Application of the Multiple Satellite Concept to Particles and Fields Research

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The concept of a three-dimensional laboratory consisting of a cluster of four satellites operating as a synergistic unit in the near-earth environment is analyzed for its applicability to investigations of solar-terrestrial relationships. Particular emphasis is placed on experiments that are capable of resolving the temporal and spatial interactions of the solar wind and the earth's magnetosphere. Mission requirements are affected by scale factors and time variations of the particles and fields under consideration as well as the characteristics of the appropriate scientific sensors. Primary system constraints are referenced to orbit choice, array configuration, and communications factors. The systems study reveals that it would be feasible, utilizing proven technology, to establish a four-satellite cluster in an acceptable orbital array using a single Thor-Delta series launch vehicle.

Introduction

SCIENTIFIC spacecraft missions during the past decade have accomplished 1) confirmation of the existence and delineation of some of the characteristics of the solar wind, 2) mapping of the earth's radiation belts and the magnetosphere, and 3) acquisition of an immense volume of data from which it has been possible to infer the nature of interactions by the earth and planets with the particle, field, and radiation emissions of the sun.

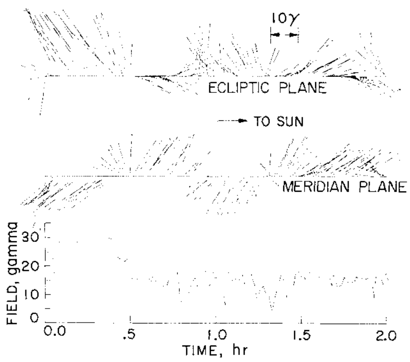

Fig. 1 "Disoriented" *H* Vectors. The interplanetary magnetic field is represented by projection of the vector field upon the ecliptic plane and the meridian plane, which is defined by the normal to the ecliptic and the spacecraft-sun line. The lower graph depicts the total field from samples taken by Mariner II at intervals of 37 sec, representing 2 hr during disturbed conditions following the interplanetary shock of Oct. 7, 1962 (after Colburn and Sonett,[1] 1966).

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However, a single spacecraft represents a moving, single-point "sensor" that travels with significant velocity through the interplanetary laboratory whose phenomena have considerable spatial extent. Consequently, certain measurement ambiguities result from 1) motion and spatial extent of the phenomena of interest, 2) time variations of the phenomena under investigation, and 3) motion of the sensor itself. Indications of the time-varying environmental characteristics of the earth-sun system are shown in Fig. 1, which illustrates the vector direction and magnitude of the magnetic field measured along the flight path of the Mariner II spacecraft,[1] and in Fig. 2, which illustrates the solar wind fluctuations as measured by Pioneer VII.[2] In each case the spacecraft velocity is small compared with characteristic velocities of the

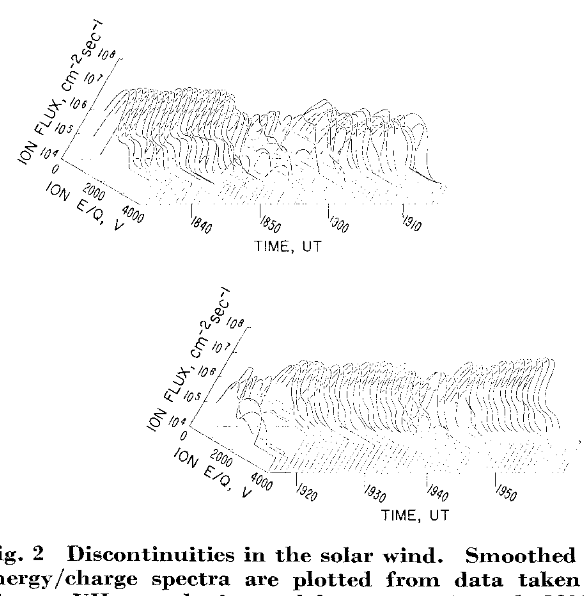

Fig. 2 Discontinuities in the solar wind. Smoothed ion energy/charge spectra are plotted from data taken by Pioneer VII over the interval from approximately 1830 to 2000 UT on September 25, 1966. Note the particularly disturbed periods from 1850 to 1920 UT and from 1940 to 1950 UT with relatively quiet conditions evident before 1850 and after 1950 UT (after Wolfe et al.,[2] 1967).

phenomena, which are propagating radially outward from the sun at approximately 400 km/sec.[3-6] What is not shown in Figs. 1 and 2 is the fine structure enmeshed in the variations, nor is their geometrical extent or time variation directly decipherable. Those few ambiguities that have been resolved have been arrived at by statistical or inductive reasoning, plus a few rather serendipitous incidents when other spacecraft with complementary instrumentation were in advantageous relative positions at the proper time.

A synergistic coorbiting multiple satellite array, with controlled and predictable separation distances, can resolve ambiguities of the type mentioned previously. A three-dimensional array providing separation distances along three orthogonal vectors would allow complete resolution of both temporal and spatial variations in the space environment along the path of the satellite array. Mathematically, a three-dimensional array, if properly instrumented, could allow full determination of the instantaneous divergence of any vector physical quantity at points along the path of the satellite system. The simplest three-dimensional noncoplanar array would involve four satellites in a tetrahedron for complete vector determination of time-position-dependent phenomena; arrangement of the array could follow other equivalent orientations such as separated crossed dumbbells or unit steps along each reference vector (see Fig. 3).

To fully resolve any instantaneous vector phenomenon, the vector satellite separation distances must be accurately known, and an accurate common time base must be maintained between the individual satellites. The proper separation distance magnitudes are related to the satellite instrumentation characteristics and the scale factors of the particular phenomena under investigation.

Scientific Applicability

Although other scientific applications such as remote sensing of earth resources or mapping the ionosphere for communications purposes might be equally appropriate for a synergistic multiple satellite system, a three-dimensional laboratory is particularly attractive for research on solar-terrestrial relationships being conducted in the near-earth environment, especially the interactions of the solar wind and the magnetosphere where the latter is very responsive to solar storms and changes in the solar atmosphere.

Early satellite and space probe investigations confirmed many general features of the interaction of the earth's magnetic field with the solar wind that were theoretically predicted as early as 1931 by Chapman and Ferraro.[7] It now seems fairly well established that the magnetosphere is a permanent feature of the earth, that its boundary on the sun side occurs at approximately ten earth radii (R_e), and that it is preceded by the magnetosheath whose boundary on the sun side is the detached bow shock wave.

In the classical sense, the earth's magnetic field can be represented by an earth-centered dipole tilted 11° from the axis of rotation. Such a dipole field would theoretically extend to infinity with decreasing strength; however, in the case of the earth, its magnetic force lines are confined within an asymmetric cavity called the magnetosphere, the boundary of which is termed the magnetopause. On the sunward side, a shock layer is formed where the solar wind contacts and compresses the magnetosphere; on the antisolar side, the magnetic cavity tails off directly away from the sun. The shape of the magnetosphere is thus molded by the solar wind, and both its size and shape are directly influenced by solar activity as manifested in the solar wind intensity.[3,8]

This unsymmetrical geomagnetic dipole field acts like a magnetic bottle to divert all but the most energetic cosmic rays on the outside, yet confining the radiation belts within. Transition from the contained field to the weak spiral magnetic field pulled out of the sun by the solar wind is marked by a region several earth radii in thickness surrounding

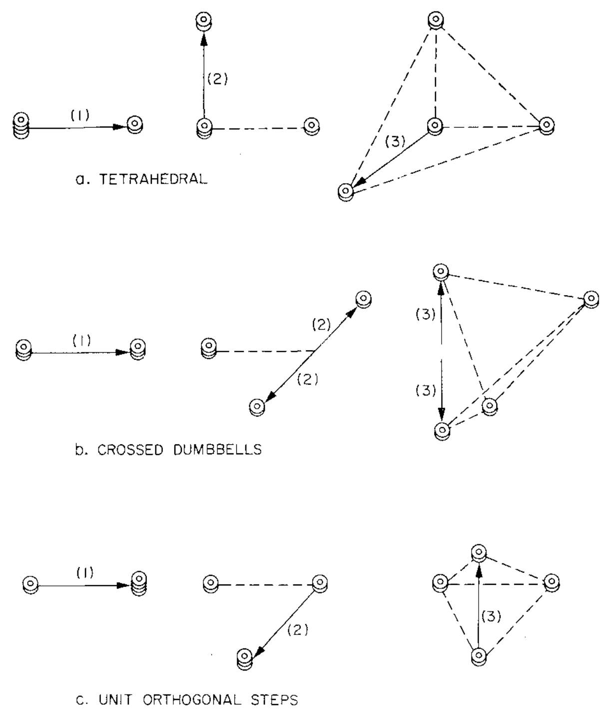

Fig. 3 Prime noncoplanar satellite arrays. Numbered steps indicate the sequence of deployment from an initial stacked configuration.

the magnetosphere and characterized by irregular, rapidly fluctuating magnetic fields and disordered plasma turbulence. Magnetic transients as high as 100 gamma have been recorded in this transition region (magnetosheath) compared to the relative calm of the interplanetary magnetic field which has a strength of only 5 to 10 gamma.[1]

The solar-terrestrial interaction presents a continuously changing situation. During the minimum of the 11-yr sunspot cycle, for instance, the solar wind decreases in intensity and allows the magnetosphere and its attendant bow shock wave to expand outward in the direction of the sun, while during sunspot maximum, the reverse is the case. This expansion and compression due to varying plasma intensity often has a magnitude of 4 R_e between peaks of solar activity.[9] But the sun does more than distort the magnetosphere with the solar wind. Immense tongues of plasma are sometimes unleashed by solar storms and these completely envelop the earth.[3-6] A magnetic field embedded in the plasma tongue superimposes itself upon that of the earth and creates additional confusion. Interplanetary "weather" outside the shelter of the magnetosphere is dominated entirely by these solar eruptions as is, to a lesser extent, the state of the earth's radiation zones, ionosphere, and upper atmosphere.

Significant phenomena occurring in this environment which are amenable to a three-dimensional investigative approach such as that represented by the multiple satellite concept are: 1) Alfven waves, 2) magnetohydrodynamic waves, 3) ion-cyclotron waves, 4) shock waves, 5) contact discontinuities, 6) interplanetary discontinuities (nonrecurrent disturbances in the solar wind), 7) turbulent eddies, 8) anomalous transport phenomena, 9) electric and magnetic field variations, 10) collisionless dissipation, and 11) boundary structure and stability. Particle physics processes also amenable to this type of treatment are: 1) the collective behavior of the solar wind particles, 2) the distribution of particle energies, 3) direct particle-particle interactions, if any, and 4) particle-field interactions.

Mission Requirements

For investigation of particles and fields in the near-earth vicinity, the system would be an array of satellites that

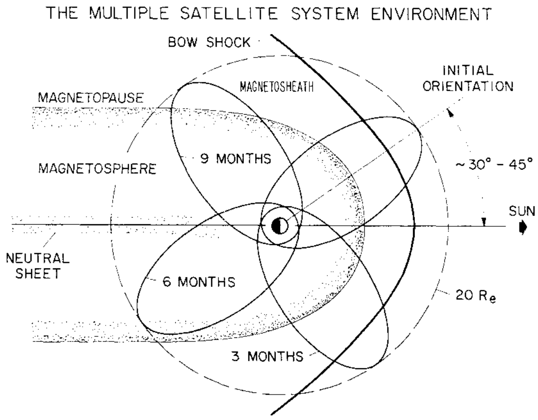

Fig. 4 Magnetospheric system environment. For an approximate sunline lead time of 30 to 45 days, orbital orientations are shown for each quarter of the first year's lifetime of the system as the orbit precesses about the earth, exposing the system to different environments.

traverses the magnetosphere, magnetosheath, and near-interplanetary space. An appropriate orbit for the mission would be one that is highly elliptical with an apogee of approximately 20 R_e and perigee in the neighborhood of 1000 km. Because the nominal orbit position of the array will remain essentially fixed in inertial space, the subsolar point will rotate with the sun line about 1° per day, so that at the end of six months in orbit the system will have rotated 180° from its initial orientation, with apogee near the subsolar line, to an apogee position that is roughly in the earth's shadow near the neutral sheet. At the end of a year the system will have returned to its starting position (Fig. 4).

Scale Factors and Time Variation

For solar plasma flux phenomena outside the bow shock and magnetosheath, the disturbances of interest propagate within the bulk solar wind at velocities from 100 to 1000 km/sec and do so typically in the general direction of the solar radials although any propagation vector direction is possible. Superposition of these disturbances on the bulk solar wind results in apparent disturbance velocities that may range up to several hundreds of kilometers per second in any direction, fluctuating further in magnitude and direction after passage through the bow shock.

These disturbances can be easily detected with single-satellite experiments as has been shown by the several instances of near-simultaneous measurements of similar disturbance phenomena by two Explorer satellites utilizing identical detectors at widely separated distances ($\sim$4 R_e) in the magnetosheath. But to accurately determine the velocity vector of such disturbances requires at least four identical sensors, each scanning at 1 Hz, and orthogonally separated by approximately 1500 km which would allow for differentiation and resolution of disturbance components for even the worst case, i.e., when they are propagating in the same direction as the solar wind.

Other types of disturbances are also triggered by solar wind fluctuations; those related to magnetic field variations have velocities of several gyroradii per second, i.e., up to 200 km/sec, whereas those disturbances associated with electromagnetic wave propagation (Alfven, magnetohydrodynamic, and magnetosonic) have characteristic velocities of the order of 30 to 50 km/sec.

Conditions within the magnetosheath and magnetosphere are highly responsive to the turbulent manifestations of plasma shocks, and disturbance velocities in these regions are generally higher than in interplanetary space, with disturbance scale dimensions often considerably greater than the gyroradii for these regions.[3,9]

Single-satellite magnetic field measurements in the region outside of the shock wave have been characterized by substantial differences in magnitude and direction on a time scale of fractions of seconds. These variations are currently interpreted as resulting from magnetic fields being "carried along" with the solar wind, and whose scale dimensions are equivalent to those associated with solar wind fluctuations.[4] Within the magnetosheath the magnetic field parameters are also highly variable but return to quite static and predictable conditions in the magnetosphere itself.

Sensor Characteristics

One goal of the first generation of the multiple satellite system would be to evaluate the three-dimensional scale factors, time variations, and vector parameters of the magnetic field and plasma phenomena discussed previously. It is imperative, therefore, that the specific particles and fields sensors carried by the individual satellites in the cluster, as well as the vector satellite separation distances, be compatible with the resolution of these characteristics.[10]

Magnetic field sensors of the fluxgate type, having the capability of providing a resolution of the total magnetic field vector in a matter of milliseconds, while still accommodating the spin rate of the spacecraft, would be appropriate. The magnetometer system is comprised of three orthogonally mounted field sensors, each having the direction response of a free space dipole and all three essentially noninteracting. Each sensor output, corresponding to the magnitude and direction of the sensed field, is sampled periodically, spin demodulated, and subjected to a second sampling process prior to transmission to the ground. The output vector directions are referenced to a line from the spacecraft to the sun so that spacecraft coordinate maintenance is essential to the experiment. Spin demodulation is used to transfer the inputs of the spinning spacecraft to a stationary coordinate system fixed with respect to the reference line. A flip-calibration mechanism allows for determination of magnetometer offset due to spacecraft perm and sensor drift which could interfere with measurements of low field levels. Additional frequency response could be achieved with the inclusion of a search coil magnetometer, if necessary. However, only one component of the magnetic field is resolvable with this instrument, and one is forced to accept the assumption that the total field varies in the same manner as the one component being sampled for these higher-frequency field fluctuations.

Plasma sensor requirements are considerably more demanding than existing instruments can provide, especially in terms of resolution capability. Typically, the quadrispherical plate electrostatic analyzer has a limited instantaneous angular field of view and requires significant integration time at each given energy level and view direction. The instrument is usually oriented to view outward transversely from a spinning satellite so that the motion of the satellite provides an azimuthal spatial scan. This scan, combined with sequencing through a series of elevation scan plates or sectors, results in coverage of most of a full spherical solid angle. The balance, two small cones parallel to the satellite spin axis, will not be sampled, but this is of little consequence since it has been established that both the solar wind flow and the anisotropy of thermal energy directions are aligned roughly in the ecliptic plane and, since the spin axis is nearly normal to the ecliptic, the sampling will actually take place parallel to that plane and be entirely sufficient.

To alleviate the timing resolutions inherent in plasma instruments of contemporary design, new techniques for rapidly scanning the solar wind energy spectra involving a linearly increasing voltage or ramp rather than discrete energy stepping should be applied. This will allow the measurement of the ionic energy spectra in a time short compared to the spin period of the spin-stabilized spacecraft.

In this configuration, the energy window of the detector is proportional to the voltage applied; the faster this voltage is changed, the more rapid is the full energy spectra scanned. Unfortunately, the range of energy spectra that can be scanned by a single instrument of this type is limited, but it can be enhanced, especially on the low energy end, by adding other detectors such as the Low Energy Proton-Electron Differential Energy Analyzer (LEPEDEA).

Rounding out a compatible particle and fields instrument complement would be an electric field meter to cover the complete electromagnetic characteristics of the regime. Such an instrument would consist of nothing more than a pair of short, crossed dipoles or some other simple antenna configuration located on a boom in such a manner as to present a preferred orientation to the satellite spin vector, thus facilitating spin demodulation.

The data output resulting from resolution of the total magnetic field vector at a frequency of 3 scans/sec is approximately 100 bits/sec if onboard spin demodulation is incorporated. The current plasma probes, on the other hand, require a data rate of approximately 1000 bits/sec to resolve their full energy scan and associated modes. A full order of magnitude reduction in this parameter can be achieved with the ramp-type plasma probe and still permit time resolution of a few milliseconds or less. Data rates for the auxiliary instrumentation would depend upon the modifications of existing equipment that would be required, but, in all probability, would not exceed a total of 1000 bits/sec including housekeeping and other operational information.

System Requirements

Certain constraints and guidelines are imposed upon the system in order to achieve the stated scientific objectives. Specifically, the nominal orbit and the satellite array geometry are fairly well bracketed by the regimes of interest and the phenomena scale factors, respectively. These, in turn, affect the subsequent scheme of operations.

Nominal Orbit and Orientation

Because the distance from the earth to the bow shock varies with the intensity of solar activity, it would be desirable to employ an orbit that is highly elliptical, extending well beyond the shock front and into interplanetary space. An orbit with an apogee radius of 20 R_e and a period of approximately 48 hr would suffice. Then, during the lifetime of the system (6 months to 1 yr), the apparent rotation of the semimajor axis will allow coverage of the subsolar region immediately after achieving orbit; primary coverage will subsequently rotate around through the magnetospheric tail and neutral sheet (Fig. 4). The individual satellites should be oriented with their spin axis normal to the orbital plane which, if properly positioned, would lie near the ecliptic. Such an orientation would optimize plasma measurements.

Array Configuration

A noncoplanar array of satellites is necessary to guarantee detection of disturbances propagating in all directions. The minimum number of satellites constituting a truly three-dimensional, noncoplanar, synergistic array is four. Ideally, the geometry should require all three orthogonal separation distances to be roughly equivalent and of the order of 1500 km. Such an array would adequately detect and resolve plasma disturbance propagation velocities in the range of interest to at least 15% or better. This uncertainty in the disturbance velocity will increase with the apparent propagation velocity and decrease with the magnitude of the separation distance. It is to be expected that the array will not remain constant throughout the system lifetime but that differential drag, initial orientation errors, and incremental errors in the deployment velocities will affect the manner in which the individual satellites behave. The array will also have a tendency to coalesce and disperse cyclically over each orbital period so that these perturbations and orbital positions must be accurately predicted to achieve a configuration compatible with the scientific and operational requirements.

Data Transmission and Tracking

Utilizing the existing Space Tracking and Data Acquisition Network (STADAN) at the proposed S-band frequencies for orbital communications imposes some restrictions on the system, the most stringent of which is the data duty cycle. During active experiment periods, scientific data would be acquired at a rate of approximately 1200 bits/sec, resulting in a total data output on the order of 10^8 bits during a typical two-day orbital period. This assumes that data would be acquired during only half of the over-all orbital time—the other half involves those portions of the orbit that are of little scientific interest. For data reduction purposes, the vector intersatellite distances should be known to accuracies of a few hundredths of a percent, requiring individual satellite ephemeris predictions of $\sim$50 m for the minimum 1500-km separation distances. Because of the cyclical variation in array dimensions, some alleviation of the data storage problem might be achieved by programing real-time operations for those periods when the entire array of satellites is covered by a single ground antenna beam or, if practicable, covered by the beams of two antennas at the same ground station or at two widely separated ground stations.

Technical Approach

An analysis was performed to determine the most propitious manner in which the synergistic multiple satellite system might be employed to study the solar-terrestrial environment of particles and fields interest and to assess the system's capability with regard to the scientific objectives.[10] Within these guidelines, the following technical system goals were formulated: 1) establish the array with a single launch vehicle of the Thor-Delta series, 2) utilize individual satellite spin stabilization with spin axis nearly normal to the ecliptic, 3) obtain maximum coverage of the subsolar region early in the orbital lifetime of the system, 4) achieve separation distances of at least 500 km (preferably 1500 km) with minimum growth rates during the system useful lifetime, and 5) attain a system useful lifetime of a minimum of 6 months (preferably 1 yr) with high reliability.

Deployment Options and Configuration

Deployment of the multiple satellite array from a single launch vehicle is desirable from an economic as well as a geometric accuracy standpoint. The alternatives require deployment of an accurate and enduring array with four separate satellite launches or two paired launches—in either case, a most formidable feat. Options in the array deployment from a single payload are then dominated by the following factors: 1) the entire Delta payload is spinning and the individual satellites are to be spin stabilized from this source of energy, 2) the spin axes of the individual satellites must be as nearly normal to the ecliptic as possible, and 3) the orbital period variation between satellites must be held to an absolute minimum to avoid excessive separation growth along the orbital path. Within these constraints, a number of deployment approaches are possible. In general, it will be necessary to rotate the payload spin axis from roughly parallel to nearly normal to the ecliptic. It is desirable to provide this rotation through the use of a single pallet or dispenser prior to satellite separation, rather than incorporate attitude reorientation capability into each of the satellites.

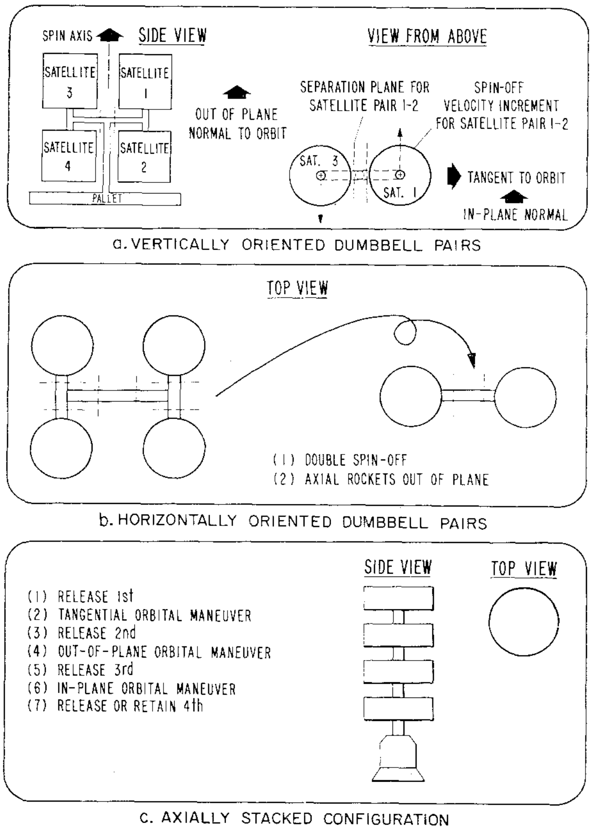

Fig. 5 Pallet arrangements for three-dimensional array deployment. Several methods of deployment are shown, all based on prior pallet orientation where the spin axis is normal to the ecliptic.

Pallet reorientation allows for several primary three-dimensional deployment options as illustrated in Fig. 5. A number of second-order variations of each of these options is also possible. One option involves the spinoff of two pairs of satellites in vertically oriented dumbbell configurations followed by axial separation with thrusters to accomplish the out-of-plane components. Another option involves the spinoff of two pairs of satellites in horizontally oriented dumbbell configurations. These dumbbells are again separated by a second spinoff which effectively doubles the total spin separation velocity vector for two of the satellites. Out-of-plane separation is then accomplished by axial thrusters provided for at least one of the satellites. Both of these options incur orbital period errors during spinoff and, due to the open-loop nature of the separation maneuver, these errors are essentially uncorrectable.

A third option involves an axially stacked payload configuration with a lateral thrusting capability incorporated into a single "master" satellite which includes the pallet. The necessary maneuvers for deployment of the array are accomplished using pulsed lateral thrust and axial thrust provided by the pallet. Satellites are simply "dropped off" at appropriate positions (conditions). All desired separation components can be provided and, since the individual maneuvers are separated by "track-and-trim" correction periods, no open-loop uncorrectable steps are involved. Variation in orbital periods between satellites can thus be reduced to an entirely acceptable level. Also, so long as the pallet system of the master satellite remains active (i.e., has sufficient residual cold gas), additional orbital maneuvers with the thrusting system of this satellite can be accomplished to partially optimize the configuration of the array during the usable lifetime of the system.

The configuration for the over-all payload as mounted on the launch vehicle is illustrated in Fig. 6. Individual, toroidal-shaped satellites are relatively large in diameter (52.2 in.) and small in height (9 in.), with a hollow center section. A 14.7-in. equipment mounting surface permits a usable volume of more than 50,000 in.^{3} per satellite. Pallet components are located in the central tube with structure attached to the master satellite mounted second from the bottom. The small length-to-diameter ratio for the individual satellites allows the total payload stack to have a favorable moment of inertia for spin stability, and this characteristic is improved as each satellite is deployed.

To satisfy the severe magnetic cleanliness requirements (i.e., to minimize the magnetic effects of the spacecraft electronics on the magnetic field sensors), each satellite will carry two 6-ft magnetometer sensor booms. These would be deployed by means of extendable telescoping sections and would be oriented slightly downward to avoid shadowing of the solar cells which would otherwise cause magnetically unacceptable transients in the power subsystem.

Orbits and Control

The characterisitics of the system payload configuration described previously are largely a result of the nature of the orbital and control dynamics associated with deployment of the system and the subsequent orbital perturbations. Launch into an orbit near the ecliptic plane by a Thor-Delta series launch vehicle will result in the spin vector of the over-all payload lying nearly in the ecliptic plane. Rotation of the payload so that its spin axis is approximately normal to the ecliptic prior to individual satellite separation can be accomplished by the use of a pulsed attitude control thruster whose firing periods are appropriately keyed during the spin cycle by reference to the sun line—similar to that which has been effectively utilized for the Pioneer and some of the Explorer series of spacecraft. Precessional disturbances resulting from the application of the erection torques are supressed by precession dampers included in the individual satellites.

The practicability of deploying and maintaining a three-dimensional array of satellites with separation distances of similar magnitude is basically governed by the nature of Keplerian orbit dynamics with superimposed perturbation effects. Fundamental deployment components for the unperturbed orbit, utilizing the accepted orthogonal relative coordinate system, are illustrated in Fig. 7. It is interesting to note that the lead time or tangential separation distance varies in a manner that is proportional to orbit velocity,

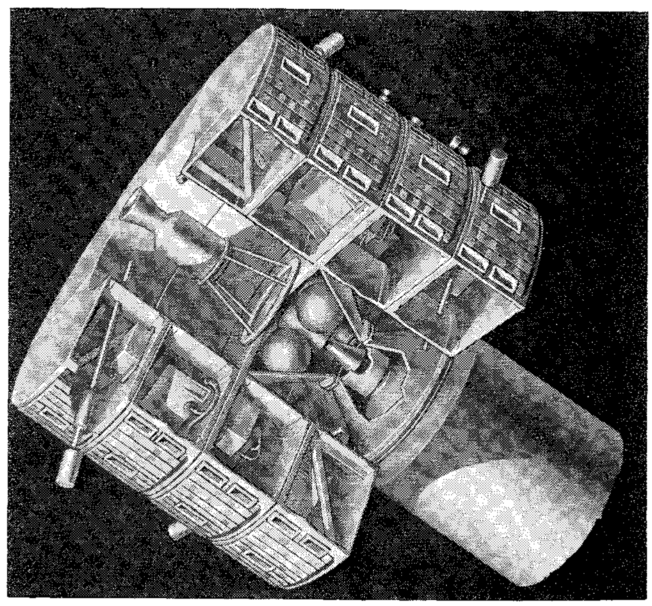

Fig. 6 Stacked payload configuration.

i.e., large separations at perigee where velocity is large and vice versa at apogee. The orbit period difference, on the other hand, results in a continuous secular growth of this same separation component. Another interesting point is that, for these geometric separations, the deployment maneuver can be accomplished at varying points in the orbit, resulting in different patterns of separation; however, the point at which the separation maneuver is accomplished will be common to both the original and the deployed satellite orbits.

It would be desirable, in the interest of minimizing the velocity requirements for separation, to obtain large sensitivities of in-plane normal (D_{in}) and out-of-plane (D_o) separation for unit separation velocity. Alternatively, since the growth of tangential (D_t) separation will generally be larger than desired due to orbit period errors, the sensitivity of this separation component to the applied velocity should be minimized. As shown in Fig. 8, these objectives result in the selection of deployment points fairly close to apogee, which provide the desired pattern of sensitivities. Such an approach, however, also produces relatively small array separations near apogee, so that the scientific experiments requiring nominal sensor separations would best be conducted lower in the orbit, e.g., in the vicinity of 10 to 15 R_e.

As indicated in Table 1, the various perturbation sources have effects on both the orbit and the array characteristics as well as on the spin axis orientation for the individual satellites. The relatively strong effect of the solar perturbation on perigee altitude combined with the extreme sensitivity of the array to small orbital period variations results in several important interactions.

The selection of a launch window which results in a nominal apogee point "leading" the earth-sun line by some 25 to 45 days (degrees) permits sufficient time to accomplish the final track-and-trim maneuvers, to fully check out the operational system by the time the orbits are passing through the subsolar region, and to provide a longer operation time

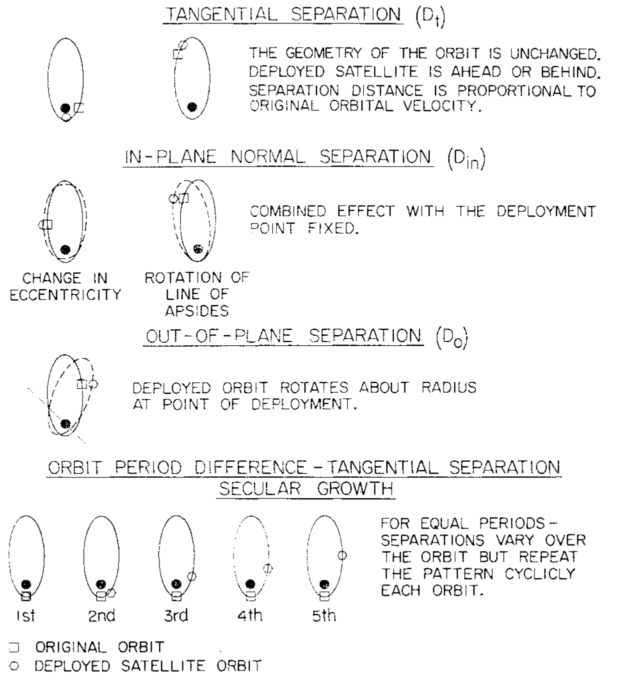

Fig. 7 Effects of deployment velocity increments. Shown are the various qualitative geometric effects associated with the orthogonal maneuvers designed to achieve a three-dimensional tetrahedron whose sides will have minimal differential growth.

within the "front" side of the magnetosphere, a region of scientific significance.

With this launch window constraint, however, the solar perturbation results in an initial decrease in orbit perigee for the first several months of the mission life as shown in Fig. 9. Since the perigees of the several deployed satellites will vary

Table 1 Perturbation effects

- Orbital effects
  - Atmospheric drag
    - Increases orbit period
    - Reduces apogee radius
    - Negligible effect on perigee altitude
    - Negligible effect on orbit orientation
  - Solar attraction
    - Strong effects on perigee altitude
    - Negligible secular effect on period
    - Affects orientation
  - Lunar attraction
    - Small effect on perigee altitude
    - Cyclical effect on velocity
    - Slight secular effect on period
    - Affects orientation
  - Earth oblateness
    - Negligible effect on orbital dimensions
    - Affects relative orientation of orbits
      (alters array configuration by virtue of apsidal precession and nodal regression)
- Spin axis drift
  - Aerodynamic torque
    - Precesses spin axis about velocity vector
    - Sensitive to center of mass location
  - Solar pressure torque
    - Precesses spin axis about sun line
    - Sensitive to center of mass location
  - Gravity gradient torque
    - Precesses spin axis about orbit normal
    - Sensitive to moment of inertia differences
  - Magnetic torque
    - Precesses spin axis about mean ambient magnetic field vector
    - Sensitive to satellite magnetic dipole moment

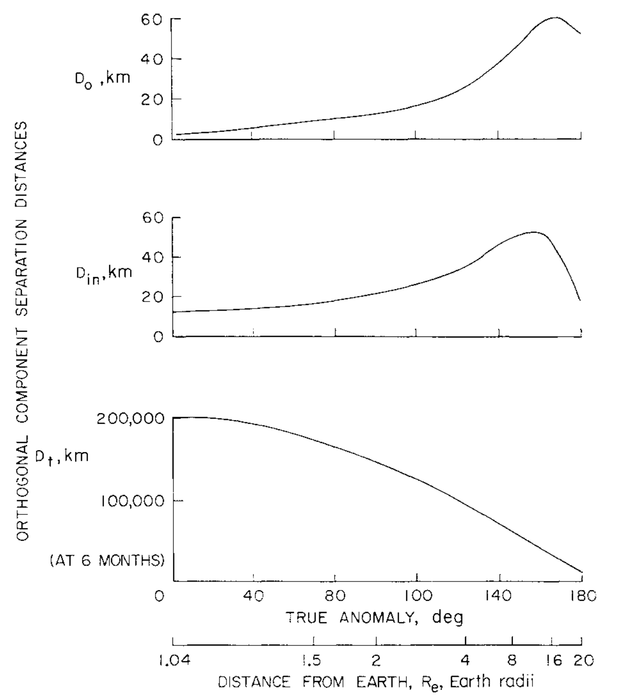

Fig. 8 Deployment location parameters. The variation in the orthogonal separation distances of the tetrahedral array is plotted against the point in the orbit at which these separations occur. The curves are referenced to a point at 12 R_e on the descending leg of the orbit where the individual separation distances are measured, and to velocity increments of 1 m/sec.

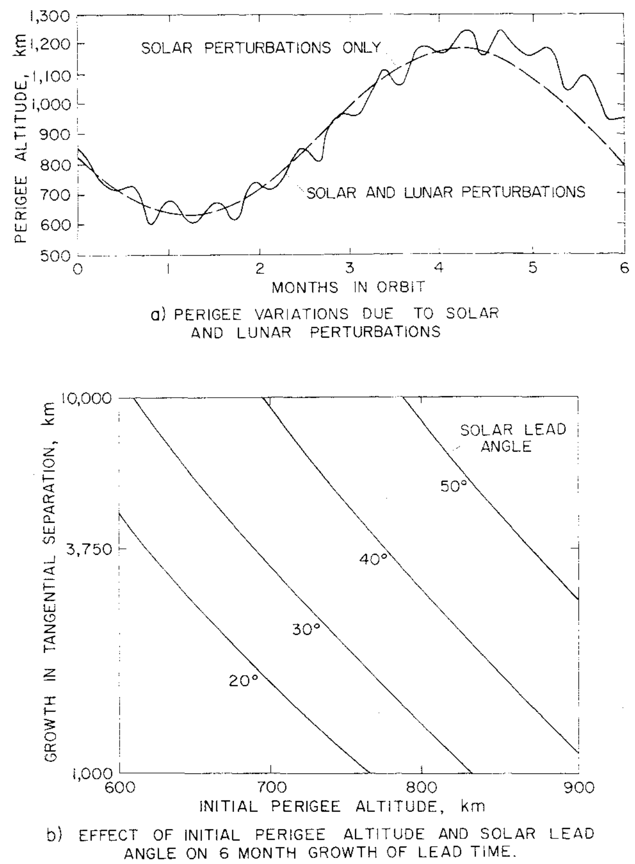

Fig. 9 Perigee altitude considerations. Note the cyclic nature of the effect of perigee altitude dependence on solar perturbations and the superimposed lunar perturbation of approximately 50 km as shown in a. The combined effect of initial perigee altitude and launch date (solar lead angle) upon the tangential separation growth indicates (b) that for a 30° to 40° solar lead angle the initial perigee altitude should be approximately 800 to 900 km in order not to exceed a tangential separation growth of 1000 km in 6 months.

due to the deployment maneuvers, the minimum perigee altitude that is reached must be sufficient to avoid excessive changes in orbit period due to atmospheric drag so as to minimize tangential separation growth during the system lifetime.

Deployment Sequence and Optimization

The deployment sequence (Fig. 10) involves a series of orbital maneuvers accomplished by the pallet propulsion system such that the satellites are released under appropriate conditions to provide the optimum final array. Deployment extends over several orbits in order that ground-based tracking can be utilized during track-and-trim periods following maneuvers to verify that each satellite's orbital period corresponds to the desired accuracy. This technique allows the ultimate growth of tangential separation distance during the system lifetime to be controlled within specified limits.

Several factors are involved in the optimization of the deployment sequence and the nature of the maneuvers completed. Perturbation effects result in a change of 100 km/month or less in the in-plane normal component of the separation, and less than 50–75 km/month change in the other components. The tangential separation growth is limited by the closed-loop nature of the deployment maneuvers; however, if some such growth is desired during the life-time of the system, it could be incorporated as a controlled factor. The in-plane array position for the master satellite incorporating the pallet components could also be adjusted subsequent to initial deployment for the period during which the lateral thrusting equipment remains serviceable.

ACS/LTS and Aspect Sensing

This deployment scheme requires a capability for thrusting laterally with respect of the payload spin axis. Since the center of gravity of the spinning payload shifts as the satellites are deployed, the Lateral Thrusting System (LTS) must incorporate a logic and control capability that allows the line of action of the thruster to be maintained through the payload center of mass for the various payload configurations encountered during deployment, as well as provide proper keying of the thrust pulses during the spin cycle. A block diagram for this system, including its integration with the Attitude Control System (ACS) for spin axis erection in shown in Fig. 11. The lateral thrust pulses are spin-phased by reference to the sun line similar to that successfully employed by the SYNCOM satellite. Capability for alteration of the thrust line to the three positions required for the LTS is provided by two nozzles and two explosively operated valves. A typical sequence of operation would be: 1) first nozzle open, second nozzle closed; 2) both nozzles open; and 3) first nozzle closed, second nozzle open.

The aspect sensing function necessary for accurate erection of the spin axis is provided by four IR horizon crossing indicators (one per satellite) and a single digital solar aspect sensor on the "master" satellite. Four additional conventional sun sensors would be located on the pallet to minimize the logic requirements associated with the control of the ACS thruster. Each IR sensor, having a dual field of view with axes 180° apart and inclined at 3.5°, would indicate the deviation from the normal to the local vertical to allow evaluation of spin axis tilt. This combination of aspect sensors would permit execution of a reorientation maneuver in less than half an orbit and achieve a spin axis orientation accuracy of approximately 0.2°.

Communications and Tracking

The data acquisition, command, and tracking support of the system operating in the solar-terrestrial investigative mode would be provided by STADAN within the following constraints: 1) experiments will be conducted over a substantial portion of a long duration (48-hr) orbit, resulting in extended periods of data acquisition; 2) all data acquired by the four satellites must be accurately referenced with respect to time and relative position, and the array dimensions are such that all satellites may not, at times, lie within the beam of a single tracking dish even when good line-of-sight conditions exist; and 3) the demands for support which can be placed upon a given station will be limited by its commit-

Table 2 Communications characteristics

Data acquisition
1,280 bits/sec data acquisition rate (5 instruments plus aspect and housekeeping data)
Record for 24 hr per orbit (from 8 to 18 R_e)
Amenable to real-time operation
Downlink
1-W S-band transmitter; 6.4-db fan antenna
1,280 bits/sec real time over the entire orbit (48 hr)
30,720 bits/sec recorder readout at 7.5 to 4.0 R_e
STADAN duty cycle in the record mode: 1 hr/station/day (less than 6 hr/station/day total)
No pallet downlink
Command
Pallet and satellite command systems
Omni receiving antennas
Tracking
Goddard Space Flight Center Range and Range Rate System
S-band beacon on the pallet
Relative satellite position accuracy approximately 4 km

ments to other programs—a nominal limit of 4 to 6 hr of support by a single station per day has been predicted.

These considerations necessitate the use of onboard data storage for each satellite although a capability for partial real-time coverage at the expected satellite data rate would be available. However, this capability should most properly be utilized for redundant acquisition of particularly critical data. With such a storage and readout sequence, the requirements on the available STADAN ground stations are well within the predicted support limits.

In accordance with the proposed S-band upgrading of STADAN, the use of S-band command, control, and tracking links is anticipated. Table 2 summarizes the over-all communications characteristics for the multiple satellite concept as currently envisaged. The 30-ft receiving antennas would be utilized with uncooled preamps while the selected satellite antennas are to be cavity-backed, slotted arrays as depicted in Fig. 12. In addition to providing the desired antenna pattern, these slotted antennas are compatible with the pay-load design criteria involving low length-to-diameter ratio, axially stacked satellites.

Tracking of the individual satellites to provide the necessary ephemeris accuracy for calculation of intersatellite separation vectors would be based on the use of the Goddard Range and Range Rate (GRARR) system as incorporated into STADAN. Each satellite will thus carry a ranging transponder and will enable the prediction of the resulting orbit ephemeris and relative satellite location to be calculated to an accuracy better than 4 km.

Table 3 Weight statements

Satellite		
Science instruments		26.0 lb
IR aspect sensor		1.5
Power system		10.0
Solar cells	4.0 lb	
Battery	1.5	
Power conditioner unit and cabling	4.5	
Data management system		27.0
Command receiver-decoder	3.0 lb	
Data processor	3.0	
Tape recorder	8.0	
Transponder	7.0	
Transmitter	2.0	
Antenna	4.0	
Temperature control		1.0
Structures and mechanisms		22.6
Primary structure	11.0	
Superstructure:		
Solar array	6.8	
Magnetometer booms (2)	2.0	
Mechanisms	2.8	
		88.1 lb
Pallet		
Data management system		2.5 lb
Command receiver-decoder	0.5 lb	
Beacon and antenna	2.0	
Power system—conditioner		0.5
Attitude control and lateral thrusting		38.9
Nitrogen gas	16.8	
Pressure reservoir	16.8	
Electronics	2.0	
Valves, nozzles, and plumbing	2.5	
Solar sensors (3)	0.8	
Structures and mechanisms		5.0
Solid rockets (2)		19.6
Solar aspect sensor		0.4
		66.9 lb
System		
Satellites (4)	352.4 lb	
Pallet	66.9	
Payload adapter	16.0	
	435.3 lb	

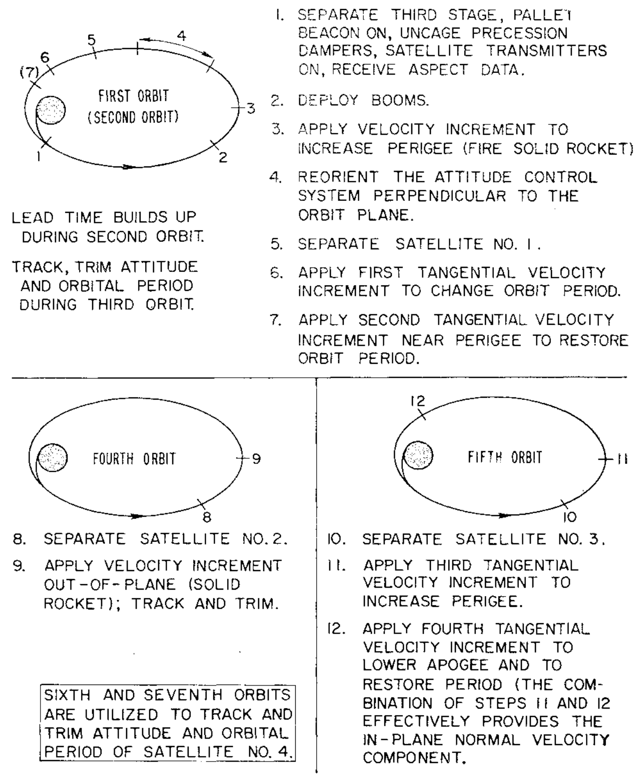

Fig. 10 Representative sequence of events designed to obtain in final array achievement and adjustment during the first 7 orbits (14 days).

Power System

A conventional solar cell/battery power system is incorporated, with the solar cells mounted directly on the peripheral surfaces of the individual satellites. Since the deployed satellites are stabilized with their spin axes nearly normal to the ecliptic, no solar cells are included on the end surfaces of them; however, these surfaces provide thermal control through the use of passive α/ϵ coatings.

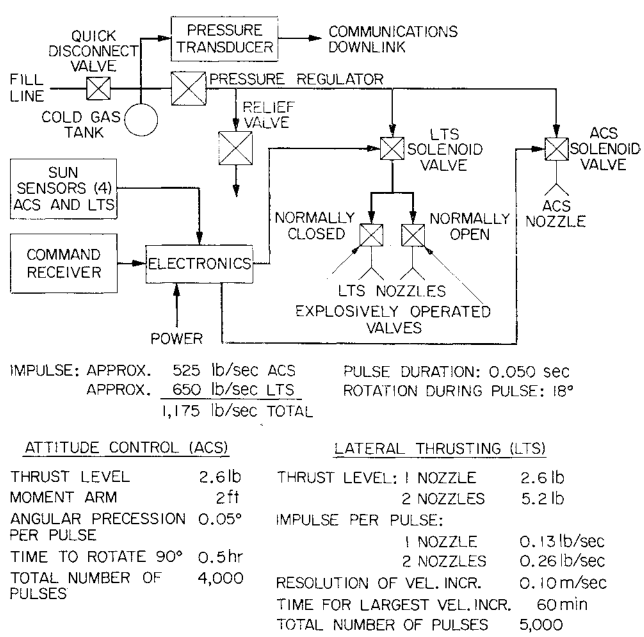

Fig. 11 Functional layout and some propulsion characteristics of the attitude control and lateral thrusting systems.

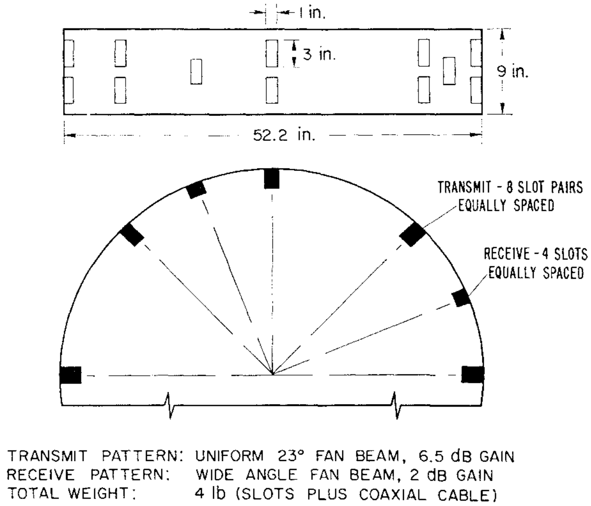

Fig. 12 Cavity-backed, slotted array antenna design. The geometry of the transmitting and receiving antennas is shown in its paired arrangement. The downlink transmitting antenna will be shared by both the telemetry and tracking transponder functions while the uplink will be used for commanding and transponding. Only the pallet assembly will be equipped with a tracking beacon.

Weight Summary

Weights for the individual satellites, the pallet components, and the over-all payload as launched are presented in Table 3. These weights are based upon use of a "master" satellite-pallet combination with some shared components. The individual satellites, although somewhat unusual in shape, are generally not too dissimilar to other current small spin-stabilized satellites. The additions required for dispensing the satellite into the proper geometrical array are not excessively complex and have been separately demonstrated in current successful spacecraft systems.

Assuming launch by a Delta series DSV-3E (Improved Delta) vehicle, total allowable payload weight of 436 lb permits a pad of some 10 lb. If it were necessary to obtain an increased weight margin, consideration would be given to the use of an alternate series DSV—either the −3J or the −3M with the TE-364 third stage—wherein an additional 80 lb of payload capability would be achieved under identical orbital circumstances. The final selection of the launch vehicle, however, would depend on the availability of vehicles of proper capability in the NASA launch stable at the time of development.

Summary

The development of a multiple satellite array appears to form a logical and timely step in the evolution of instrumentation and systems for the near-earth particles and fields laboratory. An array of co-orbiting satellites with controlled and predictable separation distances can resolve ambiguities currently inherent in data derived from single-point sensors. The system consists of four satellites attached in a stacked arrangement to a pallet that orients and deploys the individual satellites at predetermined positions along the orbit so as to achieve an ordered tetragonal (noncoplanar) array whose geometric growth with time can be minimized. The Thor-Delta Series of launch vehicles can, with an acceptable payload margin, launch the system into a highly elliptical, 20 R_e orbit. A communications and data handling system has been described which could maximize the bit rate using developed components, including data storage, while the ground network for tracking, data acquisition, and commanding could easily be handled by STADAN utilizing S-band frequencies.

Each satellite of the system would be identical to the extent that any of the four could accommodate the pallet assembly. The individual satellite configuration has been sized for maximum instrument weight, volume, and data rate based on a possible instrument complement including a ramp-type plasma probe, a fluxgate magnetometer, a search coil magnetometer, a low-energy particle detector, and an electric field meter. In addition to the scientific instruments, the satellites would contain the data handling and communications system, including a transponder and associated antenna, the power system, solar and horizon aspect sensors, attitude control and lateral thrusting systems, small solid rockets for array deployment, a precession damper, and house-keeping status reporting equipment. The magnetometer sensors would be mounted at the end of deployable booms to reduce the spacecraft-induced field intensity at the sensor. Each satellite of the array would be spin-stabilized.

The relatively conventional and straightforward functional design of the satellites and the proven technology of the pallet subsystems indicates that this synergistic multiple satellite system can be developed with extensive use of off-the-shelf components. The system can be launched and the appropriate geometric orbital array achieved in conformance with the scientific objectives without unusual extensions of launch operations or spacecraft technology. Successful system implementation could thus be accomplished with confidence at a reasonable scientific mission cost.

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