Ion Propulsion System for Stationary-Satellite Control

Jerome H. Molitor*
Hughes Research Laboratories, Malibu, Calif.

The application of ion propulsion to the 3-axes attitude control and station keeping of stationary satellites is discussed. The mission constraints that affect the engine system design, such as velocity increments associated with vernier orbit corrections, magnitude of disturbance torques, and required thrusting directions are presented. An evaluation is made of the tradeoffs between such critical parameters as attitude and station-keeping accuracy, average power utilization, duty cycle, thrust level, and satellite mass and moments of inertia. A preliminary design is given for the ion engine attitude-control and station-keeping system. (A prototype system has been developed and laboratory tested.) Such system parameters as power level, thrust level, specific impulse, and weight are specified.

Nomenclature

a_γ = tangential component of earth's gravitational field in the equatorial plane
a_s = radius of satellite orbit
D_c = thrustor duty cycle
I_0 = vehicle inertia
K^2 = earth's gravitational constant
P_a = average power available to thrustors
P_c = continuous power required
P_w = thrustor warmup power
t_{01} = attitude-control thrust interval
t_{02} = time between attitude-control thrust pulses
T = first-order earth triaxiality coefficient
T_c = control torque
T_d = disturbance torque
t_w = thrustor warmup interval
α = angular acceleration
γ = longitude of the satellite as measured from the Equator's minor axis
θ_d = attitude angular accuracy

Introduction

ONE of the first applications for ion engine systems will be the control of earth satellites, in particular, stationary satellites. To design the optimum propulsion system for this application, knowledge of both the mission characteristics and the space vehicle configuration is required in order to determine the pertinent mission constraints, such as velocity increments for vernier orbit correction, magnitudes of disturbance torques, and required thrusting directions. This information is also necessary to the evaluation of tradeoffs between such critical parameters as attitude and station-keeping accuracy, average power utilization, duty cycle, thrust level, length of thrusting interval, and satellite mass and moments of inertia.

This paper presents the results of a study which was undertaken to design an ion propulsion system and to demonstrate its capabilities for attitude control and station keeping of a satellite in a 24-hr-earth orbit. Since a mission of this type has not been defined in detail, certain satellite characteristics will be assumed.

Presented as Preprint 2666-62 at the ARS 17th Annual Meeting and Space Flight Exposition, Los Angeles, Calif., November 13–18, 1962; revision received January 16, 1964. The author wishes to acknowledge the valuable contributions of Roland A. Boucher, Marshall H. Kaplan, and Robert J. McElvain to the analyses presented in this paper.

* Staff Engineer, Ion Propulsion Department. Member AIAA.

Mission Characteristics

The attitude-control system must hold the satellite to specified accuracies in 3 axes in the presence of disturbance torques due to such natural effects as solar radiation pressure and micrometeorite impact, and such internal effects as gas leakage, moving parts, and thrust misalignment. It must be simple for the sake of reliability and yet versatile enough to counteract a range of unbalance torques. Although the disturbance torques are, in general, functions of the vehicle configuration which are rather difficult to predict accurately, estimates can be made of the unbalance due to solar pressure, which is the major cause of satellite rotation. For example, a satellite weighing 1000 lb could probably be constructed so that the unbalance torques would be on the order of 100 dyne-cm.[1]

As noted in the companion paper by Boucher,[2] the control system requirements are dictated primarily by the perturbing forces due to the triaxiality of the earth and the gravitational attraction of the sun and the moon. The former gives rise to longitude-dependent radial and tangential accelerations. The radial component causes only slight increase in the required altitude, but the tangential component causes an East-West drift. In the equatorial plane, the tangential component of the earth's gravitational field is[3]

$$a_\gamma = (2K^2T/a_s^4)\sin 2\gamma \tag{1}$$

The maximum tangential acceleration due to the earth's triaxiality is 2.22×10^{-7} ft/sec^2, corresponding to an increase in velocity of about 7 fps/yr, which must be imparted tangent to the satellite orbit by the longitude-keeping system.

The only significant effect of the sun and moon on a stationary earth satellite is a rotation of the satellite orbit out of the equatorial plane. This change in orbit inclination is caused by the components of the attraction of the sun and moon normal to the satellite orbit. These perturbing forces will cause a maximum inclination change in the satellite orbit of 0.948°/yr, as noted in Ref. 2. In order to counteract this solar-lunar perturbation, corrective thrust must be applied to the satellite in a direction normal to the orbit plane. Since the satellite orbit will rotate about an axis directed through and normal to the thrust vector, corrective thrust impulses can be applied most efficiently at the nodal points, that is, satellite crossings of the equatorial plane (as discussed in Ref. 4). It has been shown,[5] however, that this is not the optimum corrective thrust mode for an ion propulsion control system. By correcting continuously at low thrust levels, the ion engines can be run directly off the primary power source. The energy storage system required in an impulsive correction mode therefore can be eliminated. Using a continuous low-thrust corrective mode, the total velocity increment required to negate the solar and lunar perturbations is about 230 fps.

System Constraints

In order to provide 3-axes attitude control and station keeping of a 24-hr satellite, it is necessary that thrust be directed in at least nine directions (see Fig. 1). Attitude control of each axis is accomplished by thrusting in two opposite directions at the end of a moment arm, providing control over both positive and negative angular displacements. Both the North-South and East-West station keeping require a thrust vector through the center of gravity of the satellite. The latitudinal station keeping is provided by thrusting alternately in a North-South direction (depending on whether the satellite is approaching an ascending or descending node) perpendicular to the satellite orbit. The longitudinal correction is accomplished by thrusting in a single direction (either East or West, depending on the station longitude) once every 12 hr tangent to the orbit. For example, if the perturbing acceleration is eastward, the satellite will gain altitude and develop an apparent drift westward. The corrected thrust in this case is applied in a westward direction against the orbital motion of the satellite.

It has been shown[5] that a substantial weight saving could be afforded by operating the North-South station-keeping engines in a continuous thrust mode, and that an interruption in the North-South station keeping of 2 hr at the peak of the latitudinal oscillation did not noticeably reduce the effectiveness of the continuous correction mode. Since these peaks are 12 hr apart, an East-West correction can be made in these 2-hr intervals. The optimum mode of operation from a weight standpoint for an ion propulsion station-keeping system is, therefore, a semicontinuous North-South correction at a relatively low thrust level with the East-West correction occurring at each peak in the latitudinal oscillation. The corresponding thrust sequence is presented in Table 1.

It is possible to meet the nine-thrust-directions requirement either by using nine ion engines (one for each direction) or by using a lesser number gimballing for thrust vectoring or with electrostatic deflection of an ion beam. The choice must be made on the basis of the power and weight constraints necessarily imposed on an attitude-control system.

System Performance

The tradeoffs between the mission requirements, the vehicle configurations, and the engine system characteristics are somewhat unique for electric propulsion. Generally, for a mass expulsion system, thrust duty cycle (or propellant requirement) is the critical performance parameter. However, in the case of the high specific impulse electric propulsion system, the most critical performance parameter is the average power consumption. For this reason, all of the performance criteria discussed in this paper are related to the average power consumption.

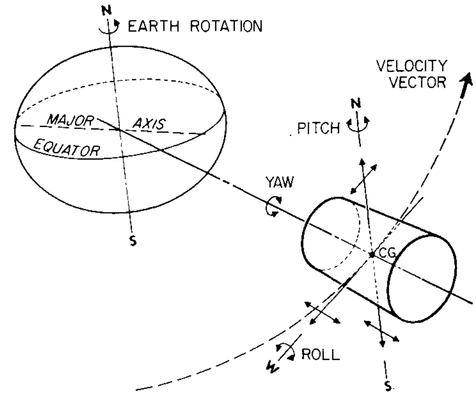

Fig. 1 Required thrusting directions for 3-axes attitude control and station keeping of a 24-hr satellite.

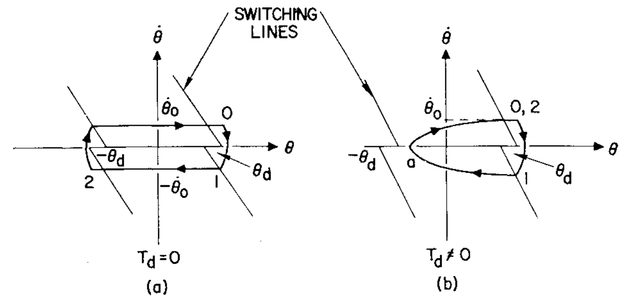

Fig. 2 Phase-plane diagram of system in limit-cycle operation.

Attitude Control

The principal tradeoffs involved in the attitude-control performance are those between accuracy desired, average power available, and fixed vehicle parameters such as moment of inertia and maximum expected disturbance torque. For this performance analysis, the following assumptions are made:

1) A single-axis analysis is applicable since cross-coupling torques are small. The results then can be applied to the 3-axes case by assuming 3 similar single-axis systems.
2) The thrust is essentially constant when applied.
3) The attitude references are initially acquired by an auxiliary method; hence the ion propulsion attitude-control system (ACS) will operate in a limit cycle or near-limit cycle mode during the vehicle lifetime.

The limit-cycle operation in the phase plane is shown in Fig. 2 for two disturbance torques T_d: 1) $T_d = 0$, hard limit cycle, and 2) $T_d \neq 0$, soft limit cycle. In a hard limit-cycle operation, thrust is applied at both sides of the prescribed angular limit, whereas the soft limit-cycle operation requires thrust only at one extremity. The latter situation occurs when the disturbance torque is large enough to decelerate the vehicle and reverse its direction before the opposite switching line (proper error signal) is reached. It can be shown that the system will go into a hard limit cycle when

$$0 \leqq T_d \leqq T_d(\max)/4$$

and a soft limit cycle when

$$T_d(\max)/4 < T_d \leqq T_d(\max)$$

where $T_d(\max)$ is the design maximum for the system.

The most stringent constraint governing the design of the attitude-control loop is the average power utilization requirement during the nonthrusting periods. Because of the unique nature of the cesium surface-contact ion engine (the only type considered here), power must be supplied to heat the ionizer for some period before each firing. (In general, since the duty cycle associated with an ACS is low, warming

Table 1 Semicontinuous mode correction cycle (24-hr period)

Engine	Thrust position	Thrust interval[a]
North	Symmetrical about ascending node	10 hr
East-West	Node plus 6 hr	18.2 min
South	Symmetrical about descending node	10 hr
East-West	Node plus 6 hr	18.2 min

[a] Thrust level-to-satellite mass ratio = 2.7×10^{-7} g's

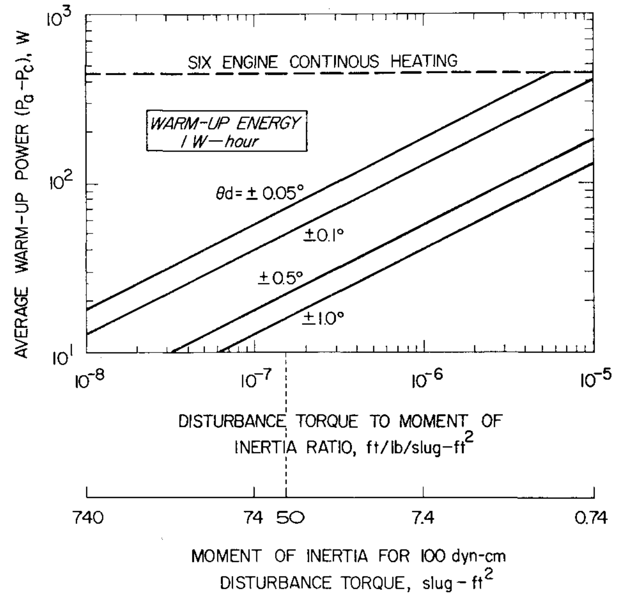

Fig. 3 Performance tradeoff for 3-axes attitude-control system.

the ionizer for each firing is more economical power-wise than continuous heating at the operating temperature.)

Since the average power available on a vehicle may be less than that required to heat the ionizer, a sufficient time must be allowed on the average between successive engine firings in order that this energy may be stored, for instance, in batteries. Hence the time from point 0 to point 2 for $T_d = 0$ must be

$$t_{02} \geq P_w t_w/(P_a - P_c) \quad (2)$$

This constraint on the coast time will then determine the minimum thrust time t_{01} which will allow a coast time of at least $P_w t_w/(P_a - P_c)$. For the case of $T_d \neq 0$, the relationship between coast time and the thrust time in the limit cycle is

$$t_{01}/t_{02} = T_d/T_c \quad (3)$$

provided that the limit cycle is "one-sided"; that is, the thrust is always applied in one direction as shown in Fig. 2. Then, from (2) and (3), the minimum thrust time must be

$$t_{01} = (T_d/T_c)[P_w t_w/(P_a - P_c)] \quad (4)$$

The achievable accuracy $\pm\theta_d$ is determined by the zero disturbance torque limit cycle once the minimum thrusting time has been chosen as a function of the maximum expected disturbance torque. For this case, the allowable attitude accuracy $\pm\theta_d$ must be at least

$$\theta_d = \alpha t_{12}^2/4 = [T_d/4I_0][P_w t_w/(P_a - P_c)]^2 \quad (5)$$

It is interesting to note that the control torque magnitude T_c is not a variable in the performance tradeoff, except that it determines the firing time t_{01} for a given disturbance torque (3).

The preceding discussion of design procedures and performance tradeoffs for a single-axis ion propulsion ACS may

Table 2 Three-axis performance parameters

$\theta_d = \pm 0.5°$, $P_a = 55$ w, $P_c = 25$ w, $T_c = 10^{-3}$ ft-lb, $P_w t_w = 1$ w-hr, and $t_{02} = 492$ sec			
T_d, dyne-cm	I_{min}, slug-ft^2	t_{01}, sec	D_c
100	50	3.7	0.011
150	75	5.6	0.022
300	150	11.1	0.110

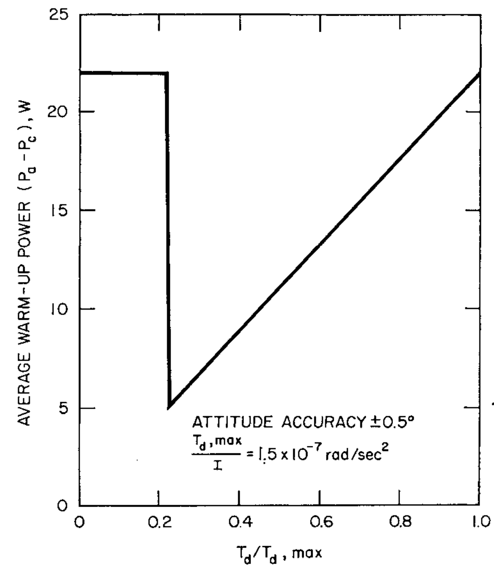

Fig. 4 Average power vs actual disturbance torque to design maximum disturbance torque ratio.

be extrapolated to obtain 3-axes performance characteristics, provided that cross coupling torques between the axes can be neglected. For this simplified 3-axes control case, the average coast time between successive firings may occur in different control axes. In the most conservative design, each axis is designed to accommodate the maximum disturbance torque, and the average time between firings for each axis must be at least $3t_{12}$. Equation (6) will then become for the 3-axes case:

$$\theta_d = [T_d/4I_0][3P_w t_w/(P_a - P_c)]^2 \quad (6)$$

Equation (6) may be utilized to determine the various performance tradeoffs for the 3-axes ion propulsion ACS. Figure 3 expresses these performance tradeoffs in a single family of curves. From this family of curves, the achievable accuracy may be determined if the vehicle moments of inertia, maximum expected disturbance torque, and average power available are known. Conversely, for a particular design objective of average power and attitude accuracy, the required vehicle parameters may be determined.

From the performance tradeoff curves of Fig. 3, the following fixed set of parameters are chosen to demonstrate the technique:

P_s (source output)	= 55 w
P_a (power conditioning output)	= 47 w
P_c (continuous power)	= 25 w
θ_d	= ±0.5°

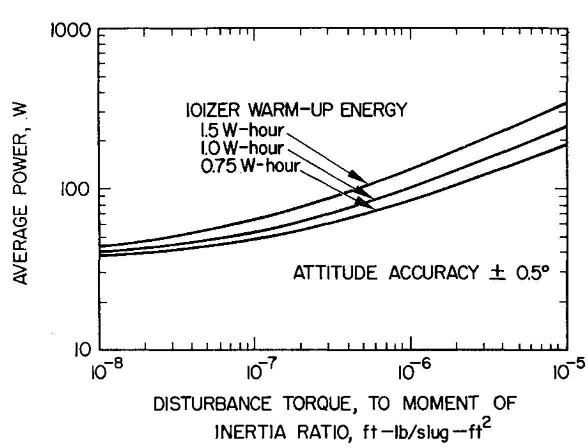

Fig. 5 Power requirement for 3-axes attitude control for various ionizer warmup energies.

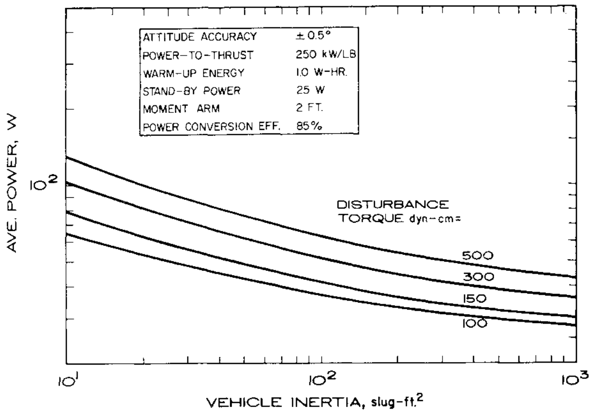

Fig. 6 Power requirement for 3-axes attitude control for various disturbance torques.

From Fig. 3 the ratio of T_d/I_0 is required to be 1.5×10^{-7} or less. In the presence of a 100 dyne-cm disturbance torque, an inertia of about 50 slug-ft^2 would be required to satisfy the average power requirements. The corresponding cycle time per axis is 8.2 min.

The limit on average power requires a coast time between successive firings of at least t_{12} for both hard and soft limit cycles. Although the propellant and average power consumption is greatest when the disturbance torque is either zero or the maximum design value, less power is required for intermediate values of torque. The average power consumed as a function of the disturbance torque is shown in Fig. 4.

The minimum thrusting time t_{01} in each axis for 3-axes control becomes

$$t_{01} = [T_d/T_c][3P_w t_w/(P_a - P_c)] \quad (7)$$

With control torque of 10^{-3} ft-lb, the disturbance torque of 100 dyne-cm, warmup energy of 1w-hr, average available power of 55 w, and continuous power of 25 w, the thrusting time is 3.7 sec. Since the control torque will be, in general, much greater than the disturbance torque, the 3-axes duty cycle D_c will be given by

$$D_c = 3t_{01}/t_{02} = 3T_d/T_c \quad (8)$$

With a control torque of 10^{-3} ft-lb, the 3-axes duty cycle for 100 dyne-cm disturbance torque is 0.022. The results of this performance analysis are summarized in Table 2 for three values of disturbance torque, T_d. The ACS performance is influenced primarily by the available average power during the coast period and the warmup power requirements of the proposed ion engine. The required vehicle inertia I_0 will increase as the square of the average coast time allowed between firings for a given value of θ_d. Since the average coast time is directly proportional to the warmup power requirements for the ion engine, this engine characteristic should be reduced to its lowest possible value. The resulting improvements in the average power and/or the vehicle moment of inertia requirements are shown in Fig. 5 for several values of the engine warmup energy. From Fig. 5 or Eq. (6), it can be seen that a reduction in engine warmup energy by a factor of 2 results in a decrease by a factor of 4 in the minimum inertia required, or a factor of 2 in the average power consumption.

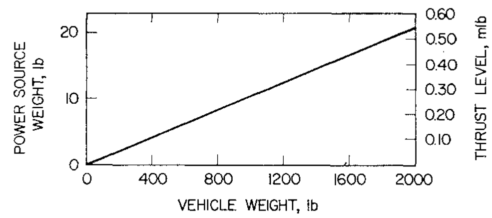

Fig. 7 Station-keeping power supply weight and required thrust level vs satellite weight.

Figure 6 shows the average power required by the ion engine attitude-control system for various values of disturbance torque (100, 150, 300, and 500 dyne-cm) as a function of vehicle moment of inertia.

Station Keeping

It was previously noted that the drift caused by the triaxiality of the earth of a satellite positioned at a point of maximum perturbation, can be negated by application of a 9.6×10^{-3} fps velocity correction every 12 hr in the westward direction. In the 12-hr period that the vehicle is allowed to drift, the satellite longitude will deviate only about 1 sec off the desired 24-hr station. The magnitude of the North-South oscillations of the satellite caused by the change in orbit inclination was seen to build up at a rate of 0.948°/yr. If these vehicle oscillations were left uncorrected for 37 days, it would reach a peak-to-peak value of 0.20° in latitude (0.1° inclination). Using the optimum correction mode, this North-South perturbation can be corrected by application of 0.63 fps velocity correction 24-hr period.

In the continuous correction mode, the weight of the satellite determines the required thrust level and the weight of the power supply. The effect of satellite weight on power supply weight and thrust level is shown in Fig. 7. Table 3 shows the station-keeping requirements imposed by 550-1000-, and 1500-lb stationary satellites on an ion engine control system. From the data presented in Fig. 7 and Tables 1 and 3, the design characteristics of the optimum station-keeping system (such as power supply requirements and weights, thrust level, duty cycle, and propellant weight), as well as the exact mode of operation, can be determined for any given satellite application. For example, three 0.3-mlb ion engines could maintain a 1000-lb satellite on station to less than 0.1° in both longitude and latitude for a 3-yr period with a total propellant requirement of less than 5 lb.

System Design

The system (Fig. 8), which shall be described, consists of 12 thrustors positioned at 4 stations, feed systems, engine control systems, power conditioning equipment, and control and logic networks. The stations are located on 2-ft moment arms at the extremities of the pitch and roll axes. Each of the control units contains three linear-strip 0.3-mlb ion thrustors, two for attitude control and the third for station keeping. In the attitude-control mode, onboard sensors will supply attitude error signals to each axis. These signals are quasi-continuous, so that simple shaping networks can

Table 3 Station-keeping requirements

Satellite mass, lb	Thrust level, mlb	Average power, w	Impulse per year, lb-sec $\times 10^{-3}$	Solar cells, lb	Propellant, lb/yr
550	0.15	75	4.1	9.6	0.91
1000	0.27	110	7.3	14	1.6
1500	0.41	149	11	19	2.5

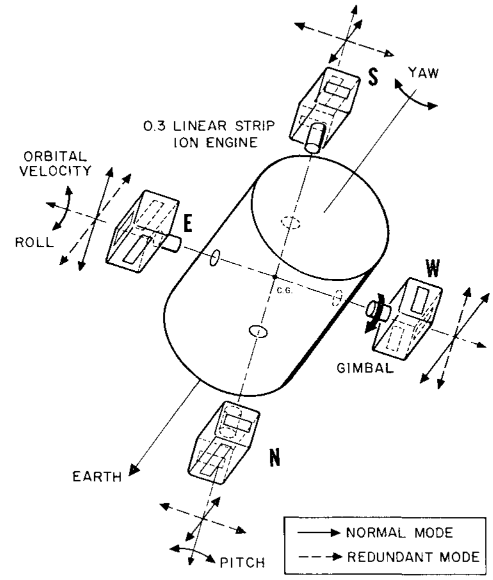

Fig. 8 Configuration of thrustors and engine stations for attitude-control and station-keeping system.

achieve the desired rate information to insure system stability. At a predetermined signal level, as defined by the attitude accuracy desired, the ionizers will be heated to temperature ($\sim$25 sec time constant), voltage applied to the electrodes, and the ion engine fired by opening a valve in the propellant feed system. After the proper firing interval, thrust is terminated instantaneously by voltage shutdown and valve closure. The station-keeping engines will operate in a similar manner but will be controlled by on-off commands from the ground.

Three of the stations are required to provide the required nine thrusting directions. The fourth station is incorporated into the system to provide thrustor redundancy and complete satellite position control. For example, this fourth station may be required to compensate for second-order effects such as translational accelerations due to the attitude-control pulses and solar pressure, or it may be desired to provide a station-seeking capability such as orbit eccentricity control. Furthermore, the control units are designed to mate with 90° rotators. Incorporation of these rotators into the system would provide complete thrustor redundancy for both attitude control and station keeping.

Engine Characteristics

For attitude control, the largest contribution to the average power requirement is the warming-up of the ionizer. The linear-strip ion engines, which are being developed at this time, will require about 1.0 w-hr of energy to heat the ionizer. However, recent data indicate that warmup energy for a 0.3-mlb linear-strip engine can be reduced to values as low as 0.75 w-hr. The effect of this improvement on system performance can be seen in Fig. 5. For the station-keeping mode, the power required to thrust is the major contribution to the average power requirement. For this reason, the power-to-thrust ratio of the ion engines used in this application must be reduced to the lowest possible value consistent with reliability.

Table 4

Basic control system weight	42 lb
Solar cell weight	14 lb
Battery weight	4 lb
Total system weight	60 lb
Propellant weight per year (including contingency)	$\sim$2 lb

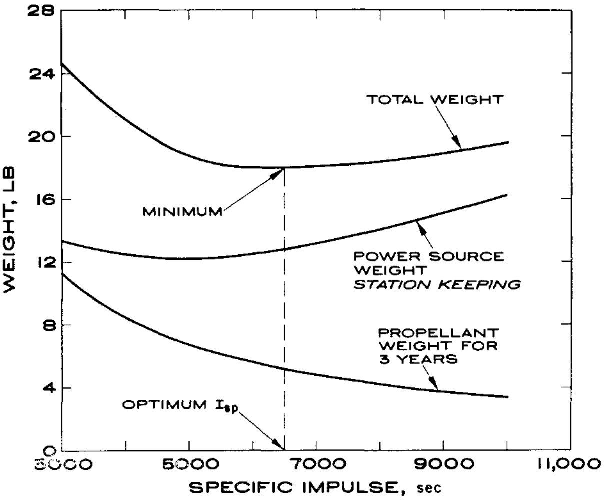

Fig. 9 System weights vs specific impulse.

The choice of thrust level for both the attitude-control and station-keeping engines is in a sense arbitrary. For the semicontinuous North-South correcting mode, however, the thrust level is determined by the weight of the satellite. In general, the lower the thrust level (for a given current density), the lower the ionizer warmup energy. For this reason, the optimum thrust level for both the attitude-control engine and the station-keeping engine is that dictated by the continuous thrust mode. These lower-thrust levels save valuable weight in the power conditioning equipment over higher-thrust devices.

The power requirement (hence power source weight) of an ion engine of specified thrust is dependent on specific impulse. It decreases to a minimum value from which it increases monotonically with specific impulse. The propellant weight varies inversely with specific impulse. Figure 9 shows the propellant weight and power source weight (station keeping only) required to maintain the orbit of a 1000-lb satellite as a function of specific impulse. The optimum specific impulse is found to be 6500 sec.

Power Source Characteristics

In the ion propulsion attitude-control and station-keeping system presently under development, it is planned that the station-keeping engines be run directly off primary source power, while batteries will provide power for the operation of the attitude-control engines. The total average power requirement for an ion propulsion attitude-control system is shown in Fig. 10 as a function of the ratio of disturbance torque to moment of inertia. These curves cover a range of attitude accuracy objectives (i.e., $\pm 0.05°$ to $\pm 1.0°$).

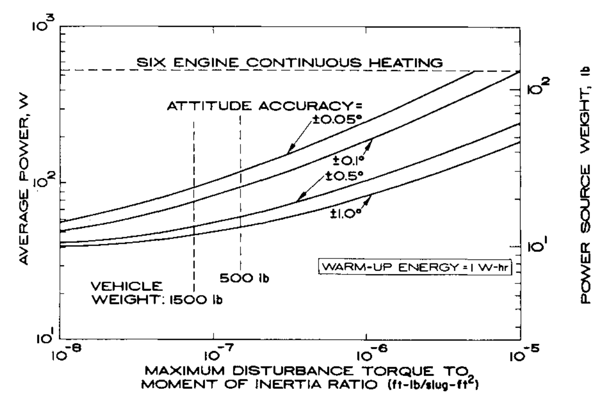

Fig. 10 Power requirement and power-supply weight for ion engine 3-axes control system.

The vertical dotted lines represent typical ratios of maximum torque to moment of inertia for satellites in the 500- to 1500-lb class. The weight of the power supply for an ion propulsion attitude-control system for such satellites will be less than 10 lb (Fig. 10). In the semicontinuous North-South mode of operation, the engines are operated directly from the solar cells so that no energy storage system is needed for the station-keeping system. However, about 30 w-hr of energy storage capability will be required to operate the attitude-control engines during the time the vehicle spends in the earth's shadow (a maximum of 1.16 hr/day). Typical attitude-control and station-keeping system weights required for a 1000-lb vehicle when using the continuous station-keeping thrust mode are shown in Table 4.

Conclusions

Three-axes attitude control and station keeping of a stationary satellite are well within the capabilities of ion propulsion systems. Some of the problems peculiar to this application of ion engines, such as the need for extremely low ionizer warmup energy, are already near solution, and others, such as engine reliability after thousands of thrust pulses, will be accomplished and demonstrated in the near future. A prototype system of the type described in this paper has been developed and laboratory tested. A comparison of the weights of several types of attitude-control and station-keeping systems (as discussed by Boucher[2]) has shown that as the desired lifetime of satellites extends to more than one year, the ion propulsion system emerges as the lightest attitude-control and station-keeping system presently under development.

References

[1] McElvain, R. J., "Effects of solar radiation pressure upon satellite attitude control," ARS Preprint 1918-61 (August 1961).

[2] Boucher, R. A., "Electrical propulsion for control of stationary satellites," J. Spacecraft Rockets 1, 164–169(1964).

[3] Ehricke, K. A., "Dynamics," *Space Flight* (D. Van Nostrand, Inc., Princeton, N. J., 1962), Vol. II, pp. 144–150.

[4] Jensen, J., Townsend, G. E., Kraft, J. D., and Kork, J., *Design Guide to Orbital Flight* (McGraw-Hill Book Co., Inc., N. Y., 1962), p. 138.

[5] Molitor, J. H. and Kaplan, M. H., "Optimization of ion engine control systems for synchronous satellites," AIAA Preprint 63-273 (June 1963).

MARCH-APRIL 1964 J. SPACECRAFT VOL. 1, NO. 2

Fluxgate Magnetometer for Space Application

S. C. Ling*
Therm, Inc., Ithaca, N. Y.

A new type of fluxgate magnetometer based on the principle of minimum flux linkage between the gating field and the ambient field is developed for space application wherein field strengths of the order of a few gammas are to be detected. Through the unique combination of several physical properties of the magnetometer, a stable decoupling factor of the order of 10^{-7} is achieved. A phenomenological theory is presented to explain the general mechanics of the fluxgate, and a discussion of the spectrum of transverse induction is made. Experimental values for a typical magnetometer and its circuitry are presented in order to demonstrate its working principle.

Introduction

An advanced type of fluxgate magnetometer was developed for space applications, in which magnetic field strengths of the order of a few gammas ($1\gamma = 10^{-5}$ gauss) are to be detected. In order to measure such weak fields, one must find a way to minimize effects of the disturbances due to the presence of the space vehicle and its associated controls and instruments. Realizing this fact, T. Gold urged the development of a magnetometer sufficiently small and sensitive to be attached to the end of a boom 20 to 30 ft away from the vehicle, so that the disturbing magnetic field would be reduced to an acceptable level according to the inverse-distance-cubed law. This concept led to the development of an improved fluxgate magnetometer, which is described in this paper.

In general, a fluxgate magnetometer works on the principle of gating the ambient field by modulating the permeability of a piece of magnetic material through the application of an alternating gating field. The resultant modulated ambient field is an even function of the gating field, and the voltage induced in a signal coil which links these fluctuating flux fields contains the fundamental frequency of the gating field and the even harmonics of the gated ambient field. In order to detect an ambient field of the order of 0.1γ a stable rejection factor for the fundamental frequency of the order of 10^{-9} would be needed, because the gating field density for most magnetic materials is of the order of $10^8\gamma$. The standard technique is to employ the method of direct bucking

Presented as Preprint 63-187 at the AIAA Summer Meeting, Los Angeles, Calif., June 17–20, 1963; revision received January 6, 1964.

* Head, Experimental Section; now Research Associate Professor, Department of Space Sciences and Applied Physics, Catholic University of America, Washington, D. C. This work was carried out on behalf of NASA, under contract NASr-46, with the Center for Radiophysics and Space Research of Cornell University. The author wishes to thank T. Gold for his general guidance on this project and H. S. Tan for his invaluable advice and inspiration. Acknowledgment is also due to H. Eckelmann for his assistance in experimental investigations and circuit design.