Attitude Control and Station Keeping of a Communication Satellite in a 24-Hour Orbit

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Propulsion requirements for maintaining the attitude and location of a communication satellite with a two-year operational life in a synchronous equatorial orbit are examined. In the case of a 1000-lb satellite, with prescribed angular limits in attitude (±3°) and longitude (±10°), the required total impulse is estimated to be 2500 lb-sec. The need for numerous low-thrust pulses of short duration makes conventional chemical propulsion impractical. Valve and pressure regulator problems and storage tank weight make the use of compressed gas propellants unattractive. Electrical propulsion with ion engines appears to provide a system with minimum over-all weight (60 lb) capable of accomplishing the task. However, on the basis of functional simplicity and probable reliability, it is concluded that a low-pressure vapor jet system employing a volatile liquid or a subliming solid is most likely to deliver the desired performance.

System Parameters

THE choice of means for controlling the position of an object in space involves consideration of the following factors: duration of the mission; required degree of control; kinds and magnitudes of perturbations; control force requirements; system configuration; mass, momentum, and energy requirements; propulsion system characteristics; reliability; compatibility.

These factors are examined in relation to station keeping and the attitude control of an experimental communications satellite in an equatorial 24-hr circular orbit with a life expectancy of two years. It is desired to hold the satellite on a meridian about midway between the United States and England (±10° of 40° longitude west) and to maintain a prescribed angular attitude in roll, pitch, and yaw within ±3° relative to earth-centered coordinates.

Perturbations

Upon being put into orbit at the desired location, the principal influence tending to make the satellite drift off station is a horizontal component of gravitational acceleration attributable to a slight dipole distribution of the earth's mass relative to the equatorial plane.[1] Analytically, it is expressed by the following:

$$g_h = -\frac{3G(I_2 - I_1)}{2r^4} \sin 2\theta$$

where

g_h = horizontal component of acceleration, ft/sec^2
G = universal gravitational constant, 1.068×10^{-9} ft^3/lb-sec^2
$I_2 - I_1$ = difference of principal moments of inertia in the equatorial plane, 1.22×10^{35} lb-ft^2
r = orbit radius, 1.38×10^8 ft
θ = angular position relative to earth's major axis in the equatorial plane

The value of θ corresponding to the desired station is about 70°. The resulting value of g_h is about 3.4×10^{-7} ft/sec^2 toward the east. If one lets the satellite mass be 1000 lb, the equivalent eastward force is about 10^{-5} lb or nearly 5 dynes.

The peculiar dynamics of orbital motion result in a westward drift of the satellite consequent to the action of a force tending to accelerate the satellite eastward. Starting from rest at the eastern boundary of the station, the satellite will drift 20° longitudinally to the western boundary of the station in about 200 days. This gives some idea of the minimum frequency with which station keeping adjustment must be made.

The gravitational accelerations of the sun and moon are several orders of magnitude larger than the horizontal component of the earth's field, but when averaged over a complete orbital period their net effects are much smaller than the cumulative effect of the earth's gravitational dipole moment in the equatorial plane. The long-term solar and lunar gravitational perturbations result in tilting the orbital plane of the satellite relative to the equatorial plane with an average angular rate of about 1 deg/yr. In the present case this represents no problem because the permissible displacement in latitude is considerably larger than the cumulative precession of the orbit in two years.

The principal influence acting to torque the satellite in respect to angular attitude is solar radiation pressure. Its magnitude is dependent upon the reflectivity and orientation of exposed surfaces, being at most about 2×10^{-7} psf at normal incidence on a specularly reflecting surface. With reasonable care in designing the satellite, it is estimated that the torque attributable to solar radiation pressure can be kept below 10^{-5} lb-ft (about 140 dyne-cm). It is possible that the outgassing of various components of the satellite could generate, for a while, perturbations larger than those attributable to solar radiation pressure, but proper selection, preparation, and disposition of materials of which the satellite is made will insure that outgassing effects will decline to negligible magnitudes in a few days.

Control Forces

The radial gradient of the earth's gravitational field is too small and the magnetic field is too weak, erratic, and not known to be of much use for stabilizing objects in a 24-hr circular orbit. Presumably by suitably deploying large reflective surfaces one could use solar radiation pressure, but the mechanical problems of stowing, unfurling, and manipu-

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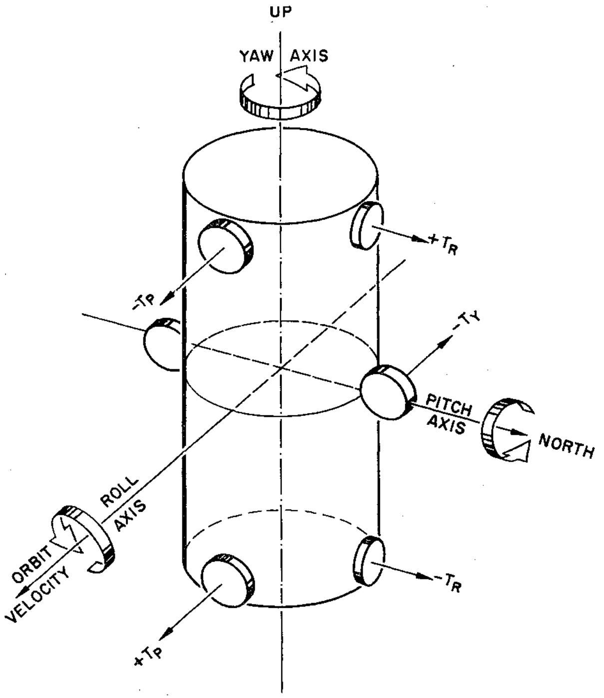

Fig. 1 Arrangement of thrust generators for three-axis attitude control and station keeping

lating such surfaces in the requisite manner make this approach unattractive. Some form of rocket propulsion is indicated.

To provide adequate margin for rapid recovery from transients following injection into orbit and accidental disturbances such as meteoroids,[2] the torque-generating capability of the attitude control system should be somewhat larger than the anticipated perturbations, but not so large as to cut the duty cycle down to the point where excessive precision is required to meter out the desired amounts of impulse. In the present case, a thrust of 2×10^{-4} lb (about 90 dynes) acting on a 3-ft moment arm relative to the satellite center of mass was chosen. This provides a torque about 60 times greater than the estimated perturbation from solar radiation pressure. The same thrust applied for translational purposes is 20 times larger than the horizontal component of force attributable to gravity.

Propulsion System Configuration

Without yet specifying the manner of generating thrust, one can imagine propulsive devices distributed about the satellite somewhat as shown in Fig. 1. The satellite is represented schematically as a cylinder aligned axially with a radius from the earth. Propulsive devices are shown as small cylindrical protuberances with arrows indicating the direction of propellant efflux. T_P, T_R, and T_Y designate thrusts relative to the pitch, roll, and yaw axes, respectively.

It is apparent that, by firing a pair of motors alternately, for example, the pitch-control motors, an oscillation relative to the pitch axis can be maintained within any specified angular limits. At the same time a translational thrust in the westward direction is obtained. Similarly, the yaw-control motors may be used to provide a translational thrust in the eastward direction. The net impulse in either direction can be varied by suitably controlling the duty cycles of the motors. The roll-control thrust is normal to the plane of the orbit and thereby permits some adjustment of the plane of the orbit if properly phased with the orbital period, but it is doubtful that this capability will be large enough to be of much use.

Mass, Momentum, and Power

With reference to station keeping, the cumulative impulse of 10^{-5} lb acting continuously for two years amounts to 630 lb-sec, equivalent to about 2 lb of conventional chemical rocket propellant. The power implicit in rocket propulsion is proportional to the product of thrust and specific impulse. Even for a specific impulse of 10,000 sec, characteristic of electric propulsion, the theoretical power requirement to produce a thrust of 2×10^{-4} lb is only 44 w. Thus, one sees that the mass, momentum, and power requirements for station keeping are theoretically rather small.

If one assumes the moment of inertia of the satellite about any axis is of the order of 5000 lb-ft^2 and lets the period for a complete oscillation with 3° amplitude be 25 min, one finds that each attitude control motor is active only about 5% of the time. Hence, the total impulse requirement for attitude control about each axis is roughly the same as for station keeping.

Thrust Generators

One has now to select a mode of generating thrust for the task at hand on the basis of capability, compatibility with other functions of the satellite, and reliability. Chemically energized propellants have been mentioned, but they are rated as poor prospects for several reasons: first, because of propellant storage problems connected with very long-term operations in space; second, because of the difficulty of metering out chemically reactive materials in amounts and at rates providing precise control of tiny increments of impulse (0.01 lb-sec) at very low thrust levels; and third, because of grave doubts concerning the feasibility of designing reliable restart capability for many thousands of cycles in a chemical rocket.

Cold Gas Propulsion

The requisite total impulse could be stored in the form of a compressed gas to be released at ambient temperature. If one assumes the following weights for component hardware

Propellant storage tank	0.005 (lb/ft^3)/psi
Pressure regulator	3 lb
Six control valves with nozzles	6 lb
Plumbing	3 lb

and specifies a total impulse of 5000 lb-sec, which is about twice the estimated two-year minimum requirement for three-axis attitude control and station keeping, one obtains the results shown in Table 1 representing four possible cold gas propellants.

Ammonia is the best of the four because of a happy combination of high storage density, low storage pressure, and relatively high specific impulse. About 1 w of thermal energy is needed to vaporize ammonia at a rate sufficient to provide a thrust of 2×10^{-4} lb. It seems likely that the propellant tank could pick up this amount of heat from the surroundings without developing an intolerably large reduction in temperature. The thermal input requirements for the other gases are much less than for ammonia. However, leakage problems in storing a compressed gas and controlling its flow with a pressure regulator and valves for a period of

Table 1 Cold gas propulsion systems

	H_2	NH_3	N_2	CO_2
Specific impulse at 60°F, sec	260	100	70	62
Storage pressure, psi	3600	110	3600	750
Propellant density,[a] lb/ft^3	1.26	38.5	17.6	51
Total system weight, lb	306	63	156	99

[a] At 60°F and storage pressure.

two years are sufficiently serious to recommend other approaches.

Presumably, one could circumvent the propellant storage problem by using photon propulsion, i.e., directed emission of electromagnetic radiation, but power amounting to 300 kw is required to produce a thrust of 100 dynes by this means. More reasonable power levels are found in connection with other electrical propulsion techniques.

Electrical Propulsion

The three main categories of electric rockets are electrothermal gas jets, magnetodynamic plasma accelerators, and electrostatic ion accelerators. The first group includes both convection heated and electric arc heated gas jets. The second can be subdivided into pulsed magnetic and steady-state crossed-field electromagnetic plasma pushers. The last encompasses a variety of forms that differ primarily in the manner of producing ions but are basically similar in the technique of electrostatically accelerating the ions into a directed stream of high-velocity particles.

Electrothermal Gas Jets

A recent article by Jack[3] reviews the subject of electrothermal gas jets and concludes that the three most efficient propellants in this application are the three lightest atoms of the periodic table: hydrogen, helium, and lithium. The first two are obviously poor prospects in the present instance because of high-pressure storage and flow-control problems, whereas lithium requires inconveniently high temperatures for storage and flow control even when treated as a very low-pressure propellant (melting point, 186°C; vapor pressure, 1 mm Hg at 723°C).

Present activity in arc-jet propulsion is directed toward thrust and power levels substantially larger than can be applied readily here.[4–9] This is understandably so because the ratio of surface to volume in the high-temperature zone is a major parameter determining the relative magnitude of thermal power loss in comparison with energy usefully employed in generating thrust. At best, it is difficult to achieve an efficiency of 50%. Consequently, the lower bound of present operable arc-jet thrustors is in the vicinity of 1 kw of power and 10^{-2} lb of thrust, about two orders of magnitude too large for the present purposes.

Magnetodynamic Plasma Accelerators

The same factors that rule out most electrothermal gas jets apply also to magnetodynamic plasma accelerators. The thrust and power levels tend to be too great, and propellants requiring high-pressure storage are used.[10–14] A possible exception is an exploding-wire plasma accelerator reported by Starr.[15] It seems plausible that a simple mechanism could be devised to feed propellant, in the form of a fine wire, into the electric discharge zone and fire the device intermittently at whatever repetition rate is needed to get the desired average thrust. An adverse feature of this mode of propulsion which would apply to all pulse-type motors is a possible incompatibility with the primary function of the communication satellite. The extremely large power surges associated with pulsed electric thrust generators radiate disturbances that probably will be very difficult to keep out of the communication channels. The entire satellite will ring like an electromagnetic bell every time the motor is fired.

Electrostatic Ion Accelerators

Two factors contribute to making ion rockets likely prospects in the present case. First, they are adaptable to low-thrust, high-efficiency operation; and second, they employ low-pressure propellants. In the conversion of electrical energy to ion beam kinetic energy, an efficiency of 80% is within the capability of a 100-dyne ion motor. This is about one order of magnitude better than can be expected of arc-jet and magnetoplasma propulsors at this thrust level. The propellants for ion motors (cesium and mercury) are stored and used at pressures considerably below 1 mm Hg.

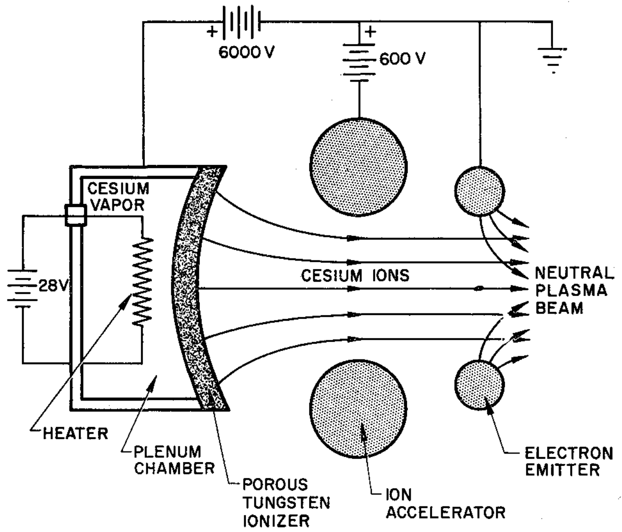

Fig. 2 Cesium ion rocket motor employing the feed-through technique with a porous tungsten ionizer

So far as space-flight hardware is concerned, ion rocket motors have achieved an advanced stage of development.[16–19] Hughes Research Laboratory, Electro-Optical Systems, Inc., and NASA Lewis Research Center now are preparing ion rocket motors scheduled for space-flight tests in 1963.

The three main competitors among ion rockets for the present application are 1) the cesium feed-through, porous tungsten contact ionization motor; 2) the cesium reverse-feed, solid tungsten contact ionization motor; and 3) the electron-bombardment mercury ion motor. Schematic representations of the three types of ion motors are shown in Figs. 2–4.

As its title implies, the feed-through type (Fig. 2) diffuses cesium vapor through a porous tungsten plate onto the ionizing surface from which the ions are driven by thermal agitation into the electrical field of the accelerator. Although this basic design has received most of the government-sponsored development effort, it has the following inherent problems:

1) To minimize the loss of neutral cesium atoms from the pore openings and to obtain a high flux of cesium ions from the tungsten surface, it is necessary to use extremely small pore sizes (about 1 μ) with very close spacing of the pore openings.[20, 21] Aside from the difficulty of obtaining the desired uniformity of material properties, present experience indicates that the permeability to vapor diffusion deteriorates with age, presumably because of continued sintering and plugging of the tungsten matrix at the temperature required for adequate ion flux from the ionizing surface (about 1600° K).

2) The microscopic roughness of porous tungsten increases its thermal emissivity such that about half of the total power loss is attributable to thermal radiation from the surface of the ionizer.

The reverse-feed technique, shown in Fig. 3, represents an attempt to circumvent the main problems of the feed-through design. There are good reasons for expecting the uniformity of ion flux and the proportion of cesium atoms that become ionized to be superior for a solid ionizer than for a porous ionizer. Moreover, the thermal emissivity of a highly polished solid tungsten surface is about a factor of 3 less than for a porous surface.

The reverse-feed design has problems of its own. The counter-current flow of neutral atoms and ions involves collisions and charge exchange resulting in the deflection and creation of ions in regions where the electric field cannot focus them into a beam. The stray ions are drawn into the accelerator electrodes, resulting in physical damage to the electrodes by a process known as sputtering. By operating with sufficiently low particle flux densities, the production of stray ions and resultant sputtering rate can be reduced to any desired level. Theoretical analysis indicates that good performance can be obtained from the reverse-feed design with tolerable electrode sputtering, and this has been demonstrated experimentally. It should be realized that the through-feed porous ionizer also is subject to sputtering effects from stray ions, since a significant fraction of cesium escapes from the porous surface without becoming ionized, and these neutral atoms diffuse into the ion stream, where they generate havoc in the same manner as in the reverse-feed design.

Figure 4 shows the electron bombardment mode of ionization employed in the mercury ion motor under development at NASA Lewis Research Center.[18] Mercury vapor is ionized in the bombardment chamber by collisions with radially accelerated electrons from an emitter at the center of the chamber. The efficiency of the bombardment process is enhanced greatly by an axial magnetic field that forces the electrons to travel in curved paths that can terminate on the wall of the chamber only by a series of collisions.

Although it is doubtful that the bombardment ionization process can achieve as high a percentage of ionization of the propellant as can be obtained with the contact ionization technique (theoretically the latter can yield better than 98% ions), the overall efficiency in terms of energy required to produce an ion seems better for the bombardment process than for the contact technique because the former does not require a large high-temperature surface for the generation of ions.

Application of Ion Propulsion to Satellite Control

The magnitude of thrust proposed for the present study is considerably smaller than the objectives of current experimental efforts in ion propulsion. Consequently, it is difficult to obtain performance data that can be scaled down with confidence to the magnitudes of thrust, power, and mass needed here. NASA Lewis Research Center was most helpful in supplying information concerning the feed-through cesium ion and electron-bombardment mercury ion motors, whereas an experimental study of the reverse-feed cesium ion motor at Lockheed provided a basis for evaluating that type of motor.

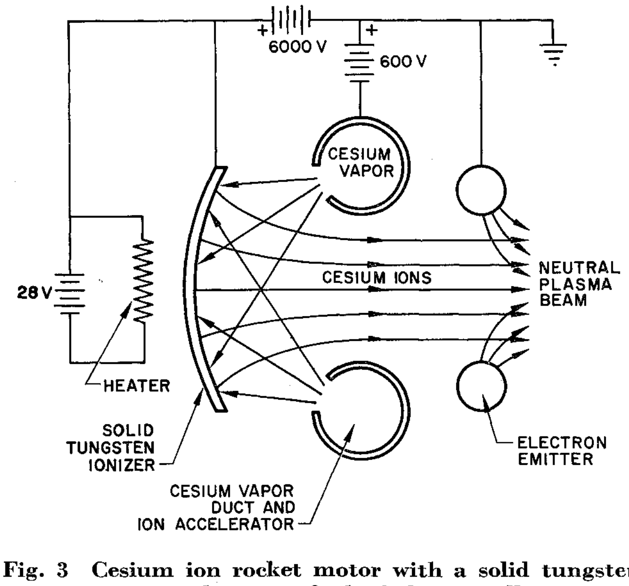

Fig. 3 Cesium ion rocket motor with a solid tungsten ionizer and reverse-feed of the propellant

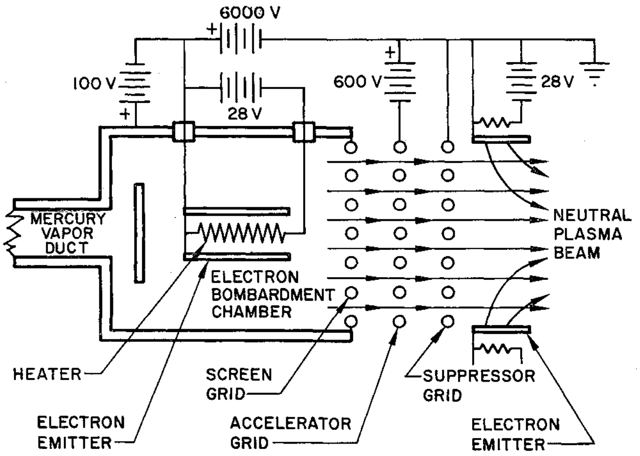

Fig. 4 Mercury ion rocket motor with ionization by electron bombardment

The system is presumed to comprise six identical motors as indicated in Fig. 1. To minimize the peak power demand, no more than one motor is required to deliver thrust at any given time, i.e., power is distributed to the motors on a time-sharing basis.

With the very high specific impulse available from ion motors, the propellant demand is so small that it is proposed to let it flow all the time. The total propellant requirement is based upon a thrust capability of 2×10^{-4} lb for each motor with an assumed propellant utilization efficiency of 90% (during thrust) for the cesium ion motors and an 80% propellant utilization efficiency for the mercury ion motor.

The inclusion of means for interrupting propellant flow is regarded as a greater loss of reliability than the saving of propellant weight is worth. To prevent deposition of propellant on insulators and other critical components of the ion motors during the time they are not thrusting, it is proposed to keep them in a standby status by leaving the heaters on all the time and modulate the thrust by turning the accelerator potential on and off. Hence, the total power requirement of the propulsion system consists of standby power for five motors plus full power for one motor.

Electrical power is obtained from solar energy converters (silicon barrier cells) with a specific power of 4 w/lb, assuming a nonoriented solar energy collection system, i.e., an array of solar cells disposed to receive sunlight from any direction. Some degree of voltage regulation is provided by nickel-cadmium storage batteries (28 v) with a specific energy capacity of 10 w-hr/lb and a charge-discharge cycle efficiency of 80%. The storage battery requirements are dictated primarily by maintenance of control during daily one-hour passages through the earth's shadow at certain seasons of the year (spring and autumn equinoxes). A portion of the electrical power is raised to the high voltages required for ion propulsion by means of d.c. chopper-rectifier techniques using semiconductor devices with an efficiency of 85% and a specific power handling capacity of 50 w/lb. To provide margin for contingencies, the power supply is designed with capacity sufficient to sustain a continuous thrust of 2×10^{-4} lb, although normal steady-state maintenance of the satellite should require such a thrust only about 35% of the time.

The foregoing considerations are incorporated in Tables 2–4, representing the performance capabilities of the three types of ion motors. The results in terms of total weight of the propulsion system vs specific impulse are shown in Fig. 5. All three types of ion motors indicate minimum system weights for a specific impulse in the vicinity of 9000 sec. For the electron-bombardment mercury ion propulsion system, the minimum total weight is 108 lb, whereas for the reverse-feed solid tungsten and through-feed porous tungsten cesium

Table 2 Cesium ion propulsion system (porous tungsten feed-through type)

Specific impulse, sec	Low voltage power,[a] w	High voltage power, w	Solar cells, lb	Storage batteries, lb	Propellant weight, lb	Total weight,[b] lb
5,000	380	22	106	41	17	173
6,000	212	26	64	24	14	111
7,000	120	31	41	16	12	78
8,000	78	35	31	12	11	62
9,000	64	39	29	11	10	57
10,000	63	44	30	12	9	59

[a] Standby power for six motors.
[b] Including 10% of propellant weight for storage tanks, 1 lb for voltage converter, and 6 lb for ion motors.

Table 3 Cesium ion propulsion system (solid tungsten reverse-feed type)

Specific impulse, sec	Low voltage power,[a] w	High voltage power, w	Solar cells, lb	Storage batteries, lb	Propellant weight, lb	Total weight,[b] lb
4,000	139	18	42	16	21	88
5,000	123	22	39	15	17	80
6,000	113	26	38	14	14	74
7,000	105	31	37	14	12	71
8,000	100	35	37	14	11	70
9,000	95	39	37	14	10	69
10,000	92	44	38	14	9	69

[a] Standby power for six motors.
[b] Including 10% of propellant weight for storage tanks, 1 lb for voltage converter, and 6 lb for ion motors.

Table 4 Mercury ion propulsion system (electron bombardment type)

Specific impulse, sec	Bombard-ment power,[a] w	High voltage power,[b] w	Solar cells, lb	Storage batteries, lb	Propellant weight, lb	Total weight,[c] lb
4,000	22	18	58	22	23	118
5,000	18	22	58	22	19	113
6,000	15	26	59	22	16	110
7,000	13	31	59	23	14	109
8,000	11	35	60	23	12	108
9,000	10	39	61	23	11	108
10,000	9	44	62	24	10	109

[a] About 100 v.
[b] Low voltage power is 175 w based upon half-power for electron emitters during standby.
[c] Including 5% of propellant weight for storage tanks, 1 lb for voltage converter, and 12 lb for ion motors.

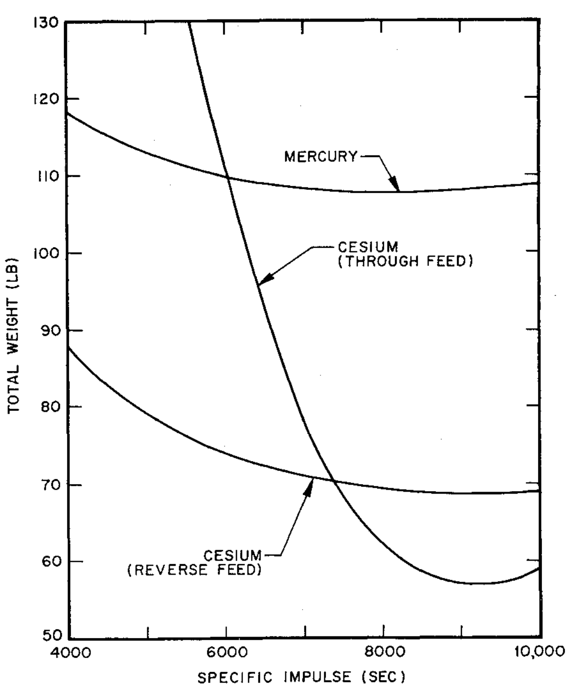

Fig. 5 Dependence of propulsion system weight upon specific impulse for three types of ion motors

ion systems, the minimum total weights are 69 and 57 lb, respectively.

The rapid increase in total system weight at the lower values of specific impulse for the porous tungsten through-feed cesium ion motor in comparison with the solid tungsten reverse-feed motor presumably is associated with increased thermal radiation losses from the porous tungsten surface. The thrust per unit area of ionizer surface declines with decreasing specific impulse, thus requiring an increase in thermally radiating ionizer surface area in order to maintain the desired thrust. The lower thermal emissivity of solid tungsten makes the performance of the reverse-feed ion motor less sensitive to specific impulse.

The relatively large weight of the mercury ion system compared to the cesium ion systems is primarily a reflection of the fact that the electron bombardment ionization technique does not scale down efficiently to small sizes as well as does the contact ionization technique. At larger thrust and power levels, it can be shown that the mercury ion motor is more competitive with cesium ion motors than in the present case.

Reliability

Although much progress has been achieved in recent years toward realizing prototype ion rocket motors for space applications, they still must be regarded as laboratory curiosities since, at the time of this report, none have been tested in space. There is no experience upon which to base a trustworthy estimate of life expectancy under actual space environmental conditions. Electrode erosion by sputtering with stray ions is one of the more serious factors limiting the useful life of ion motors. The high voltages required to accelerate the ions are a problem in regard to the reliability of devices for converting low-voltage d.c. power to high voltages. A specific impulse of 9000 sec requires about 5400 v for cesium ions and about 8100 v for mercury ions. The long-term behavior of insulators and solid-state rectifiers at such voltages under space conditions is unknown.

It would appear, therefore, that alleviation of the propellant flow control problem by the use of a very low vapor pressure propellant represents a questionable gain in reliability purchased at great expense in the form of complex electronic problems.

Compatibility

The location of the solar energy converters relative to the ion motors is a matter of great concern because it is quite certain that the impact of high-velocity propellant ions on the active surface of solar voltaic cells will damage the cells. In this respect, it is probable that stray ions that have not been directed properly into the main stream of ions will be troublesome.

The plasma beams from the ion motors may interfere with communication between the satellite and ground stations in at least two ways: first, by generating electromagnetic noise, and second, by acting as virtual antennas of indefinite size. The beam of ions and electrons emitted by an ion motor is by no means a quiescent plasma. Presumably, plasma oscillations with frequencies extending to kilomegacycles per second could be present, though there are no data yet as to the power spectrum of electromagnetic noise from an ion beam. The plasma beam can be modulated by impressed potentials so as to propagate signals much like an electrical conductor. It is not entirely clear what effect this may have on the reception and transmission of signals by the satellite. It is anticipated that results from space-flight tests of electrical propulsion devices proposed by NASA will provide answers to some of these questions.

Other Possibilities

The foregoing discussion of ion propulsion is not particularly encouraging in the present application. In retrospect, the simplicity of cold gas propulsion looks attractive. To alleviate the problems of high-pressure propellant storage, consider the use of rather low vapor pressure substances, such as water or naphthalene. At ambient temperature (60°F), the vapor pressure of water is about $\frac{1}{4}$ psi, which should be easy to control with a simple, reliable valve. The specific impulse of water vapor at this temperature is 100 sec. Hence, the performance should be very nearly like that of ammonia, which was cited earlier in this paper.

The use of water as a propellant permits some reduction in system weight compared to the ammonia propulsion system because water requires neither high-pressure storage tanks nor a pressure regulator. The flow control valves and nozzles need not weigh more than a few ounces, since the hydrostatic forces are trivial and the nozzle dimensions are small (throat diameter = 0.025 in., exit diameter = 0.25 in., length = 0.50 in.). Only 2.2 w are required to vaporize water at a rate sufficient to produce the specified thrust. A magnetically actuated valve can be designed to operate on less than 2 w. Hence, only about 1 lb of solar voltaic cells is needed to provide requisite power for flow and temperature control of the propellant.

It is desirable to avoid choking the valves with liquid water. Small electrical heaters, about 0.1 w each, will suffice to keep these components slightly warmer than the rest of the propellant system, thus tending to drive liquid water away from these parts. Probably an absorbant material like cotton will serve to hold the liquid water together as a coherent mass in the storage tank and prevent vapor bubbles from driving slugs of liquid water into the propellant distribution ducts.

Retention of liquid water in the storage tanks, although providing for admission of only water vapor to the valves and nozzles, may prove sufficiently troublesome to recommend consideration of a volatile solid propellant that can be restrained mechanically to a given location more easily than a liquid. Naphthalene is a possible candidate. At 60°F its specific impulse is 37 sec, so for a given total impulse the required mass will be almost three times larger for naphthalene than for water. The vapor pressure of naphthalene at 60° F is only about 0.03 mm Hg, which is inconveniently small for present purposes. However, at 140°F the vapor pressure is about 1.0 mm Hg, which permits the use of vapor ducts and nozzles of more moderate size than would be required at the lower pressure (nozzle throat diameter = 0.10 in., exit diameter = 1.0 in., length = 2.0 in.).

The heat of vaporization (sublimation) of naphthalene is 133 cal/g, whereas for water at 60°F it is 587 cal/g. Thus, in spite of the greater mass flow of naphthalene needed to produce the specified thrust, the thermal power needed to vaporize the propellant is only 1.25 w or about half the requirement for water. To store naphthalene at 140°F involves an additional amount of power depending upon how well the storage tank is thermally insulated from the surroundings. If one assumes that the rate of heat loss can be reduced to 1.0% of blackbody radiation at 140°F, one finds that the additional power input to the propellant storage tank is about 5 w. A few tenths of a watt of electrical power are needed to keep the valves warm enough to prevent condensation of solid naphthalene on the moving parts so as to avoid interference with their operation.

The electrical power required for low vapor pressure propellants is so small that there is no great need to be extremely conservative in its use. Hence, instead of restricting the peak power to the operation of only one motor at a time, as was done in the ion motor systems, one could very well simplify the control system by avoiding the power time-sharing feature and providing sufficient power to operate half of the motors simultaneously so that all degrees of freedom in station keeping and attitude can be controlled independently of one another. The penalty in additional system weight amounts to only a few pounds.

The results presented in Table 5 are based upon the foregoing considerations. Note that water does, indeed, yield a system weight almost identical to that for ammonia, whereas the rather poor specific impulse of naphthalene results in more than doubling the system weight.

Conclusions

1) Propulsion system forces and torques adequate for station keeping and attitude control of a communication satellite in a

Table 5 Low vapor pressure cold gas propulsion

	Water	Naphthalene
Specific impulse,[a] sec	100	40
Propellant,[b] lb	50	125
Solar cells, lb	4	3
Batteries, lb	2	1
Hardware,[c] lb	5	7
Total weight, lb	61	136

[a] Based upon 60°F for water, 140°F for naphthalene, nozzle area ratio of 100/1, and specific heat ratio = 1.33, assuming "frozen" internal molecular vibrations during expansion in the nozzle.
[b] For a total impulse of 5000 lb-sec.
[c] Propellant storage tank, valves, nozzles, and plumbing.

24-hr circular orbit are of the order of 10^{-4} lb and 10^{-3} lb-ft, respectively.

2) Electrical power requirements range from a few watts for low specific impulse (100 sec) cold gas jets to a few hundred watts for high specific impulse (10,000 sec) ion jets.

3) For a two-year mission, the total propulsion system weight is practically the same (about 60 lb) for cesium ion electrical propulsion as for cold gas propulsion using either ammonia or water vapor.

4) The reliability of vapor jet propulsion systems, e.g., water or naphthalene, is considered superior to other types because of a) inherent mechanical simplicity, b) low material stresses, and c) adaptability to the use of low wattage, low voltage, unregulated electrical power.

5) For missions much longer than two years, the potential capabilities of electrical propulsion systems are superior to those of other systems considered here because the high specific impulse of electrical propulsion makes propellant weight a relatively unimportant factor in determining the total weight of the system. However, the long-life reliability of storage batteries, electronic power converters, and electrical particle accelerators all must achieve considerable improvement over present demonstrated capability to make this approach of much practical value.

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