The Kilo-Satellite Constellation Concept

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Abstract. In this paper we outline one possible implementation of a magnetospheric constellation mission. We present a mission aimed at measuring the global instantaneous structure and time variation of magnetic fields throughout critical volumes of the magnetosphere. The concept is to place a large number of simply-instrumented nanosatellites (less than 1 kg) into elliptical orbits with apogees ranging from 5 to 25 R_E. In our study, the overarching goal is to achieve the maximum number of satellites. To implement this constellation, we consider several key aspects, including: orbit evolution; communications; nanosatellite and bus design; launch and initial orbit insertion; and ground station requirements. Our analysis demonstrates that using current technologies, a mission consisting of several hundred nanosatellites is feasible and could fit within the scope of the NASA Solar-Terrestrial Probe mission concept.

1. Introduction

Measurements on the magnetospheric particles endfields have been carried out traditionally by means of single satellites. This has led to a number of difficulties in the interpretation of the data. For example, the distinction between spatial and temporal variations of the magnetospheric flow is not inherent in the data but can only be disentangled by invoking other assumptions. More generally the global pictures of the flow and magnetic field configuration are not immediately apparent in the data but depend on considerable interpretation. Recently the concept of launching a constellation of satellites has received more attention. Several concepts are discussed in this monograph. This paper addresses the question of how large a number of satellites can be placed in orbit in a constellation so that a time dependent picture of magnetic fields can be obtained. It will be shown that a constellation composed of several hundred satellites appears to be within the scope of a Solar-Terrestrial Probe line mission costs.

Admittedly, a three-dimensional picture with this many pixels still corresponds to rather low resolution. Nevertheless, the advantages of a picture over localized measurements from individual or a small group of satellites are very significant. The maxim of a picture being worth a thousand words, in spite of over use, contains considerable truth. The availability of pictures has been the cornerstone in the development of many fields of science. Most closely related to magnetospheric flow is the field of aerodynamics where the dominant tools in development of the field have been schlieren or interferometric pictures. Optical and higher resolution microscopes have been critical in the development of biology and many aspects of solid state physics. Typically each increase in microscope resolution has led to new discoveries. A comprehensive discussion of the need for constellations to advance our understanding in magnetospheric physics can be found in the Sun-Earth Connection Roadmap, 1997.

The nominal mission under consideration would place satellites in orbits with a common perigee at $1.4R_E$ and apogees ranging from 5-$25R_E$ into five planes with inclinations ranging between $\pm 15^o$. Each satellite would measure the magnetic field at 20 second intervals and would store and then telemeter data to receiving stations at perigee. At any instant, this constellation would cover a relatively small range, less than 6 hours in local time, and during the year would observe all of the critical regions of the middle magnetosphere; the tail configuration and substorm development as well as the magnetopause, magnetosheath and bow shock in the flanks and the subsolar regions.

In addition to providing a quantitative picture of events, data from such a kilo-constellation would show incoming boundary conditions clearly. When portions of the configuration extend beyond the bow shock, incident waves would be defined not only as to their magnitude but also as to their structure and plane. The resulting three dimensional development of phenomena in the magnetosphere and magnetosheath would be observed by the inner portions of the constellation. This would avoid the ambiguities resulting from the present, limited data availability. As another example, while the constellation is in the tail, external disturbances incident on the current sheet will be detectable, thus helping to resolve the question of whether or not substorms are internally or externally triggered.

The proposed launch scenario would involve placing a bus into a $1.4\ R_E$ by $5R_E$ orbit. This bus would carry the satellites and also a rocket motor, propulsion fuel, guidance and control equipment, and satellite release mechanisms. Acceleration would occur at perigee with a slow burn and as the perigee velocity is increasing, satellites would be released individually. Thus each succeeding satellite would have a slightly higher velocity and thus apogee. Since this scenario monotonically accelerates the bus through the various satellite orbits, it eliminates a requirement for propulsion on the individual satellites and also minimizes the propulsion fuel required.

The launch scenario, as described above, would place all satellites in a single plane. Since plane changes require significant amounts of propulsion fuel, placement of satellites into several planes will be accomplished by separate ground launches. As will be shown, a Pegasus XL launch, which is relatively inexpensive, can place 48 or more satellites into a series of orbits with perigees at $1.4R_E$ and apogees ranging from $5R_E$ to $25R_E$. An alternative approach might be to use a larger rocket launch and then place five buses into different planes.

The selection of a high perigee altitude increases the range over which the satellite is visible from a ground station and, therefore, reduces the number of ground stations required. Link equation calculations show that at a range of 6378 km ($1R_E$) transmission from the satellite with power equivalent to an ordinary cellular telephone would provide accurate communication to a moderately priced ten-meter receiving dish.

In order to achieve these objectives the mass of the individual satellites must be kept to an absolute minimum. This in turn requires that only the absolutely necessary functions be retained and that they be implemented in the simplest possible fashion. For the present analysis the only measurement that will be made is the 3-axis magnetic field and, of course, the required satellite orientation. There will be no active station keeping or attitude control systems utilized.

Reliability requirements on individual satellites can be greatly reduced as compared to conventional satellites. With a large number of

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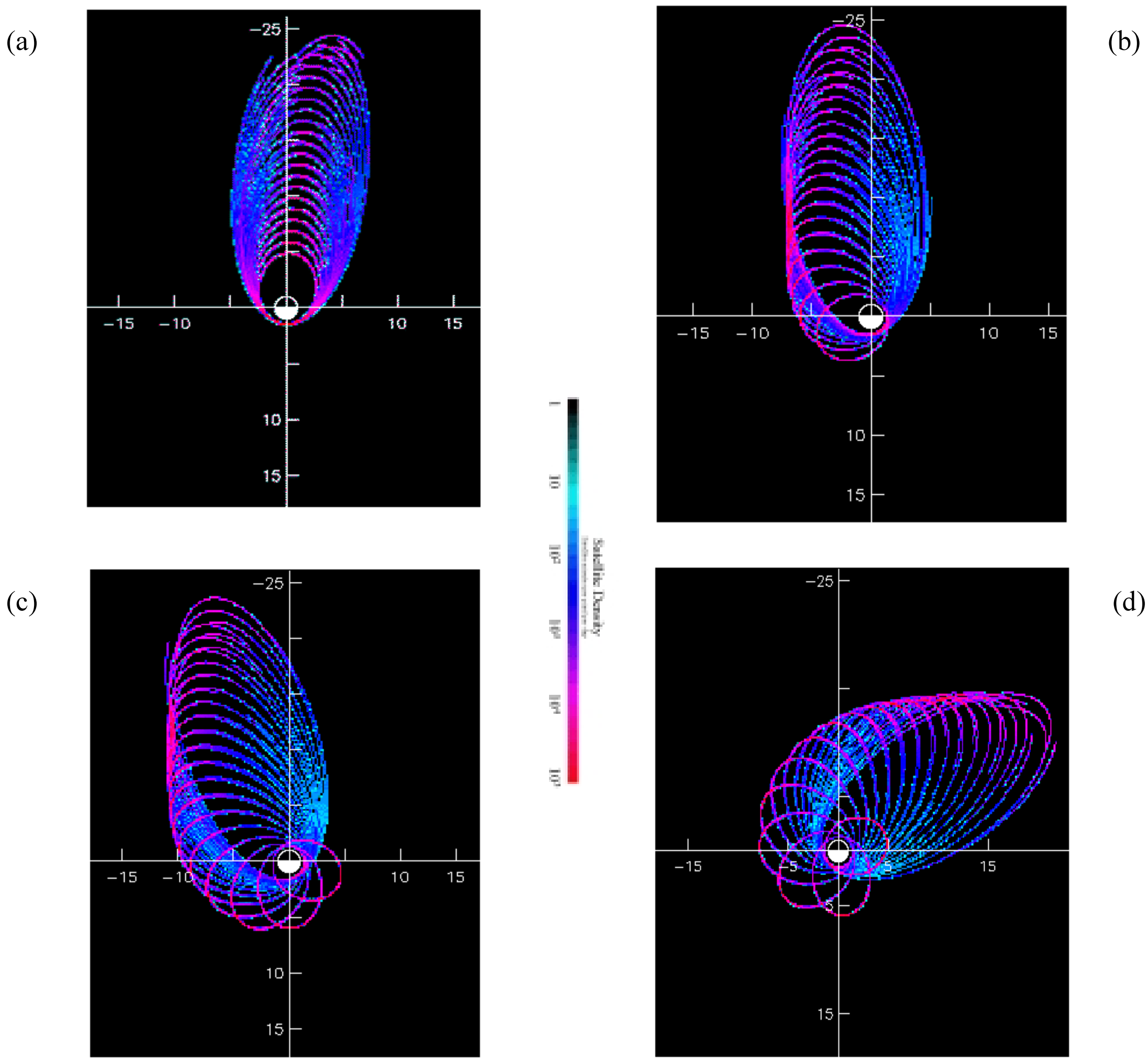

Figure 1. Constellation drift over a three year period. Density or frequency of occurrence in a specified area in arbitrary units a) at launch and b), c) and d) after one, two and three years. Shows that high spatial resolution can be maintained for several years.

satellites, individual failures correspond only to loss of the data from those satellites, not to an overall system failure. Designing to 99% reliability would correspond to loss of only 1% of the data. The associated reduction in redundancy can correspond to significant mass reduction.

In the following sections, we discuss several critical aspects of implementing a constellation of this type. These elements are summarized below. Section II will discuss drifts and precession of the satellite orbits, showing that the configuration retains reasonable coherence over several years. Radiation exposure on the selected orbits is also calculated. It becomes as high as 1Mrad/yr for some of the satellites. As discussed in later sections, this is taken into account in selection of satellite components and passive shielding.

Section III will discuss the required data rates and the resulting requirements on the communication link, in particular: transmitter antenna requirements, transmission power and receiver properties.

Section IV discusses the design of the satellite and its critical components as well as the status of a breadboard of the electronics which is presently under construction within the Boston University Center for Space Physics. This includes estimates of the power and mass requirements.

Section V discusses the launch scenario using a Pegasus XL launch vehicle. The analysis includes the propulsion requirements of the bus, the packaging of the satellites in the bus and some aspects of the satellite release mechanisms.

Section VI discusses some aspects of the ground station requirements such as number and distribution of stations, satellite acquisition and tracking.

2. Precession and Configuration Drift

One of the important considerations about this constellation configuration is how long will it stay together when the orbits precess due to the departures from spherical symmetry of the Earth's gravitational field and the effects of the lunar and solar fields. To check this we have calculated orbits using the Merged Simplified General Perturbations Propagator (MSGP4), in Satellite Tool Kit, version 4.03, produced by Analytical Graphics Inc. This software includes moments of the Earth's gravitational field up to the fourth geopotential coefficient, J_4, and includes lunar and solar gravitational effects.

Twenty-one satellites were started at the same time and in the same plane with a perigee of 1.4 R_E and apogees at $1R_E$ intervals from 5 to 25 R_E. Orbital periods range from about 0.3 to 2.8 days. The major axis of the initial orbits was chosen along the Earth-Sun line with apogee on the night side. The arbitrarily chosen launch date was January 1, 1999, and results are insensitive to the actual launch date. Although several planes near the equatorial plane are expected in the actual configuration only the orbital plane at 10 degrees to the equatorial plane was calculated in detail. Others planes with different inclinations will be quantitatively similar. Figures 1a, b, c and d show the satellite orbits at annual intervals. The plots are in inertial coordinates and are shown as a projection onto the equatorial plane. The color-coding indicates an effective density of satellites. This density is obtained by observing the number of satellites in an area of $0.1R_E$ by $0.1R_E$ every minute. The sum of these numbers over a three-day period is defined as the density at that point. The three-day average allows complete orbits even for the high apogee orbits. However, some aliasing is apparent in the initial picture and persists in the later ones.

In addition to the high satellite density due to a common initial perigee for all satellites, all of the orbits show high densities near apogee because the satellites are moving most slowly there. During the first year, relatively little precession occurs and the dominant motions relative to the magnetosphere are that the orbits are fixed in inertial space. This allows observation of the tail, flanks and subsolar region during the course of the first year. Precession is a stronger effect on the low apogee orbits since they spend more time in the near-Earth distorted gravitational field. At the end of three years the low apogee orbits have precessed about 360 degrees ahead of the high apogee ones. It is significant, however, that even at the end of three years, particularly the high apogee orbits, a high-density configuration remains allowing high-resolution coverage.

Using the same orbital parameters and the CRESRAD 94 program, radiation doses per year were calculated for extremely active conditions. The lowest apogee orbit remains in the radiation (both inner proton and outer electron) belts for the largest fraction of the orbital period and therefore receives the maximum dosages of 1 Mrad/yr (in SiO_2) behind an assumed equivalent thickness of 30 mils of aluminum. The dosage for higher apogees decreases roughly as the reciprocal of the period. Therefore orbits beyond a $9R_E$ apogee will receive less than 500krad/yr (in SiO_2). The electronic components selected are all radiation hard to 1Mrad and 30 mils aluminum equivalent passive shielding is included in the design.

3. Communication Requirements

Initially our mission concept had considered laser communication by means of a modulated retro-reflector on the satellite. This system turned out to be barely feasible and would certainly have required significant development of both ground and satellite components. By contrast, as discussed below, an RF system can be developed based on standard components.

The rate at which data must be sent to the ground is critical in determining the required parameters of both the transmitter on the satellite and the ground receiving station. In order to minimize RF power requirements the satellites will store data taken over most of the orbit and download it at perigee.

3.1 Required Data Transmission Rate

Data points taken every 20 seconds should insure a sufficient data rate to capture phenomena of interest to the development of macroscopic magnetospheric phenomena. The primary data to be obtained is a set of three-axis magnetometer readings. Each field component should be measured to the larger of ±0.5nT resolution or 1%. In principle, each measurement can be stored in a 12-bit floating point number (1 sign bit, 4 exponent bits, and 7 mantissa bits). This requires 36 bits per 3-axis data point. Additionally 2-axis spacecraft attitude information is needed but the data transmission requirement for this can be reduced by calculation based on periodically updated measured spin properties. Allowing for housekeeping information and other possible data requirements, 128 bits of data transmission per data point has been allowed for.

The orbital periods for the furthest members of the constellation are slightly less than three days. Assuming one 128 bit data point every 20 seconds along the orbit, the satellites must carry approximately 1.6 Mbits of data storage. A conservative memory requirement for each satellite is then 4 Mbits, which, in the worst case, allows for the possibility of storing data for two orbital periods. It should be noted that several safety margins have been included to arrive at the 4 Mbit memory specification, and the true safety margin as compared to transmitting only the 3 axis magnetometer data is more than a factor of eight even for high apogee orbits.

One minute is a convenient data download time. The satellite only moves about $0.1R_E$ in that time and will stay easily in view of a receiving station. Additionally a large constellation requires the ground station to receive many separate data streams. If a one minute transmission from each of 500 satellites is read every orbit, the receiver duty cycle would be about 30%. As will be discussed in Section 6, once a satellite orbit is known, allowances for acquisition time do not add appreciably to this. Also overlapping transmissions will be significantly reduced by having each satellite transmission repeated for ten minutes while the satellite is near perigee. Nevertheless a duty cycle of 30% or less seems advisable.

This requires a data transmission rate of ~70Kb/sec.

3.2 Link Equation

The following assumptions have been used in calculating the transmission power needed on the satellite:

1. A Scientific Atlanta 11.3m receiving dish as the ground station receiver having a gain over temperature (G/T) specification of 25.35 dB.
2. An S band transmission frequency.
3. A maximum range 6378km ($1R_E$) at which accurate transmission must be achieved.
4. Bi-Phase shift keying (BPSK) modulation at the 70 kbps data rate.
5. A signal to noise ratio of 10.8 dB which gives a bit error rate of $<10^{-6}$.
6. A dipole transmitting antenna on the satellite with the dipole axis perpendicular to the plane of the orbit.
7. Losses of 6dB.

8. A margin of 7dB.
9. Rate one half convolutional "Viterbi" encoding of the data stream.

These assumptions require a transmitter power of 50 mW. Commercially available cell phones radiate comparable or larger powers. For example the Qualcomm Q™ transmits a maximum power of 200mW. The total weight of the entire cell phone including the LCD, keys, case, microphone, speaker, vibrator, batteries sufficient for 1.5 hours talk time, etc. is 162 grams. Eliminating the components that are not needed, it will be assumed that about 40 grams of RF circuitry would be sufficient for the satellite application. Required redesign has not been considered yet. One of the open questions is the need for radiation hardness. As discussed below, all of the components in the main board are available in versions that are radiation hardened to 1 Mrad. The parts in the RF system are all bipolar. Since these are much less sensitive to radiation we do not anticipate a major difficulty.

The assumption (3) of a maximum range of $1R_E$ and the earlier assumption of a perigee of $1.4R_E$ (2640km altitude) are based on keeping the number of ground stations required for continuous data retrieval from the constellation at a reasonably low level. They were derived on the idealized geometric picture that if data could be retrieved along a horizontal line of sight these numbers would allow a receiving station to receive from satellites whose position is within 45 degrees of the ground station, corresponding to a requirement of only four ground stations around the globe. More realistically, limitations on possible ground station locations and the fact that at least 5 or 10 degrees from the horizon must be allowed for the line of sight it is likely that closer to ten ground stations will be required.

The assumption (5) that the transmitting antenna is a dipole oriented roughly perpendicular to the plane of the orbit gives a roughly 2 dB gain over an omni-directional antenna but does require satellite orientation. This will be accomplished by releasing satellites with a spin and including a damping mechanism so that rotation will be maintained around the axis with the largest moment of inertia. Thus no active mechanisms are required.

4. Satellite Design

The general satellite configuration is shown in Figure 2. The outer cylindrical surface consists of solar cells. The transmitting antenna is along the axis of the cylinder. Satellite spin will maintain the orientation of both the antenna and the solar cells within 10 or 20 degrees. The satellite electronics including batteries will be concentrated in the center. The actual magnetometer location is dependent upon being able to minimize spacecraft magnetic fields. It is expected that it can be located within the cylindrical volume. The sun sensor looks radially outward and will determine the phase of the rotation. Additionally in conjunction with the magnetometer it will determine the direction of the spin axis.

As compared to typical satellite designs this mission is particularly stringent in terms of requiring low mass, low power and a relatively high degree of radiation hardness. Additionally, in view of the large numbers of satellites involved, the design must address manufacturability–simplicity of fabrication, assembly and calibration. On the other hand, the large number of satellites also reduces the reliability requirements. Failure of a few satellites simply reduces the number of data points but it does not lead to mission failure.

Figure 3 shows a general schematic of the electronics. The separate sections will be discussed below except for the RF circuitry, which has already been discussed in Section 3.

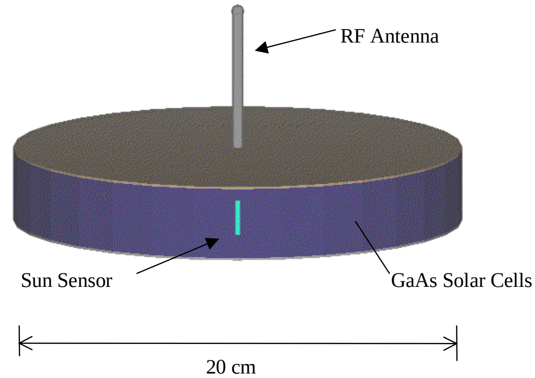

Figure 2. External view of satellite configuration. Circuitry will be near the center and magnetometer will be internal if adequate magnetic cleanliness can be achieved.

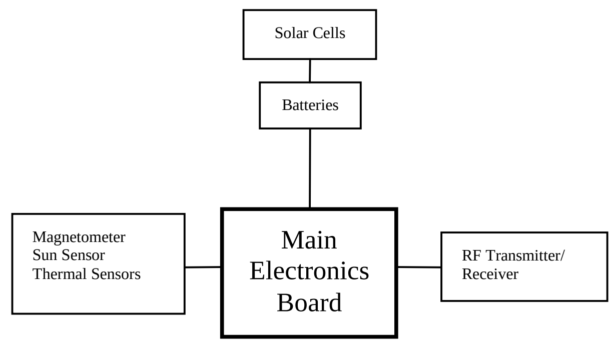

Figure 3. Overall block diagram of circuits.

4.1 Power Supply

The power supply is being designed for one Watt average power from solar cells. Peak power requirements up to five watts largely for data transmission will be available by using battery storage. Both of these numbers somewhat exceed the present estimates of power requirements and allow for some of the inherent inefficiencies due to voltage regulation. Additionally some of the higher apogee satellites may be eclipsed for about four hours and battery storage must be sufficient to maintain operation during that time.

The overall size of the satellite is determined predominantly by the solar cells and the average power requirement. GaAs solar cells have an efficiency of 18.5%. Allowing for a 10% degradation in power output due to 1Mrad of radiation damage, a cylindrical radius of 10cm and a height of 3cm will provide over one watt of average power. The radiation degradation is consistent with 12 mils of glass over the solar cell that brings the mass of the shielded cells to 40 g (Ray et al, 1993).

Ten minutes of data transmission at five watts requires approximately one watt hour of energy storage. Operation through eclipse for four hours at one watt is more stringent and requires four watt hours of energy storage. This can be accomplished with the equivalent of five AA, rechargeable NiCD batteries in series. The basic electronics is designed to operate at +5 volts with a switched capacitor power inverter to provide a small amount of power at –5 volts.

4.2 Main Electronics Board

The main board interfaces with the power supply, the sensors and the RF communications. A breadboard version has been designed and is presently under construction at the Boston University Center for Space Physics. Only components that exist in a version that is radhard to 1Mrad are being used. However, the breadboard is being constructed from commercial equivalents. The main board will be enclosed in a 30 mil aluminum box.

The breadboard employs an 8086 processor, 1Mb RAM and 64Kb ROM. An ACTEL FPGA handles memory access, error correction, data acquisition, spacecraft time, performs watchdog functions and interfaces with the RF transmitter/receiver. For flight the ACTEL A1280XL will be used. To facilitate prototyping two A1020s are being used instead.

The main board also performs analog signal conditioning for the payload magnetometer, temperature sensors and sun sensor. All of the analog data will be read by a 12-bit Analog-to-Digital converter.

In an effort to minimize power consumption the analog circuitry will be powered off about 90% of the time on a rapid cycling basis so that it will not affect data collection. Additionally the static processor and memory will be halted between data samples.

4.3 Magnetometer and Attitude Sensor

A 3-axis magnetometer will be accurately aligned relative to the spacecraft axes. The satellite will have been released with spin principally around the axis of the cylinder that is designed to have the largest moment of inertia. A damping mechanism will be included so that the spin will decay to rotation only around this axis. In order to determine the fields in absolute coordinates both the two angles defining the direction of the spin axis and the phase of the rotation must be known.

The direction of the spin axis can be determined by using the magnetometer and a sun sensor in combination and noting that the spin axis should remain fixed over an orbit. When the satellite is within $\sim 2R_E$ of the earth the magnetic field is accurately known and only minimally variable. The magnetic field along the axis should remain constant as the satellite spins. The ratio of that field and the known magnitude of earth's field is the cosine of the angle between the spin axis and the field axis. The other two components will vary sinusoidally with the phase of the rotation angle. The maximum occurs in the plane defined by the magnetic field and spin axis directions. Using a sun sensor the phase of the rotation relative to the satellite-sun line can be determined and therefore the absolute direction of the satellite spin axis.

While it is reasonable to extrapolate that the spin axis remains fixed as the satellite moves around its orbit, it is not possible to know the spin rate with sufficient accuracy to determine the phase of the rotation at the time that a measurement is taken. Therefore, the sun sensor will be used while the magnetometer measurements are being made to determine the rotation phase angle.

The sun sensor will look radially outward from the satellite. It will consist of a slit and a sharply defined photodiode. We expect that the angular accuracies of the magnetic field measurements will be better than two degrees.

A flux-gate magnetometer will be utilized. A specific design has not been selected. However, Applied Physics Systems 553 has adequate sensitivity, weighs 20 grams and uses 200mW of power. Radiation hardness is not known but the fundamental flux-gate mechanism is insensitive to radiation. This system is strongly temperature sensitive, $\sim$1nT/^{o}C. A temperature sensor and thermal compensation must be included. The circuitry is being designed for 0.5nT or 1% accuracy whichever is less stringent.

Historically, magnetic field sensitivity is attained by sufficient magnetic cleanliness on the spacecraft. While the fact that very little power is used on the spacecraft should help in this regard, the small dimensions make it more difficult. We hope to be able to avoid having the magnetometer on a long boom. Minimizing magnetic fields depends upon keeping paired conductors close together and stringent care in eliminating current loops. We anticipate considerable testing and adjustment of the mechanical properties of the electronics before being able to define a configuration and a magnetometer location.

4.4 Design Summary

Thus far neither satellite structure nor thermal balance have been considered in any detail. The mass of the structure will be estimated as comparable to mass of the components. Some of the satellites can be eclipsed for up to four hours. We have included sufficient batteries to maintain electrical operation during this time. Thermal shielding will be required to maintain operating temperatures during this time.

The major power consuming component on the main board is the 12-bit ADC which in continuous operation requires 595mW. As mentioned above we plan to run this on a 10% duty cycle so that the power consumption will be cut to 60mW. Similarly some of the other components can be run on 10 or 20% duty cycles while others such as the clock must run continuously. Taking these factors into account the average power consumption on the main board will be 340mW. Based on the Applied Physics Systems magnetometer power consumption for continuous operation would be about 300mW. Average power consumption by the RF system is negligible since it is used for such a small fraction of the time. Thus the total average power consumption estimate is 640mW as compared to the 1W generated by the solar cells.

The mass of the components in the breadboard version of the main board have been weighed at 140g and the aluminum enclosure adds 60g. The shielded solar cells weigh 40g and the magnetometer adds 20g. The five AA equivalent batteries weigh 125g. The estimated weight of the RF system suggested above is 40g. Thus the total component mass is 325g. With an allowance for structure somewhat smaller than the component mass the total mass is under the 1kg target.

5. Satellite Launch and Release

Several vehicles were considered for launching the constellation into the final orbit. The best suited candidate was determined to be the Pegasus XL from Orbital Sciences Inc. Because of its relatively low cost, each plane could be launched on its own vehicle. If a larger launch vehicle were used, several plane changes would be required. The large fuel requirements for these burns could significantly reduce any gains in launch performance.

5.1 Launch Scenario

The launch sequence is made up of four steps that will end with the insertion of the satellites into a range of orbits. The first step is to use the Pegasus XL to launch the bus into a 200 km altitude circular parking orbit. At this time the bus will separate from the Pegasus and continue to the final orbit. Step two consists of a STAR 27E kick motor to raise apogee to 5 R_E. After the completion of this burn, the STAR kick motor would be jettisoned and a hydrazine rocket would perform the remaining burns. The third step is a hydrazine burn that raises perigee to 1.4 R_E. The final burn occurs during the release of the satellites. A more detailed description is outlined below.

Using the Pegasus XL it is possible to lift 465 kg into a 200 km altitude parking orbit (Orbital Sciences, 1997). At this point the bus would separate from the Pegasus and continue to the final orbit. The separation mechanism standard on the Pegasus has a mass of 12kg allowing the bus to have a mass of 453 kg. The change in velocity required to raise perigee from a radius of 6578 km (200 km altitude) to 5 R_E is then calculated as 2240 m/s. Using the rocket equation we can calculate the ratio of propellant to satellite mass that is require to complete this burn. The effective specific impulse, I_{SP}, of the STAR 27E motor is given as 287.4 s. The required propellant mass is 248 kg leaving 204 kg injected into this orbit. The STAR 27E is recommended by Orbital Sciences Inc. for use with the Pegasus XL for acceleration to GTO and is the smallest STAR motor that will accommodate the amount of fuel required. If we allow 25 kg for the motor casing and 15 kg for the mounting and release mechanism we have 165 kg remaining.

Again using orbital mechanics the required change in velocity to raise perigee from a radius of 6578 km (200 km altitude) to 9020 km (~1.4 R_E) is 280 m/s applied at apogee. Using the Kaiser-Marquardt model 20 hydrazine motor, the specific impulse, I_{SP}, is 235 s. Again using the rocket equation we find that we need 20 kg of propellant to complete this burn.

The final burn requires a raise of apogee from a radius of 5 R_E to 25 R_E while releasing satellites. Using the same method outlined above, the change in velocity at perigee is determined to be 846.2 m/s. Using the conservative assumption that all of the satellites would be accelerated to 25 R_E apogee, 45 kg of propellant would be required. In fact, release of the satellites during orbit raise would require somewhat less fuel due to the constantly decreasing mass to be accelerated.

In order to accommodate the required hydrazine, we have selected tank #80364-1 from Pressure Systems Inc. The capacity of this tank is 68 kg which easily accommodates the 64 kg of hydrazine required for both burns. Also under consideration is the Kaiser-Marquardt model 20 hydrazine motor which has a mass of 1.6 kg and a nominal thrust of 455 N. This would allow the first burn at apogee to occur in less than two minutes and be effectively instantaneous. The perigee burn would require about five minutes corresponding to about one release every six seconds for 48 satellites. Due to motion of the bus this burn would occur over $\pm$1500 km at perigee. This range is small enough that the orbital calculations assuming the burn occurred at perigee should be sufficiently accurate for the present estimates.

After subtracting all of the separation systems, motors and tanks we are left with 93 kg for the bus and satellites. If we allow approximately 45 kg for the bus, then we can accommodate 48 satellites at 1 kg each. If it is found that the bus or satellite mass can be further reduced, additional satellites could be flown. This mass breakdown is shown in Figure 4.

5.2 Packaging and Release System

Preliminary design concepts for the packaging of the satellites and their release from the bus have been developed. One promising approach follows. The satellites will be carried in the bus in six stacks of eight satellites each as shown in Figure 5. These stacks will extend radially outward from a hexagonal center core. Allowing a 3 cm height plus a 1cm gap between satellites, the length of a spoke of 16 satellites plus the center core will be 99 cm, leaving an adequate margin within the 116.8cm internal diameter at the base of the Pegasus XL payload compartment. With 20cm diameter satellites this stack would require less than 30cm space along the bus axis. The other components carried on the bus that might require significant length in the payload are the thruster (40 cm), hydrazine fuel tank (20 cm) and STAR kick motor (90 cm). Taking these measurements into account, there should be no problem fitting into the payload compartment that is 214 cm long. The individual satellites will be released from the outside of the stack and the inner satellites will be delivered mechanically to the outside release position.

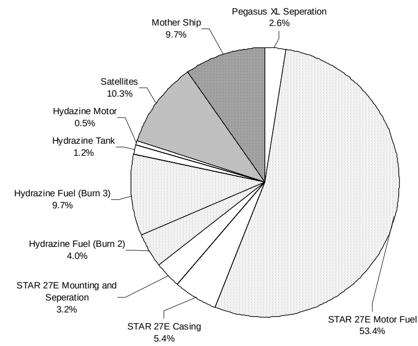

Figure 4. The launch mass budget. The fraction of the mass that is launched into a 200km circular orbit that is applied to each stage of the deployment process. Based on a Pegasus XL launch which places 465 kg into the low Earth orbit.

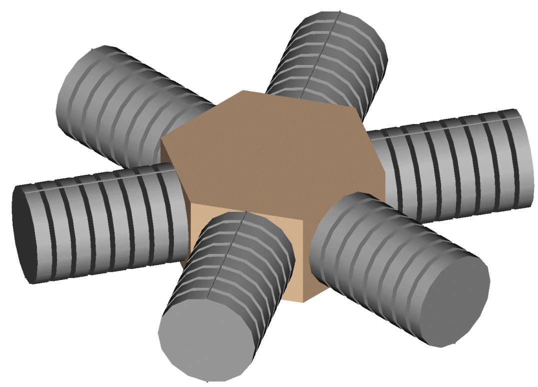

Figure 5. Satellite packaging arramgement in Pegasus XL payload compartment. Satellites will be moved mechanically to the outside position prior to release.

One of the constraints on the release system is that while the bus is likely to be spin stabilized around its velocity vector, the satellite must spin about an axis which is roughly perpendicular to the plane of the orbit. This requirement must only be met to $\pm 30^o$ since both the solar cells and the RF transmission depend only on the cosine of the angle. The above satellite storage arrangement was chosen so that the outside satellite in the stack can be released at the proper bus orientation – when the satellite axis is perpendicular to the orbit. As each satellite is released it will be given a separation velocity and a spin.

The satellite spin immediately after release will be complicated, consisting of a combination of the bus rotation rate and the rotation added on release. We plan to include a damping mechanism on the satellite so that its final rotation will be only around a single axis, the cylindrical axis of the satellite that will be designed to have the largest moment of inertia. In order to have the final spin axis close to perpendicular to the orbit plane, the release spin rate must be several times larger than the bus spin rate.

Specific mechanisms for holding the satellites, ratcheting them outward, providing the release velocity and spin have not yet been determined. Various combinations of springs, motors, gas jets and miniature explosive bolts are under consideration.

6. Ground station requirements

As indicated above we anticipate about ten low latitude ground stations distributed as evenly as possible around the Earth. Each ground station would be equipped with a Scientific Atlanta 11.3 m dish, data providing updated information on coordinates and expected arrival times of each of the satellites, and data handling capabilities. The angular spread corresponding to the receiving dish size is about 0.6° and at the minimum distance from the satellite (2640 km) the spot size will be about 30 km. The satellites will utilize their magnetometers to turn on transmission at a specified magnetic field (corresponding to altitude) and begin transmission. The triggering magnetic field strength will be selected to allow for a ten minute transmission period symmetrically displaced around perigee.

Initial orbital coordinates will be known from the known position and velocity of the bus at the time of release of the satellite. We anticipate that since the release location and the location at transmission are both close to perigee, uncertainties in release position and velocity should have little effect on azimuth and elevation of the satellite near perigee. However, uncertainties in the magnitude of the satellite velocity can cause significant errors in the orbital period and therefore the time of satellite return. For the higher apogee orbits a 1m/s error in perigee velocity will cause a ten minute variation in period. On the initial orbits, acquisition will, therefore, require that the receiver wait for satellite arrival. For later orbits of the same satellite the period will have been measured and the anticipated arrival time will be known.

Details of the logistics of this acquisition process have not been worked out. However, each satellite will send an identifying signal during transmission. The satellite will be initially programmed to transmit its identifier continuously until it receives an instruction to proceed normally. This will also be used to return to identifier transmission if contact is lost or to shut itself off completely under appropriate conditions.

The receiving dish slew rate is in excess of 10°/s whereas the satellite will be moving overhead at less than 0.2°/s so that tracking should not be a problem. This also means that the dish should have sufficient agility to move from one satellite to another in times that are typically fractions of a minute. Additionally angular accelerations of the dish in excess of 15°/s^2 are possible in both angles. This would allow oscillating the beam at frequencies in the 1Hz range in order to detect more precisely the position of the satellite within the receiver spot size thus providing information for updating orbital parameters. If necessary this could also be used to scan for acquisition of the satellite.

7. Summary

A constellation mission capable of deploying hundreds of magnetometry nanosatellites in the magnetosphere has been studied. We find that such a mission appears to be feasible within the scope of the new NASA Solar-Terrestrial Probe line. While the challenges of implementing a constellation are largely new ones (e.g., developing queuing algorithms and data analysis tools for hundreds of satellite data streams), we are confident from our study that constellations are possible, even with today's technologies. The scientific values derived from these missions are great and therefore these concepts should continue to be pursued vigorously.

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