Preliminary Design of a Solar-Electric Test Satellite

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This paper describes a non-nuclear orbital test system capable of providing the long-duration space environment and power level required to qualify for space the electric propulsion engines now under NASA development. The vehicle makes use of a deployable flat-panel, solar-cell array operating in a near-polar orbit to provide reliable continuous electrical power for periods in excess of 150 days. The operation of electric thrusters will cause the test vehicle to spiral slowly upward, thus presenting an excellent platform from which to conduct such scientific experiments as a spatial survey of the intensities of the energetic particles trapped in the earth's magnetic field and a World Magnetic Survey.

Introduction

RESEARCH and development programs in the area of electric propulsion have met with considerable success during the past few years, and already several types of electric thrusters have passed the laboratory demonstration stage. Within the next few months, a series of space tests will be initiated to demonstrate the feasibility of these devices as space propulsion systems. As a result of the rapid progress made to date, it is now possible to consider electric thrust devices for near future missions.

Any feasible lunar or interplanetary missions utilizing electric propulsion systems will require power-supply specific weights that are practical only with nuclear powerplants. The stretchout in the development of space-oriented nuclear power systems has made it evident that to capitalize on the recent excellent progress in the development of the electric thrusters, it is desirable that an orbital test system using non-nuclear power sources be developed to provide the long-duration space environment required to space-qualify the electric systems.

The Hughes Aircraft Company has conducted preliminary studies which indicate that a solar-electric test satellite can provide all the requirements for the forementioned system and that solar electric-propelled earth satellites can perform useful scientific missions, and these studies are being continued.

The purpose of this paper is to present a preliminary system design of a solar-electric test satellite which is capable of performing certain scientific missions of a more comprehensive and sophisticated nature than its chemical rocket-propelled counterparts can perform. The principle characteristics are given in Fig. 1 and Tables 1 and 2. The missions include such scientific experiments as a spatial survey of the intensities of the energetic particles trapped in the earth's magnetic field and a World Magnetic Survey.

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Summary

The Solar-Electric Test Satellite (SETS) system described herein has the capability of accomplishing electric rocket engine tests and can accommodate a scientific package weighing up to 150 lb. It uses the Thor-Agena B launch vehicle, utilizes solar cells for primary power, and has a three-axis cold-gas attitude-control system for the spacecraft proper and an independent two-axis power gimbal drive attitude-control system for the scientific package. An onboard high-density magnetic tape storage system is provided with a capacity in excess of 11×10^6 bits. The design mission provides for 90 days or more of useful engine test time. The initial orbit will be a twilight orbit inclined 97.7°. As the electric engines are operated, the orbit will be raised at a rate of 10 to 25 miles/day which will cause the precession rate to diminish and thus limit available test time before the vehicle enters the earth's shadow. At 300 naut miles and an engine thrust of 0.01 lb, the vehicle will remain out of the earth's shadow for about 150 days.

Scientific Mission Considerations

Energetic Particle Survey

One experiment that appears most compatible with the capabilities of the SETS system is that of making a spatial survey of the intensities of the energetic particles trapped in the earth's magnetic field; such determinations would be made as a function of positional coordinates, direction, and time. As indicated in Fig. 2, the SETS system has the capability of penetrating the entire inner radiation zone.

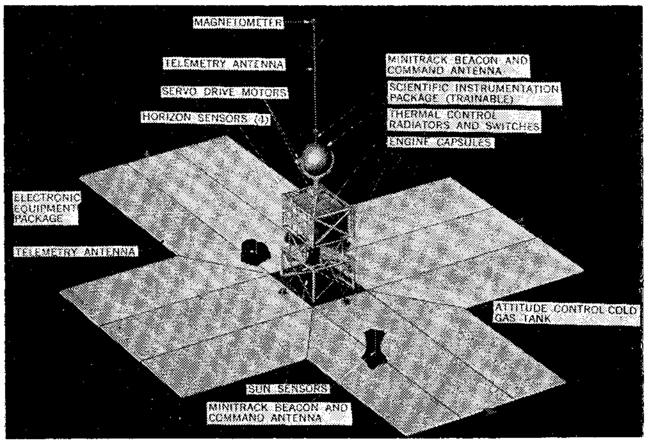

Fig. 1 Solar electric test satellite.

Most of the spacecraft launched to date have carried instruments designed to study the energetic charged particles. Information from present experiments continually indicates the need for new experiments, and these result in either the discovery of new and unexpected phenomena or the need for still more definitive experimentation with improved instrumentation. For example, experience from recent satellites shows that considerable care must be taken in the choice of detector size and sensitivity to avoid pulse pileup and counter overloading due to high particle fluxes.

Table 1 Principal characteristics of solar-electric test satellite, scientific mission version

Physical parameters	
Weight	1140 lb
Diameter	
Extended	316 in.
Retracted	60 in.
Over-all height	180 in.
Extended	180 in.
Retracted	148 in.
Attitude control system	
Satellite	
Type	Three-axis gas jet
Command	Attitude or rate control
Orientation	Roll axis slaved to sunline with earthline midway between yaw and pitch axes
Accuracy	Within $\pm 1°$ of any axis
Scientific package	
Type	Two-axis power gimbal drive
Orientation	*B* axis slaved to local earth magnetic field
Accuracy	$\pm 2°$
Telemetry system	
Satellite	
Carrier frequency	225–260 mc
Transmitter power	1 w
Antenna	Quarter-wave radiator
Antenna gain (min)	−3 db
Modulation	PCM/PM/PM
Subcarrier oscillator frequency	4096 cps
Data rate	1024 bits/sec
Data storage capacity	64,000 bits
Channel capacity	70, commutated at $\sim$100/sec
Scientific package	
Carrier frequency	250 mc
Satellite transmitter power	10 w
Satellite antenna	Quarter-wave radiator
Satellite antenna gain	−3db min
Ground antenna diam	60 ft
Ground antenna gain	30 db
Maximum range	3000 naut miles
Receiver noise figure	5.5 db
Over-all margin	9.8 db
Probability of bit error	10^{-5}
Modulation	Suppressed HF carrier
Data rate	40,960 bits/sec
Storage capacity	11,059,200 bits
Minitrack system	
Carrier frequency	136–137 mc
Beacon power	1 w
Antenna	Quarter-wave radiator
Antenna gain	−3 db min
Command system with minitrack antenna	
Carrier frequency	120–150 mc
Command data rate	512 bits/sec
Command storage capacity	Up to 64
Solar panels	
Total area	400 ft^2
Active area	90%
Design temperature	120° F
Total output power	3.10 kw
Type of cell	1 × 2 cm silicone N-P
Panel substrate	Al honeycomb with Ti facing
Batteries (2, with 12 cells each)	
Nominal voltage (Ag-Zn, LR-40)	18 v
Total capacity	1440 w-hr
Depth of discharge	40%

Because of the unique orbit of the SETS, it is extremely well suited for exploration of the inner belt. The fact that the inner belt is eccentric with respect to the earth (and the test vehicle's orbit) means that on each orbit the test vehicle will traverse different portions of the zone. This fact, coupled with the slow outward spiral, will permit measurement of any time variations in the inner belt. At present, the energy spectrum and its time variation are of interest in order to clarify the source and loss mechanisms for the inner belt electrons. Use of instruments very similar to those on Relay would permit measurement of the proton (1.0 to 60 Mev) and electron (0.25 to 1.5 Mev) spectrum. These instruments are compatible with the space, weight, and power capabilities of the proposed vehicle. Although it would be desirable to extend the measurements to lower particle energies, in the absence of other tried systems the package should be designed for the instruments on hand.

World Magnetic Survey

The International Geophysical Year (IGY) program in geomagnetism included a proposal for a world-wide survey of the earth's magnetic field, but major phases of the magnetic survey were deferred until a time of solar quiet so that the field could be surveyed with a minimum of interference from solar-induced disturbances. Present planning calls for the World Magnetic Survey (WMS), which uses data gathered since 1955, to heighten its efforts during the International Year of the Quiet Sun (IQSY), April 1964 to December 1965, when it is predicted that the minimum of solar activity will occur.

The actual geomagnetic field is somewhat irregular as a result of inhomogeneities in the earth's crust and, possibly, in the upper mantle as well. These inhomogeneities may be, for example, concentrations of rock that is rich in iron. Most

Table 2 Weight breakdown

Item	Weight, lb
Power supply	
Solar panels	400
Power converters (4)	160
Battery	34
	594
Engine pods	
Arcjet (2)	60
Ion (2)	40
	100
Electronics	
Data storage	8
Receivers	8
Transmitters	10
Antenna	4
Radiators and thermal switches	18
Miscellaneous	14
	62
Control	
Sensing	4
Electronics	4
Jet valve	1
Propellant	6
Tankage	9
	24
Separation	10
Scientific package	150
Structure	160
Contingency	40
Total	1140

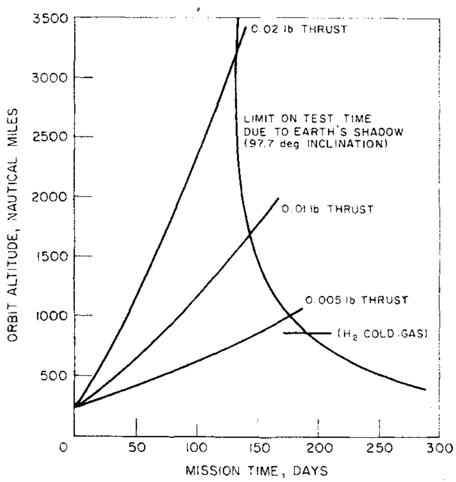

Fig. 2 Vehicle flight profile.

rocks, however, are magnetic to some degree, and large-scale features of the earth's crust commonly have magnetic anomalies or departures from uniformity in the field associated with them. Magnetic anomalies may be large enough at some locations to mask the character of the general geomagnetic field, but in mapping the field of the entire earth, the local anomalies and small-scale distortions tend to be averaged out. However, with the SETS this need not be the case. In traversing the orbital altitude range from 300 to 3000 naut miles the altitude is raised at a maximum rate of less than 2 miles/orbit, which is more than adequate for the detection of these local anomalies.

Scientific Mission Capabilities of SETS

Mission Requirements

A problem that immediately arises when a magnetometer is to be carried on a spacecraft is one of controlling the magnetic fields generated by the spacecraft. This control consists first of utilizing material and design techniques to reduce the spacecraft fields to a practical minimum, and then, if necessary, calibrating the spacecraft so that compensation can be made for the effect of any remaining fields. Further, if required, it may be necessary to place the magnetometers at a distance from the major source of the stray magnetic fields where the field intensity is well below the sensitivity of the magnetometer.

At any point the measured flux intensity depends on the orientation of the instrument acceptance angle with respect to the plane perpendicular to the magnetic field vector. Therefore, to obtain usable data, either the instrument package must be aligned with respect to the field vector, or the orientation of the instrument acceptance angle with respect to the field vector must be measured and this information stored with the count rate data. Although in principle either technique yields the same over-all information, the former is preferable because of the ease of data analysis. In fact, a detailed analysis of the two methods indicates that alignment of the instrument package with respect to the magnetic field is almost mandatory for precise measurements. Since the directions perpendicular to the magnetic field vector form a plane perpendicular to the vector, only two degrees of freedom need be provided to align the instruments.

Both the WMS experiment and energetic particle survey experiments require data storage capacities well in excess of that required for the electric rocket orbital tests. Typical data storage requirements for energetic-particles satellites today are in excess of 10^6 bits/orbit, which implies a maximum storage capability commensurate with acquisition capabilities of available tracking stations.

System Design

Since the basic SETS design, if used for engine tests only, has a gross weight of 866 lb, and the Thor-Agena B launch vehicle has the capability of placing a 1140-lb payload into a 300-naut-mile polar orbit, a substantial weight margin is available to accommodate scientific experiments. The design philosophy adopted here is to derive a spacecraft that can accommodate a scientific package having the maximum feasible size and weight which are compatible with the booster payload capability and nose cone dimensional limitations and which meet the engine test mission requirements. The design evolved (Fig. 3) can accommodate a scientific package having a gross weight of 150 lb and a maximum diameter of 27 in.

Practical considerations dictate that the spacecraft design should be such that the scientific package can be either an existing system, such as that for Explorer XII or Injun II(b), or one that can be launched into a useful orbit by a smaller launch vehicle, such as the Thor-Delta. The Injun II(b) scientific payload package has a gross weight of 72 lb, is contained within a 20-in.-diam sphere, and requires less than 4 w of power. The Explorer XII scientific payload has a gross weight of 83 lb, has all the scientific instrumentation within an octagonal package 5.5 in. high and with a 26.2 in. side-to-side diam, and requires 16 w of power. An examination of

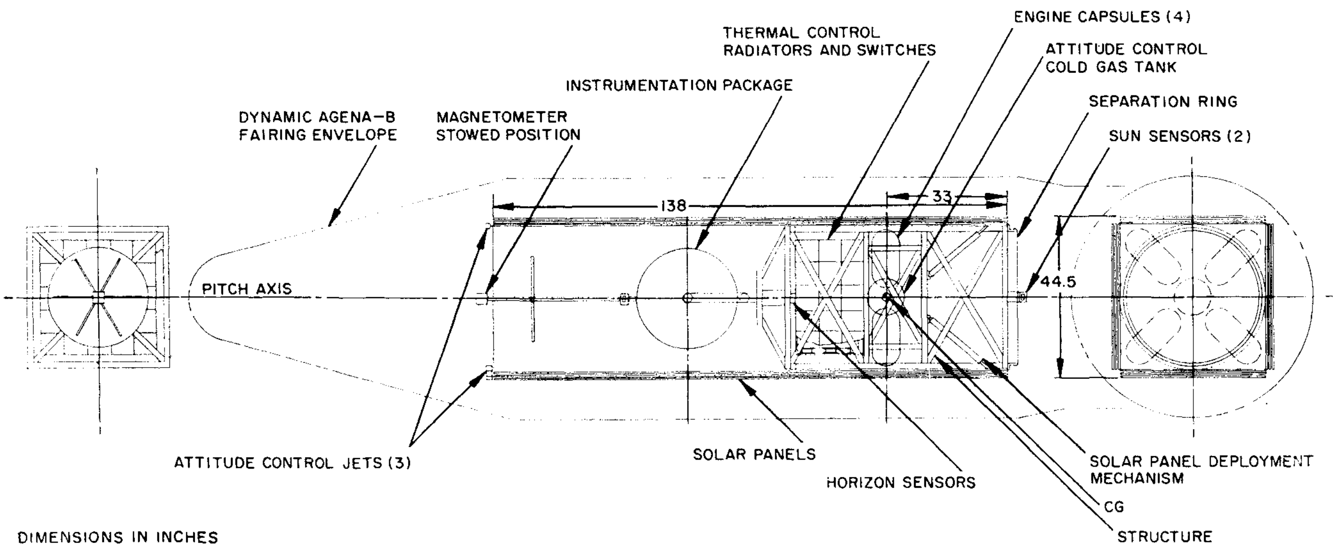

Fig. 3 Solar-electric test satellite general arrangement, launch condition.

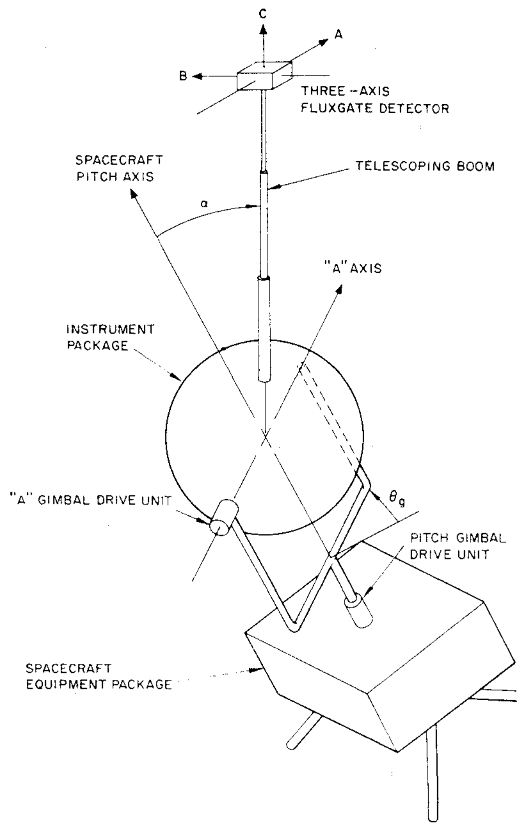

Fig. 4 Scientific package orientation system schematic.

Fig. 3 and Table 1 indicates that any of these packages can be accommodated by the present design, with only a minor vertical shift in the location of the basic electronic package and engine pods for balance purposes and to maintain the thrust plane through the system center of gravity.

Orientation System

The method selected for orientation of the scientific package to provide fields of view perpendicular to the local earth magnetic force lines is illustrated schematically in Fig. 4. A two-axis gimbal system of the fork type is used to mount the instrument package to the spacecraft. Power gimbal drive units permit the use of conventional bearings (selected for the space application). A three-axis flux-gate detector mounted to the instrument package by a pneumatically actuated telescoping boom provides two functions. First, it supplies the information required to control the drive motors to align the B axis of the detector (and, therefore, of the instrument package) to the earth field direction. Second, the field intensity measured by the B axis detector is telemetered to the earth to provide the magnetic field magnitude information. The two gimbal angles are also telemetered, and together with the known, controlled attitude of the spacecraft, provide information on field direction. Thus, the package orientation system inherently provides the information required for magnetic field mapping.

In operation, the pitch gimbal is driven in the appropriate direction to null the magnetic field component along the sensor A axis, thereby constraining the total field vector to the sensor B-C plane. The gimbal A axis is simultaneously driven in the appropriate direction to null the sensor C axis field component. Thus, the sensor B axis is held in continuous alignment with the local earth field by the two independently operating drive units.

If this simple two-gimbal suspension, without any angle resolution functions, were required to operate under unrestricted geometric conditions, several serious problems would result. These would include gimbal lock effects (or, stated in other terms, high-speed gimbal slewing requirements) and strong interactions between the two-axis control loops. The particular arrangement described, when used in a polar orbit, avoids these problems. The pitch gimbal rotates continuously at approximately constant velocity through one revolution (relative to the spacecraft) per orbit. Since the spacecraft revolves once per orbit, the instrument package, in inertial coordinates, rotates twice per orbit as required to follow the magnetic field vector. The gimbal A axis slowly oscillates, in the general case, through an angle never exceeding about 25°, depending on the varying "magnetic inclination" of the orbit plane. Therefore, the system never approaches a "locked gimbal" configuration, and the nonlinear effects, because the sensor C axis and the pitch gimbal drive axis do not coincide, are always quite small. The latter effects produce only small nonlinearities of the open-loop properties of the control servoloops and have no first-order effects on the positioning accuracy obtained.

Electrical connections from the sensor and the instrument package can be provided by means of flexible leads at the gimbal A axis bearing. At the pitch axis bearing this is not possible, since continuous rotation is involved. The most straightforward method of handling the connections at this point is the use of slip rings within the sealed and pressurized drive unit. Hermetic sealing is not required. A Teflon O-ring seal on the output shaft will provide an adequate seal in the vacuum environment, and a supply of lubricant within the unit with a vapor pressure of perhaps 10^{-2} mm Hg will insure that, even with the sizable seal leak rates, hard vacuum effects on material surface properties will not occur during the spacecraft operational life.

A conventional three-axis flux-gate system similar to the system used on the NASA-Hughes Surveyor Lunar Lander program provides higher accuracy than is required for this application (approximately ± 1 gamma). Figure 5 provides details of the actual detector elements. The core is formed by rolling a 1 mil 3×4-in. sheet of material with high permeability into a long thin scroll (0.1 in. in diameter) to reduce eddy current effects. This core is inserted into a plastic form; primary and secondary windings are wound on top of each other at each end. The two primaries will be connected in opposition and excited from a 400-cps source driving the core past saturation each half-cycle. The secondaries will be in series in an additive sense, so that without an external field the signals in the two secondaries will be equal and opposite due to the primary connection, and no net signal from their series connection will result. With an external field, however, one primary will go into saturation before the other, producing an unbalancing in the harmonics detected in the secondaries, especially the second harmonic.

This unbalance in the second harmonic will be amplified and phase detected to give a d.c. signal proportional to the net d.c. field in the sensor. The detected signal will then be used as feedback to excite a field in opposition to that induced in the sensor by external fields. Instead of using a separate winding for d.c. feedback, the secondary windings are used for this purpose. Sensitivity of the unit is such that a change of 10^{-3} oe in the net d.c. field of the sensor will change the output of the amplifier by 10 v, which is more than sufficient to meet accuracy requirements. The complete magnetometer will consist of 3 orthogonal elements mounted in a plastic block.

The gimbal drive unit incorporates a bidirectional stepper motor, gear train, and potentiometer position pickoff. The unit incorporates the sealing technique described earlier and is currently in the early phases of flight qualification for use in several applications on the Surveyor spacecraft.

Stepper motors are selected for this application for the following reasons.

1) They consume power only when stepping, for 0.1 sec/step. Using 1° steps at the maximum rate of one each 20

sec (pitch gimbal at low altitude), peak power consumption is 20 w and average power consumption is 0.1 w.

2) In the event of a circuit failure or some unforeseen difficulty, this type of motor can be readily operated in a command override mode by which a specific number of pulses will move the drive through an accurately known angle.

3) The control circuits can be operated in a switching mode, thus providing higher reliability than is obtainable with linear amplifier designs.

4) The unit is already in an advanced state of development and will be manufactured in significant quantities.

The telescoping boom will incorporate a keyway to prevent rotation during extension so that the flux-gate sensor orientation is restricted to that shown in Fig. 4. However, if it proves to be too difficult to control the orientation in this fashion to $\pm 1°$, as would be required to achieve over-all field mapping accuracies of $\pm 2°$, the keyway will be used to provide somewhat less accurate control of alignment (perhaps $\pm 5°$), and a coil in the spacecraft will momentarily produce a local field of known orientation to determine actual sensor orientation.

The attitude control system of the spacecraft as designed for the engine test mission is appropriate for this expanded mission objective with one exception. The attitude control precision, which was $\pm 10°$ for that system, must be improved to $\pm 1°$. There are two important consequences of this change.

1) The accuracy requirement of the horizon sensors must be changed to approximately $\pm 0.75°$ (including alignment errors). This change, while significant, is still well within the state of the art, since horizon tracking accuracies of $\pm 0.5°$ or better are achievable. A similar change is required in the sun sensors, but this presents no problem since an appropriate unit with $\pm 0.2°$ accuracy is already in final development for the Surveyor spacecraft.

2) The improved control accuracy will necessarily result in increased fuel consumption by the cold-gas jets. The parameters of the system will therefore be optimized from the standpoint of minimizing gas consumption. Since this tends to dictate very low jet thrust levels (limited, in effect, by the disturbance torque levels only), it may prove desirable to use two levels of thrust to avoid long durations of the initial stabilization and orientation operations. Again, if this modification proves desirable, the components required already exist in forms that have been specially designed for space environment applications. However, it is likely that only refinement of the control system parameters and a modest increase in total gas weight will be required.

Data Storage System

For continuous high-frequency mapping of radiation levels and magnetic fields for relatively long periods of time, a high-density magnetic tape storage system is proposed in addition to the core storage system used for the electric engine data. The upper limit of stored bits from these experiments depends on the minimum time available for data acquisition when in sight of a tracking station, and on the peak spacecraft transmitter power available. The average recording rate depends additionally on the maximum time interval between data acquisitions.

The least time available for data acquisition is in the lowest altitude orbit (300 naut miles) and is 478 sinϕ sec, where ϕ is the half-angle subtended horizon to horizon by the flight path relative to the tracking station. For a 300-naut-mile orbit and ϕ of about 35°, the spacecraft is 10° above the horizon (for a period of 270 sec). Assuming two tracking stations 180° apart, not in polar regions and 300-naut-mile orbits, the proposed storage period is 12 hr.

A tape storage system is proposed with a bandwidth of 1 kc/in./sec or 6280 rad/in./sec. Using a serial binary modulated suppressed high-frequency carrier and allowing 6.28 time constants per bit, storage density of 100 bits/in. is achieved. Two separate heads are used for record and playback, at rates of 0.256 and 40.96 ips, respectively. Single-channel recording is utilized. Up to 12 hr of recording at 256 bits/sec may be accomplished with a 6.25-in. reel of 8 mm $\times$ 1.5-mil Mylar base tape. Playback is at 40,960 bits/sec for 270 sec, the maximum allowable time with safety allowance for data acquisition from a 300-naut-mile orbit. For higher altitude orbits, longer time periods are available for data transmission.

Environmental Control

For purposes of calibration and control, it is useful to break down spacecraft-generated magnetic fields into three categories. First, many materials when exposed to a magnetic field retain some magnetism when removed from the field. This is called the permanent field. Second, when permeable materials (iron, cobalt, nickel, etc.) are placed in a magnetic field, they distort the field. This results in an additional field which is superimposed on the original field and is called the induced field. Third, electric currents have magnetic fields associated with them which are called current loop fields.

The spacecraft will contain many possible sources of these magnetic fields. The power-generating system, consisting of solar panel and battery, is immediately suspect because of large currents that will flow. The electric propulsion engines must be considered both from the standpoint of currents and use of magnetic materials. The basic bus, which might be defined as consisting of the structure, telemetry, and attitude-control system, will generate fields primarily because of presence of magnetic materials.

Calculation of fields due to the solar panels indicates that, if all panels were wired so that maximum field is produced on the axis of symmetry perpendicular to the plane containing the panels, the field would be about 300 gammas ($1\ \gamma = 10^{-5}$ gauss) at 15 ft and at full solar panel load. The panels can easily be wired to cancel fields due to adjacent solar cell module strings. It is estimated that this would reduce the field to below 10 γ, a negligible value.

Nearly nonmagnetic batteries have been developed for space use. The Surveyor spacecraft battery, which is similar to that proposed for this vehicle, has a permanent field

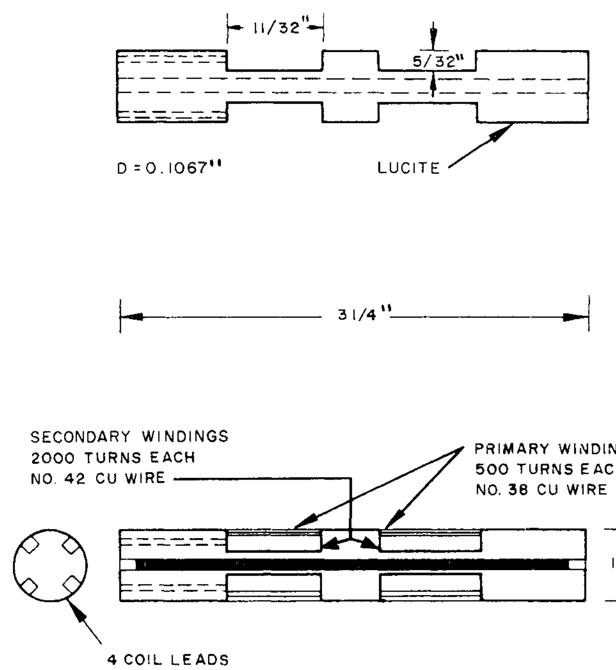

Fig. 5 Flux sensor components.

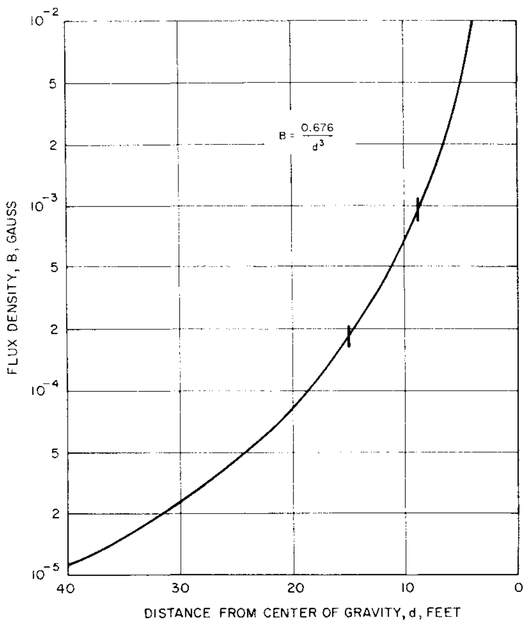

Fig. 6 Stray magnetic field strength of solar-electric test satellite.

(nonoperating) of less than 0.01 γ at 15 ft and produces about $\frac{1}{16}$ γ/amp of current drawn.

The spacecraft structure will be fabricated of nonmagnetic material and can be expected to produce no measurable field. Experience on other spacecraft has indicated that the fields (permanent, induced, and current loop) produced by electronics assemblies operating at the power levels expected here can easily be held to less than 1 γ at 15 ft. Fields due to the electric propulsion engines are not accurately known. The engines contain very small amounts of magnetic materials and can be shielded, at least partially.

Magnetometer Placement

Magnetometers are sensitive to stray magnetic fields within the ambient field. It is therefore necessary to place them at a distance from the major source of a stray magnetic field such that the field intensity is well below the sensitivity of the magnetometer. Ideally, the stray field variation could be estimated by the inverse cube law. The representative stray magnetic field associated with the proposed vehicle is shown graphically in Fig. 6. The errors in estimating the stray fields may be about 20%. Sensors should be mounted symmetrically to the total magnetic field of the spacecraft. This will make the simplest mounting in terms of the required structure and relationship to the ion engines. For general field measurements the threshold level of the instrument need not be less than 10^{-3} gauss. Figure 6 shows that placement distance of only 8.8 ft is required. If the threshold level is specified as 2×10^{-4} gauss (20 gammas), the placement distance becomes 15 ft. These are known stray field flux densities and can be calibrated.

Only the sensors will be placed at the required distance. The electronics equipment will be placed with the other electronics equipment near the center of gravity.