scenaria, it was required that full-time solar wind monitoring should be available. The mission specification also required that mission operations should last at least one year from the time that all spacecraft are in place and commissioned.

Science Closure and Enabling Technologies for Constellation Class Missions, edited by V. Angelopoulos and P. V. Panetta, pp. 36-43, UC Berkeley, Calif., 1998.

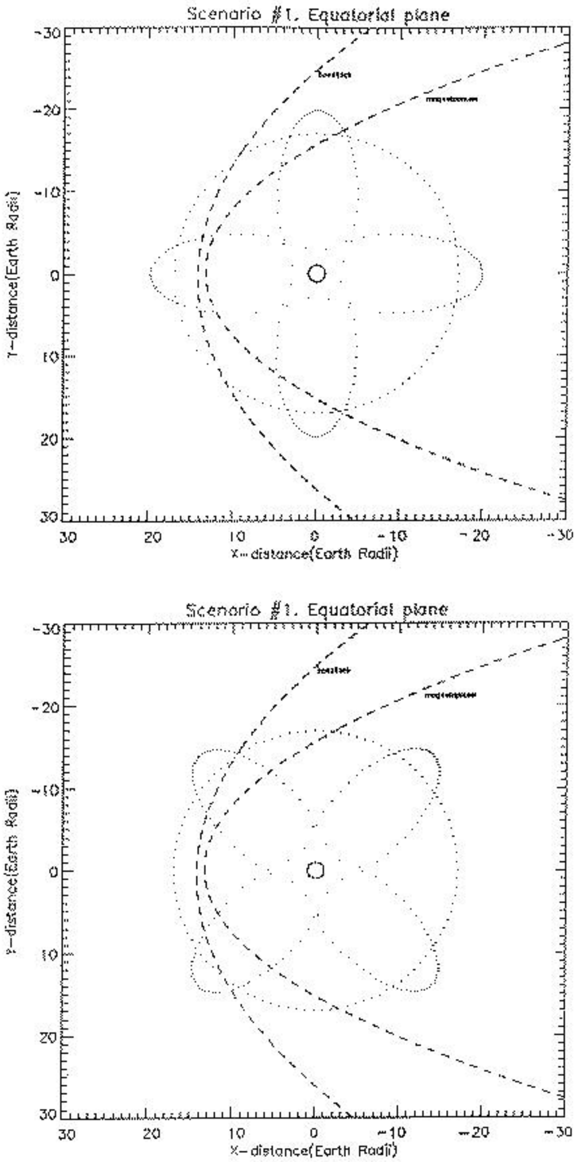

Figure 1. Illustration of Local Time coverage for Scenario 1 by the equatorial-plane-orbiting magnetospheric spacecraft, and also showing the solar wind monitoring orbit. The upper and lower figures illustrate the rotation of the constellation about the Earth as time passes. The projection of inclined plane magnetospheric spacecraft orbits is not shown. Note that the inclined orbits rotate about Earth at a different rate than the equatorial orbits. Model magnetopause and bowshock positions are shown.

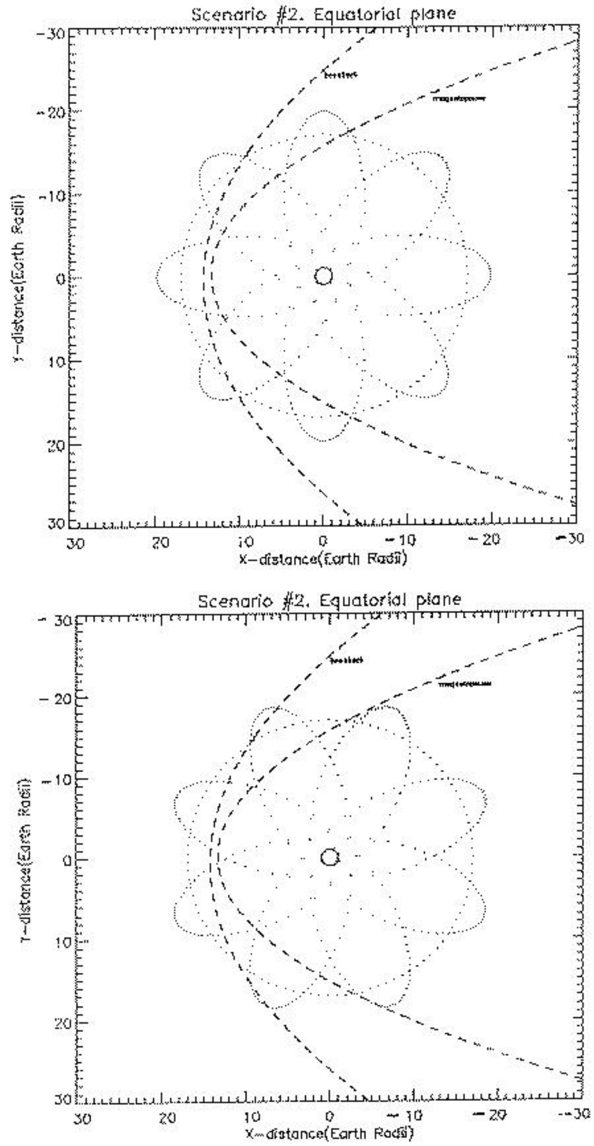

Figure 2. Illustration of Local Time coverage for Scenario 2., using the same format as Figure 1.

2.3 Spacecraft and Payload

Taking advantage of the economy of scale necessitates that a single spacecraft and payload design be adopted for all spacecraft. The spacecraft should be of simple design, have low mass and volume, be cheap to build, and should operate for at least a year after completion of the constellation. If it takes a year to assemble the constellation, there is either an implicit requirement for a two year spacecraft lifetime, or for the ability to replace a failed spacecraft to maintain the constellation's integrity. The question arises whether it is more cost-effective to produce a small number of relatively expensive but reliable spacecraft, or to produce a larger number of less expensive and perhaps less reliable spacecraft on the assumption that spacecraft which fail can be replaced by on-orbit spare spacecraft. The minimum science payload was to consist of a magnetometer and an E/q type ion analyzer. The payload was constrained to require no more than 3 kg and 5 Watts (excluding a possible magnetometer boom).

3. Description of the Study Mission Design

3.1 Solar Wind Monitoring Orbit

Several approaches to solar wind monitoring were considered. Some solar wind coverage can be provided by satellites from the magnetospheric constellation, while they are near apogee. However, each dayside satellite spends only part of its orbit in the solar wind, and the fraction of time in the solar wind reduces as the apogee LT gets further from noon. If all satellites were phased to reach apogee together, the

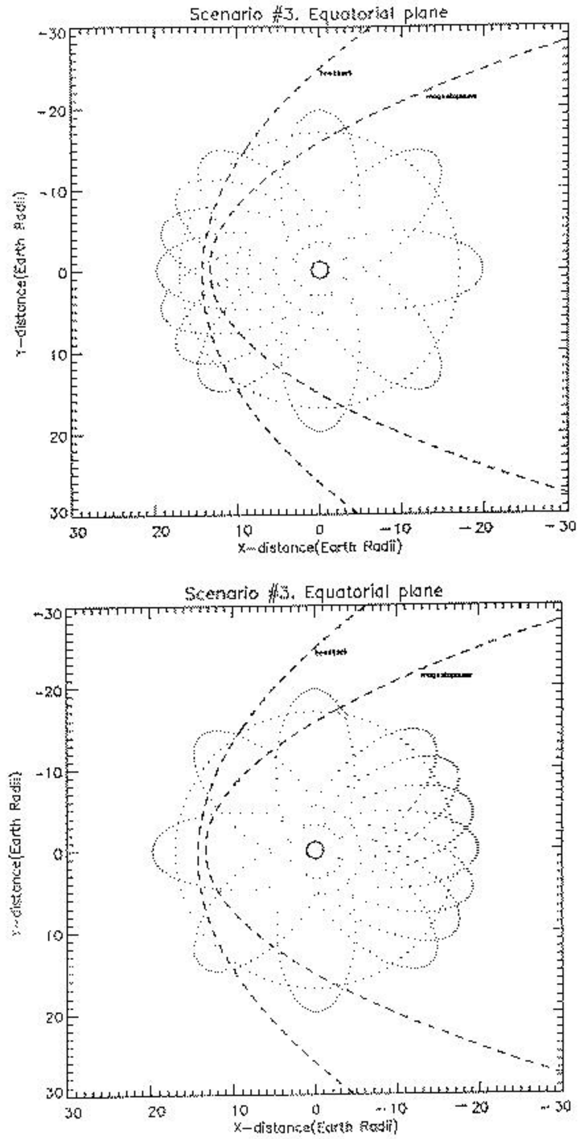

Figure 3. Local Time coverage for Scenario 3. Same format as Figure 1.

near-noon satellites would provide solar wind data while other dayside satellites made boundary crossings. With a single satellite per orbit, the total constellation solar wind coverage would only be about 50% to 60% of the time, so that the design requirement of 100% coverage is not met. Raising the apogee of those satellites would improve the proportion of solar wind coverage, but would undesirably reduce the frequency of their passages through the magnetosphere.

An L1 halo-orbit monitor suffers from undesirable uncertainty in estimation of propagation time to the magnetosphere and would be expensive to launch. Also, an L1 spacecraft could not be identical to the magnetospheric spacecraft as it would need (at least) a more powerful transmitter and a correspondingly enhanced power generation system. This spacecraft would probably have to be unique, so that extra manufacturing care (and expense) would be needed to ensure a low probability of failure. Instead of taking advantage of mass-production cost-savings, an L1 satellite might introduce significant additional design and build costs, on top of extra launch costs.

Thus a third approach was chosen, which was to use four satellites (of the same design as the magnetosphere survey spacecraft) in an equatorial circular orbit of radius 17 R_E, period 4 days 2 hours. The satellites are spaced 90° apart. The spacing guarantees that one, and often two satellites are in the solar wind at any time, even for extremes of bowshock motion. Under average conditions, three satellites are sufficient for full coverage. Thus if one satellite fails, the other three could be rearranged at 120° apart and still provide good coverage most of the time. A lower altitude orbit would give poorer coverage, while a higher one would be more expensive to achieve. When not in the solar wind, the satellites are useful magnetotail monitors.

3.2 Magnetospheric Survey Satellite Orbits

The magnetospheric monitoring satellite orbits were chosen to have perigees at 1000 km altitude, apogee distance of 19.8 R_E, and thus a period of 2 sidereal days. These parameters were chosen for several reasons. The orbital period was originally chosen to be a multiple of a sidereal day in order to simplify communication to ground. Initially it was thought that with this orbital period, a satellite would return to the same point above the Earth at each perigee, but it was later recognized that orbital precession prevents this. Worse still, the nodal and apsidal precession of the inclined orbits is different from that of the equatorial orbits. The perigee was chosen to be above altitudes where atmospheric drag is a serious problem (with the potential to alter the orbits) and below the radiation belts, to allow low altitude communications without endangering the transceiver electronics. Perigee communication was the initial strawman approach, although it was later discarded. The apogee was required to lie some distance beyond the average bowshock location so as to guarantee that plenty of orbits crossed the bowshock, even away from noon LT. However, a very large apogee distance would reduce the orbital period and the frequency of boundary crossings (a 3 day orbit would have apogee near 27 R_E). Given the constraints on perigee and orbital period, a 19.8 R_E apogee was the natural choice.

The inclined orbits were intended to survey higher magnetic latitudes, either north or south, and were the source of considerable discussion (see Figure 4). Three options were examined. One was a "Molniya-like" orbit, with an inclination of 63.4° (no apsidal precession), entering and leaving the magnetosphere at roughly the same latitude. It was decided that these orbits provide insufficient coverage of the cusp region. Another was an 82.5° inclination near-polar orbit, with its line of apsides in the equatorial plane. Such orbits have their perigee near the equator, which fulfilled an initial communications requirement (later relaxed) and they also provide cusp coverage at lower altitudes. The third orbit option was also an 82.5° inclination near-polar orbit, but with its line of apsides raised to an angle of 45° relative to the equatorial plane. This orbit was proposed later on in the project, and was preferred to the other alternatives, as it was thought to provide good high latitude coverage, including the high altitude cusp, with each orbit crossing the boundaries at two well-separated latitudes.

As noted above, the orbital period and perigee were initially constrained by a communications philosophy which was later revised. It is interesting to consider the consequences of altering the perigee (while holding apogee fixed so that a change in period is driven by the change in perigee). A higher perigee would give a longer orbital period, and thus less frequent passes through the magnetosphere and less frequent boundary crossings. A higher perigee is more expensive in terms of achieving the orbit. Total time spent in the radiation belts, and hence accumulated dose and probability of spacecraft or payload system failure, can also increase with perigee altitude. There could be an upper limit on perigee height if GPS is used, as some time spent below 2 to 4 R_E is needed for good position measurements. A significantly higher

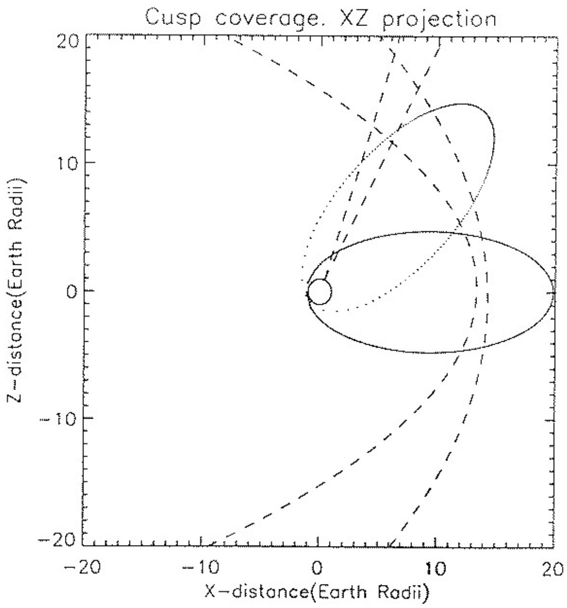

Figure 4. The figure shows the configuration of the two kinds of 82.5° near-polar orbits which were considered. The Molniya-type orbit is not shown (its plane lies perpendicular to the plane of the figure). In this figure the Sun is to the right. An approximate cusp region is indicated in the upper part of the figure.

perigee prevents measurements of certain lower altitude phenomena (although these are not emphasised in the model mission specification). On all these grounds, a raised perigee is undesirable in terms of the design goals. On the positive side, a higher perigee reduces the rate of precession, producing a constellation in which the motion of the inclined orbits relative to the equatorial orbits is reduced, and both are easier to track.

3.3 Populating the Magnetospheric Orbits

We chose to populate each orbit with a pair of satellites separated by about 25 minutes (their separation in space varies with orbital phase). The use of a pair of satellites with a small separation allows dual-spacecraft measurements including estimates of boundary motion (cf. ISEE 1 & 2 or AMPTE-IRM and AMPTE-UKS) and provides redundancy for the boundary location measurements.

The constellation populations, including the 4 solar wind monitoring spacecraft, are thus: 20 spacecraft for Scenario 1; 36 spacecraft for Scenario 2; 52 spacecraft for Scenario 3.

It was decided that the magnetospheric survey spacecraft should all approach apogee at roughly the same time. Alternative phasings were considered. For example, it is possible to imagine a constellation in which all outbound dayside spacecraft cross a model boundary at roughly the same time on a particular day. However, it is not possible to design a passive constellation to guarantee near-simultaneous boundary crossings at all relevant LTs at all epochs, as the constellation makes an annual rotation about the Earth. Even if we ignored the natural variability in boundary locations, we would have to use frequent spacecraft maneuvers.

In Scenario 1, a 2 day cycle of the constellation orbits (from apogee, through perigee and back to apogee) will typically make 32 boundary crossings. The ratio of bow shock to magnetosheath crossings varies with the rotation of the constellation relative to the Earth-Sun line, as is illustrated by Figures 1 to 3. The number of boundary crossings per orbit approaches 100 for Scenario 3.

For reasons associated with the problems of launching the spacecraft (see below) some inclined orbit spacecraft have Northern Hemisphere apogees, while others have Southern Hemisphere apogees. This may be a useful property of the Constellation, in terms of broadening latitude coverage, since the magnetic equator exhibits a seasonal motion relative to the Earth-Sun line.

3.4 Spacecraft Design

As noted above, the same spacecraft design is used in all orbital slots, as this approach minimises spacecraft production costs. The spacecraft is built around a motor and its fuel tanks, using a hexagonal structure. The model spacecraft mass is 155 kg (111 kg is fuel) and can be contained within a cylindrical volume of diameter 1.14 m and height 1m. The selected motor is the Marquardt KMHS Model 20, which has mass 1.6 kg and length 0.36 m. This is a flight qualified monopropellant (hydrazine) engine. Monopropellant engines are less complex, less massive and need only one fuel tank. It may be possible to use the same hydrazine fuel tank to feed the attitude control system. The engine is restartable, which is a useful capability for the magnetospheric survey spacecraft (allowing orbit correction) and a necessary one for the solar wind monitoring spacecraft (two burns are needed to move these spacecraft to a high altitude circular orbit).

The importance of controlling spacecraft and payload mass is clear when one considers that an increase by 1 kg in spacecraft dry mass increases the fuel required for insertion to the magnetospheric survey orbits by 2.3 kg.

We considered the following subsystems: structure, environment and protection systems, attitude control, power, communications, possible onboard position finding equipment and the onboard computer. Power is supplied by Lithium-Metal batteries charged from Gallium Arsenide solar cells. These batteries are not yet space-qualified, although they are being investigated by NASA. They have highly desirable depth of discharge performance (50% for 50 cycles) and high energy density (140 Whrs per kg). They may meet a desire (not a requirement) of operating the spacecraft and payload even during long eclipses. The choice of Gallium Arsenide solar cells is based on their greater efficiency and radiation tolerance compared to silicon solar cells. The satellites are spin stabilised, so traditional circumferential mounting of the solar panels was chosen. Some form of attitude control is considered essential in order to enable relatively straightforward interpretation of the data from the payload. Spinning, rather than three axis stabilisation has the well-known advantage of sweeping the two dimensional field of view of a particle analyzer across the entire sky, and also facilitates interpretation of magnetometer data.

Microstrip antennas were selected for the spacecraft communication system (see below). Advantages include their ability to conform to the satellite surface shape (useful when designing a satellite which can be packaged with many others in a launcher fairing), low mass, and low cost. Two omnidirectional dipole antennas are needed for handling GPS signals (see below).

The study team spacecraft was largely costed on the basis of off-the-shelf components, with 1.6 MECU set aside for design and development (1 ECU was approximately equal to $1 at the time of the study). A much more detailed analysis would be needed to investigate issues such as whether savings in production facility and manpower costs resulting from specialised sub-systems could justify such an approach.

There are likely to be advantages in investigating how spacecraft design and assembly practices can be optimised so as to achieve rapid production without loss of quality.

Scenario Number:	1	2	3
Number of Spacecraft	20	36	52
Cost per Spacecraft	0.91	0.88	0.86
Scenario Cost	18.2	31.68	44.72
Cost / Scenario 1 Cost	1	1.74	2.45

Table 1. Estimated spacecraft cost analysis, including environmental testing, broken down by Scenario (cost figures in MECU)

3.5 Payload

The satellites have only a minimal payload, consisting of a magnetometer and a plasma analyzer to measure electrons and ions (an electron capability was added by the later to the specified ion capability). A lightweight energetic particle analyzer would have been an interesting addition.

It would be difficult to remain within the specified resource constraints using traditional particle analyzers. Thus the students examined two "real-world" particle analyzer instruments designed with small size in mind. One is the SouthWest Research Institute's "MOSS" prototype miniature ion analyzer whose relatives will be used on Deep Space 1 and Rosetta. The other is the Swedish Institute of Space Physics/SouthWest Research Institute's MEDUSA miniature ion and electron analyzer (1.5 kg, 2.5 W including DPU) to be flown on ASTRID-2 and MUNIN). In addition, we studied instrument development work at MSSL. We finally proposed our own paper design, EIMA, related to MEDUSA, but tailored to our mission requirements (for example including adaptations to improve the quality of measurements of solar wind ions). EIMA would operate at all times except during radiation belt crossings. The satellite 2 second spin period would allow collection of a full three-dimensional velocity distribution with the 360° field of view analyzers every second.

A flux gate magnetometer on a short (0.75m) boom was chosen. A body-mounted magnetometer might also be feasible, and would remove the risk of problems due to failed boom deployment. It was recognised that some care is needed to assure magnetic cleanliness of the spacecraft and payload. Experience suggests that if this is planned for from the beginning, it need not be very costly. Another cost issue that emerges relates to the accuracy with which knowledge of the magnetometer (and indeed spacecraft) orientation is required. There seems to be less scope for reducing magnetometer resource requirements than for the plasma package.

A common DPU for the plasma analyzer and the magnetometer is a natural approach to save resources. The proposed telemetry return per satellite was set at 57 Megabits per 2-day orbit. This was chosen to include housekeeping data, 4 vectors per spin from the magnetometer, density and velocity vectors from each of ions and electrons at an impressive 1 second resolution, and a three dimensional distribution function for each of ions and electrons once every 10 minutes (300 spins). The distribution data are useful for checking the health of the plasma analyzer, confirming the onboard software operation and providing context to the moments data (in some situations the moments data are insufficient to distinguish between possible plasma environments). As an alternative to collecting distributions at regular 10 minute intervals, a more sophisticated instrument could use onboard event recognition software to collect short time series of distributions from intervals during an orbit (e.g. at boundary crossings). The listed parameters were deemed sufficient to meet the specified science goals (although it would probably be worth transmitting some higher order moments data at the cost of slightly less frequent distribution data). The telemetry acquisition rate per spacecraft thus averages about 360 bit/sec (after a factor of 2 compression of science data). For comparison, the nominal telemetry acquisition rate for a Cluster spacecraft is 22,000 bit/sec (and a factor of 6 larger in burst mode). The daily data return from a 52 spacecraft Scenario 3 Constellation would be 185 Mbytes/day, compared to between roughly 600 and 1000 Mbyte/day for four Cluster spacecraft.

Scenario Number:	1	2	3
Number of Spacecraft	20	36	52
Unit Production Cost	0.145	0.130	0.123
Unit Testing and Cal. Cost	0.020	0.020	0.020
Production Cost	2.9	4.7	6.4
Testing and Cal. Cost	0.4	0.72	1.04
Cost per Spacecraft	0.165	0.150	0.143

Table 2. Estimated payload cost analysis, broken down by Scenario (costs in MECU)

Estimates were made of cost savings associated with production of many instruments, although the conclusions are rather tentative. The suggested figures are shown in Table 2. A reduction in unit manufacturing costs relative to single instrument build programs, due to bulk purchase savings in component costs, seems a reasonable expectation. On the other hand, the unit costs of assembly, testing and calibration may be hard to reduce (and were assumed constant here). These are usually specialized and manpower-intensive tasks, and some aspects of the testing and calibration of such sensors require prolonged use of expensive facilities such as vacuum chambers and magnetically clean environments. Creation of a constellation would require the production of many instruments at an unprecedented rate, but in quantities which may nevertheless be insufficient to justify investment in a heavily automated production line. It may be that an investment in specialised test and calibration facilities to allow parallel processing of multiple sensors would be the best approach when creating several tens of sensors. For comparison, one could use the case of a slower production rate, building up a stockpile, however such a cost trade-off was not attempted in this study.

As with the spacecraft, it seems reasonable to suggest that an instrument which is to be produced in relatively large quantities should be designed with the production process in mind; for example, ease of assembly and sub-system exchange might be given more attention than is usually the case. A prototype would ideally be built. We estimated that about 1.2 MECU should be set aside for this activity.

3.6 Communication

Any constellation class mission has the problem of maintaining communications links between multiple spacecraft and the ground. Compared to a communications satellite constellation, the difficulty is probably greater if the constellation is to be composed of small, low power scientific spacecraft travelling to very high altitudes. In this study, the problem is compounded by having many spacecraft spread across the full range of LT, in highly eccentric orbits of different inclinations, and all approaching perigee at the same time (except the solar wind monitors).

Continuous communications would require a prohibitively large number of ground stations, so the spacecraft must use onboard storage with data dumps during short intervals of their orbit to achieve continuous data coverage.

Consider the magnetospheric survey spacecraft. These are to be small spacecraft, with as little power-generating capacity as they can manage with. Perigee transmission was the original assumption of the study because it requires the least amount of power (transmission distance is small). However, the satellite visibility interval is very short, only a few minutes for the perigee distance used here, as the satellite moves rapidly across the field of view of the ground station. Also, perigee communications would imply high loads for short intervals, separated by long gaps which is not a very efficient use of ground station resources. The most serious problem is that the wide LT spread of the orbits requires ground station coverage across the full range of longitudes, creating problems both because of the number of stations required, and the difficulty of installing them in the ocean regions (especially the Pacific).

Apogee transmission allows much longer satellite visibility periods, reducing the number of ground stations required. However, the communications segment could still be problematic. High rate communication would require spacecraft with sufficient power generation capacity (and suitable transceivers) to transmit across the large distances. Conversely, a low power spacecraft might have to transmit at a very low rate, or might have to turn the payload off to provide enough power for communication from apogee. Alternatively, the ground stations could use larger antennae, which is also expensive.

A compromise solution involves communication while at mid-to-low altitude. As the satellite visibility intervals are longer than for low altitudes, a more widely-spaced ground station network is sufficient to see all the satellites, and this approach needs fewer ground stations than for the perigee example.

The frequency chosen for transmissions was selected on technical grounds to be Ku-band. Issues of licensing were not addressed. The required contact time to transmit an orbit's accumulated data is estimated to be about 20 minutes, at a distance of about 3.5 R_E (roughly in the slot region). This does not include the time to retarget a receiver antenna when switching from one spacecraft to another. If antenna retargeting is rapid, a single ground station should have time to contact more than one spacecraft at roughly 3.5 R_E even if all spacecraft have the same orbital phase (in some cases a spacecraft can be contacted on both the inbound and outbound legs). A detailed analysis of how many ground stations would be needed to guarantee full data retrieval from the constellation, taking into account inclined and equatorial orbits and geographical constraints is necessary. However, a first cut proposal is to use four ground stations spaced roughly equidistantly around the equator, each with 2 antennae per Scenario (thus using 8 antennae in Scenario 1 and 24 in Scenario 3). Data recovery was found to be feasible with 4 m diameter dishes, which were considered to be relatively inexpensive.

If the solar wind monitoring spacecraft are able to use the same communication system as the magnetosphere satellites (as assumed) their transmission rate to a 4m dish is necessarily lower than for the other spacecraft due to their larger range. A single dedicated antenna, larger than 4 m, might be required. The high altitude and long orbital period of these spacecraft reduces tracking problems compared to the magnetospheric satellites.

The question arises as to whether it is more effective to build a dedicated communications system (in particular the ground segment) or to use existing infrastructure (service-provider systems). It is usual practice for science missions to build a dedicated system, which is then often used only for a relatively short time. The logical extreme of the service-provider approach is to utilise "satellite telephone" satellites as a relay to ground, while the science spacecraft are near perigee. This idea seems to face technical problems such as compensating for Doppler shift, and perhaps economic problems such as paying the phone bill. The possibility of using the NASA Tracking and Data Relay Satellite System was rapidly rejected on cost grounds alone.

3.7 Spacecraft Position Finding

Traditional position finding techniques are based on tracking the satellite radio signal and comparing results with predictions from an orbit model. This method has the advantage that it adds nothing to the spacecraft resources budget, but the disadvantage that the associated ground segment is rather expensive. In addition, since significant tracking durations are required for each spacecraft, ground station availability (as well as orbit analysis effort) becomes a problem when dealing with a large number of spacecraft.

Ideally a spacecraft would have autonomous position finding capability. A number of systems are in development or use at the moment (mainly for low altitude applications) although few seem applicable here in their present form. For example, MANS which relies on observations of the Sun, Moon and the Earth was successfully tested on the USAF STEP satellite in a LEO application in 1994. However in that case the system weighed 16 kg and consumed 30 Watts (future developments might have reduced those requirements). The DORIS system developed by CNES and used on SPOT-2 and TOPEX-POSEIDON (again in rather low orbits) is also interesting, but unfortunately weighs 43 kg and consumes 21 Watts. It relies on a (presently in place) ground-based network of radio beacons. Onboard position analysis is possible in real-time, and improved knowledge is achieved by ground based analysis. Deep Space 1 is experimenting with the use of observations from an onboard camera as the basis for autonomous navigation, using a 12 kg system.

Global Positioning System technology will probably give reliable data at altitudes below 2 to 4 R_E and those data, together with traditional orbit modelling, may give satisfactory knowledge of positions at higher altitude. The estimated mass and power requirements of a suitable GPS receiver were put at 5.1 kg and 13 W in our study. Some GPS testing was achieved at higher altitudes by the Equator-S mission (apogee at about 10 R_E) and a discussion of the difficulties can be reached from that mission's Internet home page.

The most cost-effective future implementation may be a GPS Translator (GPST), rather than the more familiar GPS Receiver (GPSR). The GPST has about 25% the mass and 30% the power requirement of the GPSR. The GPST acquires the data needed to infer the spacecraft position, but the position calculation is not performed onboard. Instead the signals are translated to another frequency (S-band) and then retransmitted for analysis on the ground (perhaps after an interval of onboard storage if necessary). Two omnidirectional dipole antennas are required; one to receive GPS signals, the other to retransmit them.The GPST is not presently commercially available; it is used in missile systems.

3.8 Launch Solutions

Scenario Number:	1	2	3
Number of Spacecraft	20	36	52
Scenario Cost/MECU	118.18	176.46	234.74
Unit Launch Cost	5.91	4.64	4.51
Cost / Scenario 1 Cost	1	1.5	1.99

Table 3. Launch cost analysis, broken down by Scenario.

A review of operational launch vehicles (in 1997) was performed. The cost figures for the chosen solutions are shown in Table 3 and came from the launch companies, so they are considered to be reliable costs. We did not take into account political restrictions on launcher selection which might affect "real-world" missions. A multi-small-satellite launch is more severely constrained by fairing volume (and shape) than by spacecraft mass. The fairing must also accommodate a spacecraft support/release system. Certain assumptions were made regarding the dimensions of the structure supporting the satellites. It is also assumed that there are no additional launch charges for using non-standard support structures.

In retrospect, we expect that a smaller spacecraft could be designed which would meet the mission goals. Accommodation considerations might then allow different and probably cheaper launcher selections, compared to the choices set out here, or alternatively more spacecraft could use the chosen launchers.

For the solar wind monitoring spacecraft, the most cost-effective solution is the Proton K (D1-e). Four satellites are placed in a circular geostationary orbit by the Proton upper stage, from where they have sufficient fuel to use their onboard engines to reach the chosen orbit. These satellites need restartable engines to achieve a circular orbit.

An important constraint on magnetospheric survey satellites is that separate launchers are needed for equatorial orbits and for highly inclined orbits in the same LT sector. For equatorial orbits, the most cost-effective solution is to launch up to 10 satellites on the long fairing Ariane 40-03, which places them in a 1000 km circular equatorial parking orbit. Suitable timing of satellite engine firings can send the spacecraft into orbits with their apogees in the desired LT sectors, from the shared parking orbit.

The single Ariane is cheaper than several launches on a smaller vehicle such as Taurus and also has a good reliability record. There is the possibility of launching 10 satellites where only 8 are needed, allowing 2 orbiting spares per Ariane. These satellites could be left in the parking orbit so that they can be injected later on into any LT sector where a replacement spacecraft is required. Alternatively, in the case of Scenario 3, one could save 40 MECU by using only 2 Arianes if one accepted that 4 of the 12 LT sectors (equatorial orbits) would have only one spacecraft, rather than a pair (and no orbiting spares).

A different strategy was proposed for populating the inclined orbits, as the orbital plane is different for each LT sector. Opposite LT sectors can be reached from a single inclined parking orbit if one spacecraft has a Northern Hemisphere apogee while the other has a Southern Hemisphere apogee. Such methods are capable of producing Scenario's 1 and 2, but the four additional orbits in Scenario 3 have no partners at opposite LT and a different approach is required. The injection of satellites into inclined orbits in adjacent LT sectors from a single low altitude (1000 km) parking orbit, can be achieved by launching spacecraft into a parking orbit in a plane suitable for injection to the first LT sector. Spacecraft are placed in orbits in that sector after which the parking orbit plane will then precess through an hour of LT in about 20 days. It is then possible to place the spacecraft for the second LT sector into their allotted orbit. The most economical way to populate the inclined orbits was found to be to use several small Tsyklon rockets. Each one can accommodate four satellites, enough for a pair in each of two LT sectors.

We concluded that there was no large rocket capable of launching to the inclined parking orbit. If there were such a rocket, one could in principle populate all the LT sectors after injection of multiple spacecraft into one or two inclined parking orbits, although it would take some time.

A single Ariane costs about 40 MECU, a single Tsyklon costs 8.4 MECU and a Proton costs about 60 MECU. Each scenario needs a single Proton. There is one Ariane and two Tsyklons for Scenario 1, twice that for Scenario 2 and thrice for Scenario 3.

4. Conclusion

A variety of constellation missions suited to magnetospheric science applications can be constructed, each tailored to different goals. The experience of performing this study supports the view that the first step in planning a magnetospheric constellation mission must be to choose a single, well-defined science goal. Various mission design solutions can then be examined in depth to arrive at the most cost-effective approach. The work discussed here shows the breadth of issues to be considered in the search for a cost-effective plan. While we considered several aspects of the problem, we nevertheless omitted some important ones. For example, there was only limited investigation of the use of spacecraft sub-systems which are not available off-the-shelf, and the issues associated with co-ordinating and managing such an undertaking were examined only cursorily.

Issues related to launch illustrate the interdependence of many factors affecting the search for lower costs. For example, the question of whether it is better to build redundant sub-systems into spacecraft, or to build redundancy into the constellation by launching spare spacecraft, is affected not only by launcher costs, but also by ease of access to science-specified constellation orbits. In our study, it was easy to use an on-orbit spare to rapidly replace a spacecraft in an equatorial orbit, but would require an additional launch to replace a spacecraft in an inclined orbit.

The cost figures given here are quite large, but are dominated by launch costs. A further development of our study would concentrate on the question of whether two spacecraft buses could share a single motor (the Review Panel felt that the study spacecraft could have been made significantly smaller and lighter, even with conventional technologies). If so, these could be launched as one spacecraft to the desired orbit, and then parted to form a pair at the desired separation. The result would be to roughly halve the launch costs. Alternatively, one could launch twice as many spacecraft and have mini-Clusters rather than pairs of satellites in each orbit.

The estimated costs for each scenario, including contingency and inflation estimates, were put at roughly 220 MECU for Scenario 1, 325 MECU for Scenario 2 and 435 MECU for Scenario 3. As launch costs are probably the most accurate part of the figure, we can be fairly confident that if we could halve the weight of the spacecraft, Scenario 3 could be carried out with a cost savings of the order of 100 MECU. Such a savings could take the cost into the range of an ESA Medium Class mission.

The mission cost per spacecraft launched was found to fall as constellation size increases, as expected. The estimated costs for the third scenario is roughly double that of the first, but the number of spacecraft, and the potential scientific return are certainly improved by a factor greater than two. Perhaps we should say that if a constellation mission is worth doing, it is worth doing it properly -with plenty of spacecraft!

Our study suggests that an initial investment of effort in prototyping virtually all aspects of the mission would be rewarded by a more cost-effective mission. Certainly work should be done with the payload, the spacecraft and the production and testing facilities.

The costs of testing have been recognised as a major expense by the manufacturers of communications satellite constellations. Their solution is to build many, but test only a few, and accept a few on-orbit failures. It is not clear that such an approach can be translated to scientific payloads. One would need to think very carefully about the required accuracy of measurements before foregoing calibration on an instrument by instrument basis.

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