NASA TECHNICAL MEMORANDUM

NASA TM X-1936

NASA TM X-1936

COMPARISON OF SOLAR-ELECTRIC PROPULSION SYSTEMS FOR THE SYNCHRONOUS EQUATORIAL SATELLITE-RAISING MISSION

by Frank J. Hrach

Lewis Research Center

Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • JANUARY 1970

1. Report No. NASA TM X-1936	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle COMPARISON OF SOLAR-ELECTRIC PROPULSION SYSTEMS FOR THE SYNCHRONOUS EQUATORIAL SATELLITE-RAISING MISSION		5. Report Date January 1970
		6. Performing Organization Code
7. Author(s) Frank J. Hrach		8. Performing Organization Report No. E-5254
9. Performing Organization Name and Address Lewis Research Center National Aeronautics and Space Administration Cleveland, Ohio 44135		10. Work Unit No. 124-09
		11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		13. Type of Report and Period Covered Technical Memorandum
		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

A comparison among five solar-electric propulsion systems is made for the mission of raising a direct-broadcast television satellite from a subsynchronous Earth orbit to a synchronous equatorial orbit. The electric propulsion systems considered all use the solar-electric power provided by the satellite but use different thrusters. The electric power and payload mass of the satellite are assumed to be specified. The electric propulsion system is used in combination with a hypothetical launch vehicle in the Atlas-Centaur weight class, a vehicle which is unable to deliver the specified payload mass to the final orbit, even with the help of an additional chemical upper stage.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Solar-electric propulsion Solar cells Electric propulsion Low-thrust propulsion Synchronous satellite Television broadcast satellite	Unclassified - unlimited

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	26	$3.00

*For sale by the Clearinghouse for Federal Scientific and Technical Information
Springfield, Virginia 22151

COMPARISON OF SOLAR-ELECTRIC PROPULSION SYSTEMS FOR THE SYNCHRONOUS EQUATORIAL SATELLITE-RAISING MISSION

by Frank J. Hrach

Lewis Research Center

SUMMARY

A comparison of various solar-electric propulsion systems is made for the mission of raising a direct-broadcast television satellite from a subsynchronous Earth orbit to a synchronous equatorial orbit. In each case the satellite power source supplies the electric thrusters and, therefore, the electric propulsion system is not charged for a power source. The electric systems are assumed to be used in combination with a booster that is not able to deliver the required payload mass to the final orbit with chemical propulsion only. The analysis is conducted because of the prospect of a future cost advantage resulting from the use of an electric propulsion system in combination with a small launch vehicle.

The assumed booster is a hypothetical chemical booster in the Atlas-Centaur weight class. It is assumed that the required gross payload of the satellite, which includes a 25-kilowatt solar-electric power source, is about 2800 pounds (1270 kg) and that all the satellite power is available for use by the electric propulsion system. The selected orbit from which electric propulsion is initiated is circular and has a 28.5° inclination. The altitude of the orbit is not specified, but instead a range of values is considered. Five different electric propulsion systems are studied. The systems, identified by the thrusters which they employ, are (1) resistojet thruster; (2) 30-centimeter-diameter insulated-grid ion thruster; (3) 50-centimeter-diameter ion thruster; (4) magnetoplasmadynamic thruster; and (5) 15-centimeter-diameter ion thruster.

The system employing the resistojet thruster is able to perform the mission in the least time. However, because it requires hydrogen propellant and a large initial mass, the resistojet system encounters disadvantages not present in the other systems. The 30-centimeter-diameter ion thruster system is next in propulsion time required. The 15-centimeter-diameter ion thruster system requires more time than the other systems but utilizes hardware presently under development for the SERT II mission.

INTRODUCTION

A possible future application of electric propulsion is the raising of a direct-broadcast television satellite from a subsynchronous Earth orbit to a synchronous equatorial orbit. A major incentive for considering electric propulsion for a mission of this type is that a large amount of electric power is available on the satellite for use by the electric thrusters and the electric propulsion system is not penalized for the power source. This fact gives hope that the use of electric propulsion might be economical for this application, especially if suitable thrusters have been developed for other purposes or if development cost can be distributed over a large number of missions. Although boosters have been developed which are capable of performing the mission with chemical propulsion only, future worldwide requirements for direct-broadcast television may justify the use of a smaller (and presumably lower cost) booster in combination with an electric propulsion system. An analysis involving cost and other important mission factors (not undertaken in this study) would be required to determine whether the use of electric propulsion for this application offers a future cost advantage.

Nations presently without a large booster capability are unable to place the estimated required mass of a communications satellite into synchronous equatorial orbit with chemical propulsion alone. They are led to consider the use of electric propulsion in combination with their small boosters to increase present payload capability. References 1 and 2 are two rather recent studies of the use of electric propulsion for this application.

In both of these references, as well as in previous studies in this area, the initial orbit from which electric propulsion is begun is assumed to be circular and to have a low fixed value of altitude. In this analysis, the initial orbit is assumed to be circular but its altitude is varied to determine the effect of this parameter. A second unique feature of this analysis is the optimization of specific impulse for electric thrusters having an efficiency curve that rises with increasing specific impulse.

Five different electric propulsion systems, identified by the thrusters they employ, are considered for the mission and compared to reveal the advantages and disadvantages of each. The systems studied employ thrusters of the electrothermal, electromagnetic, and electrostatic types. The mass and the electric power of the satellite are assumed to be determined by the final application (direct-broadcast television satellite), the electric power source is assumed to be a solar-cell array, and all the power is assumed to be available for use by the electric propulsion system. It is also assumed that the propulsion systems are used in combination with a booster that is not able to deliver the required payload mass to the final orbit by means of chemical propulsion only; otherwise, the electric propulsion systems would be of no benefit.

Considerations important in selecting an electric propulsion system for the mission are as follows:

(1) Generally, a propulsion system employing high-specific-impulse thrusters is able to deliver a large payload mass but at the expense of long propulsion time, and consequently long mission time; on the other hand, a propulsion system employing low-specific-impulse thrusters might not be able to deliver the required payload mass.

(2) The thruster efficiency of the propulsion system, which depends upon the specific impulse at which the thruster is operated, determines the useful power in the exhaust, and as a result, affects the propulsion time.

(3) For any given power level, different electric propulsion systems have their own mass requirements for such items as thrusters and power conditioning, and systems using different propellants have their individual tankage mass requirements. These masses subtract from the final mass to reduce the payload.

ANALYSIS

Assumptions

The following assumptions are made for a performance comparison of the electric propulsion systems:

Electric power. - A solar-electric power source of 25 kilowatts is part of the gross payload mass of the direct-broadcast television satellite. The entire 25 kilowatts of power is available for use by the electric propulsion system to transfer the spacecraft (satellite and propulsion system) to the desired orbit.

Payload mass. - The gross payload of the direct-broadcast television satellite is approximately 2800 pounds (1270 kg). This estimated value is based upon a power source specific mass of 50 pounds per kilowatt and upon present-day level of technology for the other equipment.

Booster. - A hypothetical chemical launch vehicle in the Atlas-Centaur weight class is used to establish the initial orbit from which electric propulsion is started.

Mission profile. - The electric vehicle is launched due east from Cape Kennedy into an initial circular orbit resulting in an orbit inclined approximately 28.5° with the equator. The altitude of the assumed initial circular orbit is not specified but a range of values is considered. Electric propulsion, which both increases the altitude and reduces the inclination of the orbit, is employed until a synchronous equatorial orbit is attained (altitude, 19 300 nmi or 35 800 km).

Thrust steering. - The thrust vector is directed according to a near-optimal thrust steering program which is described in reference 3. This assumption is applied for the initial comparison of the various systems; a nonoptimal thrust steering program for a rigid, and hence simpler, spacecraft is considered in a later section.

Van Allen radiation damage. - The vehicle is exposed to a negligible amount of radiation from the Van Allen radiation belts. Factors which reduce the exposure of the vehicle to this type of damage are a high initial circular orbit altitude and a relatively high thrust level which reduces the time spent at low altitudes.

Continuous electric propulsion. - The vehicle does not enter into the Earth's shadow, which causes propulsion to be interrupted, until after the final orbit conditions are achieved and electric propulsion is terminated. Factors which reduce the possibility of entering the Earth's shadow are the same as those which reduce the exposure to Van Allen radiation (i.e., a high initial circular orbit and a relatively high thrust level which reduces propulsion time). An additional factor influencing continuous propulsion is the orientation of the initial orbit relative to both the Earth and the Sun which is determined by the time of year and the time of day the mission is begun.

The last two assumptions may be violated by some of the systems which are to be compared; therefore, any results must be examined to determine whether this is the case. If the assumptions are violated, the proper corrections must be applied. The correction for radiation damage is a degradation of the electric power, a mass penalty for thicker solar-cell cover glasses, or some combination of these corrections. The corrections for interrupted propulsion is an increase in the mission time because of the time wasted in the Earth's shadow, and possibly a mass penalty resulting from the requirement for cyclic operation of the electric propulsion system.

Systems Considered

Five electric propulsion systems, identified by the electric thrusters they employ, are considered for the mission. They are listed in table I, which also includes information on thruster type, propellant, specific impulse, and efficiency. The five thrusters are selected as representative of present technology as far as thruster efficiency and its dependence upon specific impulse is concerned; they may not be suitable flight items, however, because of other factors such as thruster lifetime. The electrostatic thrusters are classified according to their diameters. The values used are those of laboratory devices presently being developed; the actual thrusters need not necessarily be these sizes. The efficiency characteristic of the 30-centimeter-diameter insulated-grid ion thruster should be viewed as typical for insulated-grid ion thrusters, and that of the 50-centimeter-diameter ion thruster should be viewed as typical for large conventional-grid ion thrusters. The number of thrusters employed in each of the propulsion systems is equal to the total amount of available power divided by the power requirement of each thruster. Figure 1 illustrates the dependence of efficiency upon specific impulse for each thruster. The ef-

TABLE I. - ELECTRIC PROPULSION SYSTEMS CONSIDERED

Characteristic	System				
	A	B	C	D	E
	Thruster				
	Resistojet	30-Centimeter-diameter insulated grid ion	50-Centimeter-diameter ion	Magnetoplasmadynamic (MPD) arc jet	15-Centimeter-diameter ion (SERT II)
Thruster type	Electrothermal	Electrostatic	Electrostatic	Electrothermal; electromagnetic	Electrostatic
Propellant	Hydrogen	Mercury	Mercury	Ammonia	Mercury
Specific impulse, I, sec	840	Optimized	Optimized	Optimized	4770
Thruster efficiency, η_{th}	0.79	Optimized	Optimized	Optimized	0.71
Power conditioning efficiency, η_{pc}	1.0	0.88	0.88	1.0	0.88
Overall efficiency, η_{ov}	0.79	0.88 η_{th}	0.88 η_{th}	η_{th}	0.62
Reference	4	5	6	7	8

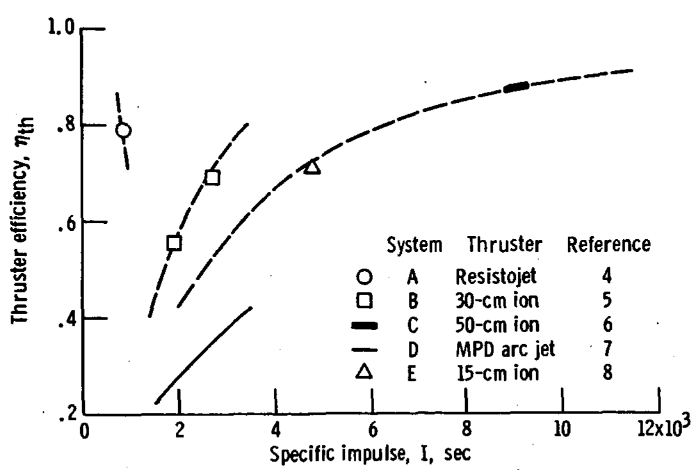

Figure 1. - Electric thruster efficiencies. No power conditioning or magnet losses included.

ficiency characteristics shown in the figure are discussed in the following paragraphs:

Resistojet thruster. - The efficiency of a resistojet thruster drops sharply as the specific impulse is increased because of the rapid increase in the amount of energy lost in the propellant as the temperature, and hence the specific impulse, is raised. The circled point on the curve (fig. 1) is an experimental data point for a resistojet thruster obtained from reference 4.

Thirty-centimeter-diameter insulated-grid ion thruster. - Two experimental data points for the 30-centimeter-diameter insulated-grid ion thruster obtained from reference 5 are shown in figure 1. An efficiency curve is drawn for this thruster, assuming that the thruster losses are constant over the range of specific impulse shown and equal to the losses of the 1900-second data point.

Fifty-centimeter-diameter ion thruster. - The 50-centimeter-diameter ion thruster curve is a conservative extrapolation of experimental data in the 9000-second range of specific impulse. The extrapolation is based upon the data reported in reference 6.

Magnetoplasmadynamic (MPD) arc jet thruster. - The efficiency curve (fig. 1) for the MPD arc jet thruster was drawn through data obtained at the Lewis Research Center from a performance evaluation of an Avco Corporation thruster which uses ammonia as a propellant. A curve similar to the one in figure 1 appears in reference 7.

Fifteen-centimeter-diameter ion (SERT II) thruster. - The SERT (Space Electric Rocket Test) II thrusters operate at a specific impulse of 4770 seconds with a corresponding thruster efficiency of 71 percent according to the most recent performance estimate (ref. 8). This point is indicated by a triangular symbol in figure 1.

The specific impulse for the resistojet thruster is specified to be that of the experimental data point shown in figure 1. The specific impulse for the 15-centimeter-diameter ion thruster is specified to be the value predicted for the SERT II mission. These values along with their associated efficiencies are listed in table I. Values of specific impulse for the 30- and 50-centimeter-diameter ion thrusters and the MPD arc jet thruster are not specified, but are optimized along the efficiency curves of figure 1. The procedure involved is explained in the next section.

The power conditioning efficiency for each of the electrostatic thrusters, the 30-, 50-, and 15-centimeter-diameter ion thrusters, is assumed to be 88 percent because of the electrostatic system requirement for high voltage levels. For the other thrusters, the resistojet and the MPD arc jet, a power conditioning efficiency of 100 percent is assumed.

Basic Equations

The initial mass of the electric spacecraft can be written as

$$M_0 = M_L + M_{adp} + M_{ps}$$

(All symbols are defined in the appendix.) The gross payload mass of the spacecraft includes the 25-kilowatt solar-electric power source, and would be equal to the gross payload mass for a chemical propulsion system if the electric system had a separate station-keeping propulsion system. (The electric propulsion system can be used for station keeping for the cases in which the propulsion system is integrated with the gross payload. This results in a slightly lower gross payload requirement for the electric system.) An adapter structure would be required only if it were necessary to separate the gross payload and the propulsion system after the final orbit is achieved. The mass of the propulsion system consists of the following items:

$$M_{ps} = (M_{th} + M_{sup} + M_{pc} + M_{har}) + M_p + M_{tk}$$

The power conditioning mass for which the propulsion system is charged is for the additional equipment required by the electric thrusters. Other power conditioning, required even if electric propulsion were not used, is included in the gross payload. The mass of each item included in the parentheses is dependent upon the electric power level. Because a particular value of power is assumed in this analysis, the total mass of these items is fixed for each system. This total is defined as

$$M_{fix} = M_{th} + M_{sup} + M_{pc} + M_{har}$$

The mass of the propellant tanks is assumed to be directly proportional to the propellant mass. With these assumptions and definitions, the gross payload can be written as

$$M_L = M_0\left[1 - \frac{M_p}{M_0}\left(1 + \frac{M_{tk}}{M_p}\right)\right] - (M_{fix} + M_{adp})$$

By means of the rocket equation, the propellant fraction which appears in the preceding equation can be written as

$$\frac{M_p}{M_0} = 1 - e^{-\Delta V/Ig}$$

The initial mass of the electric vehicle, which also appears in the payload equation, is the gross payload mass of the chemical booster used to establish the initial circular orbit. The available electric vehicle initial mass as a function of the initial circular orbit altitude is given in figure 2. Let this function be represented as

$$M_0 = F_1(h_0)$$

The curve of figure 2 is an estimate of the performance of a hypothetical launch vehicle in the Atlas-Centaur weight class.

The low-thrust ΔV requirement for the electric propulsion system, which appears

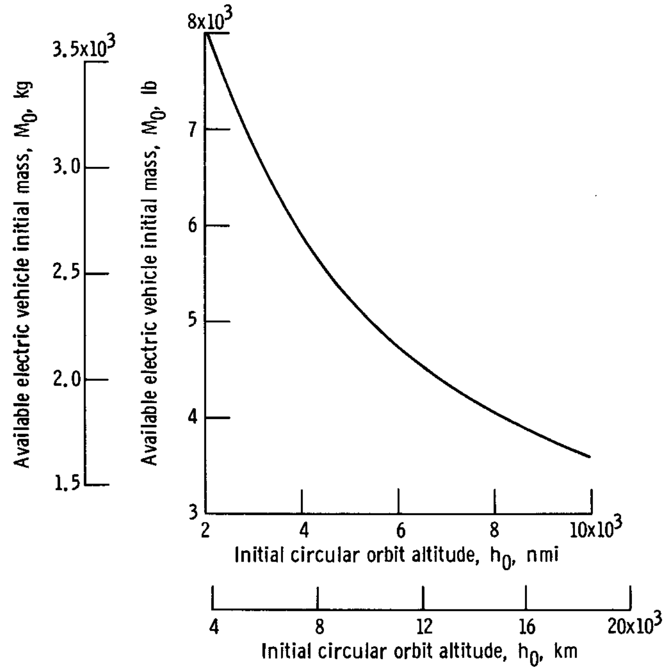

Figure 2. - Launch vehicle performance for hypothetical chemical vehicle in Atlas-Centaur weight class. Due east launch from Cape Kennedy.

in the propellant fraction equation, is presented as a function of initial circular orbit altitude in figure 3. Let this function be denoted by

$$\Delta V = F_2(h_0)$$

This propulsive requirement, obtained from the results of reference 3, is for a near-optimal thrust steering program in which both the altitude and inclination are changed as the vehicle goes through what is assumed to be a series of circular orbits (expanding circle approximation). Substituting for the propellant fraction, the initial vehicle mass,

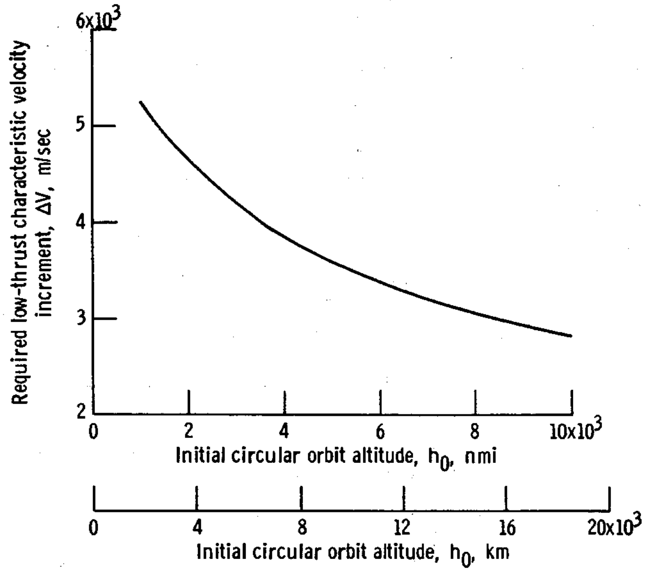

Figure 3. - Near-minimum low-thrust characteristic-velocity-increment requirement for synchronous equatorial satellite raising mission (cf., ref. 3). Inclination change, 28.5°; characteristic velocity increment, $\Delta V = \left[V_0^2 - 2V_f V_0 \left(\cos \frac{\pi}{2}\right) i_0 + V_f^2\right]^{1/2}$ for thrust-to-weight ratio less than 10^{-2}.

and the low-thrust ΔV, the expression for gross payload becomes

$$M_L = F_1(h_0)\left[1 - \left(1 - e^{-F_2(h_0)/Ig}\right)\left(1 + \frac{M_{tk}}{M_p}\right)\right] - (M_{fix} + M_{adp}) \tag{1}$$

Estimates for the fixed mass, the adapter structure mass, and the tankage fraction for each of the systems considered are presented in table II. The table also includes the estimated required gross payload mass for each system. Because of the large size of the resistojet propellant tank, it is assumed that the resistojet propulsion system is separated from the gross payload when the final orbit is reached and, hence, cannot be used for station keeping. An additional 157 pounds (71 kg) of mass was estimated to be required for a separate station-keeping propulsion system for this case. With the estimates presented in table II, the gross payload is a function only of the initial circular orbit altitude and the specific impulse:

$$M_L = F_3(h_0, I)$$

TABLE II. - ELECTRIC PROPULSION SYSTEM ASSUMPTIONS

System	Thruster	Estimated required gross payload, M_L		Number of thrusters assumed	Mass assumptions								Total, fixed mass of electric propulsion system, M_{fix}		Adapter structure, M_{adp}		Assumed tankage and insulation fraction, M_{tk}/M_P
					Thruster or thrusters, M_{th}		Thruster support structure, M_{sup}		Power conditioning, M_{pc}		Electrical harness, M_{har}						
		lb	kg		lb	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb	kg	
A	Resistojet	[a]2963	1344	1	150	68	10	4	0	0	26	12	186	84	195	88	0.206
B	30-Centimeter-diameter insulated-grid ion	2806	1273	16	100	45	40	18	109	50	34	15	283	128	---	--	.051
C	50-Centimeter-diameter ion	↓	↓	4	80	36	10	4	149	68	31	14	270	122	---	--	.051
D	MPD arc thruster	↓	↓	1	75	34	5	3	9	4	7	3	96	44	---	--	.068
E	15-Centimeter-diameter ion (SERT II)	↓	↓	22	165	75	50	22	109	50	34	15	358	162	---	--	1.00

[a]Additional 157 lb (71 kg) required for station-keeping propulsion system.

The electric power used for propulsion can be written as

$$P = \frac{K\dot{M}(Ig)^2}{2\eta_{ov}}$$

The propellant fraction of the electric vehicle can be related to the propulsion time as follows:

$$\frac{M_p}{M_0} = \frac{1}{M_0}\dot{M}t = 1 - e^{-\Delta V/Ig}$$

Solving this equation for propulsion time and substituting for the mass flow rate from the power equation yield

$$t = KM_0 \frac{(Ig)^2}{2\eta_{ov}P}\left(1 - e^{-\Delta V/Ig}\right)$$

The overall efficiency, which is equal to the product of the power conditioning efficiency and the thruster efficiency, is a function of the specific impulse through the thruster efficiency (fig. 1). Let this function be represented as

$$\eta_{ov} = \eta_{pc}\eta_{th} = F_4(I)$$

When this expression for overall efficiency is used and the initial vehicle mass and the low-thrust ΔV are substituted for as before, the expression for propulsion time becomes

$$t = KF_1(h_0)\frac{(Ig)^2}{2F_4(I)P}\left[1 - e^{-F_2(h_0)/Ig}\right] \tag{2}$$

It can be seen in equation (2) that, for a particular value of electric power, propulsion time is a function only of the initial circular orbit altitude and the specific impulse:

$$t = F_5(h_0, I)$$

For the resistojet and the 15-centimeter-diameter ion thruster systems (for which the values of specific impulse are specified), values of gross payload and propulsion time can be obtained from equations (1) and (2), respectively, for various values of initial circular orbit altitude. Gross payload for these systems can then be presented as a function of propulsion time with initial circular orbit altitude as a parameter. For each of the other systems (for which the specific impulses are not specified), the value of specific impulse which yields the largest final mass, and hence the largest gross payload, for a particular propulsion time (or equivalently, the least propulsion time for a given value of final mass) can be obtained iteratively. Results of the form described above can then be presented for the optimized values of specific impulse. The problem of optimizing the specific impulse for electrostatic engines used for this mission and the sensitivity of the optimum value to changes in the input assumptions are examined in detail in reference 9. Reference 9 also demonstrates that for this mission maximizing final mass for a particular propulsion time is equivalent to minimizing the propulsion time for a particular value of final mass.

RESULTS AND DISCUSSION

System Performance Comparison

Figure 4 presents the performance of the five systems. For each system, gross payload is shown as a function of propulsion time with the corresponding initial circular orbit altitudes in units of 1000 nautical miles (1850 km) indicated on the curves. From the figure it can be seen that in each case the gross payload mass that can be delivered to syn-

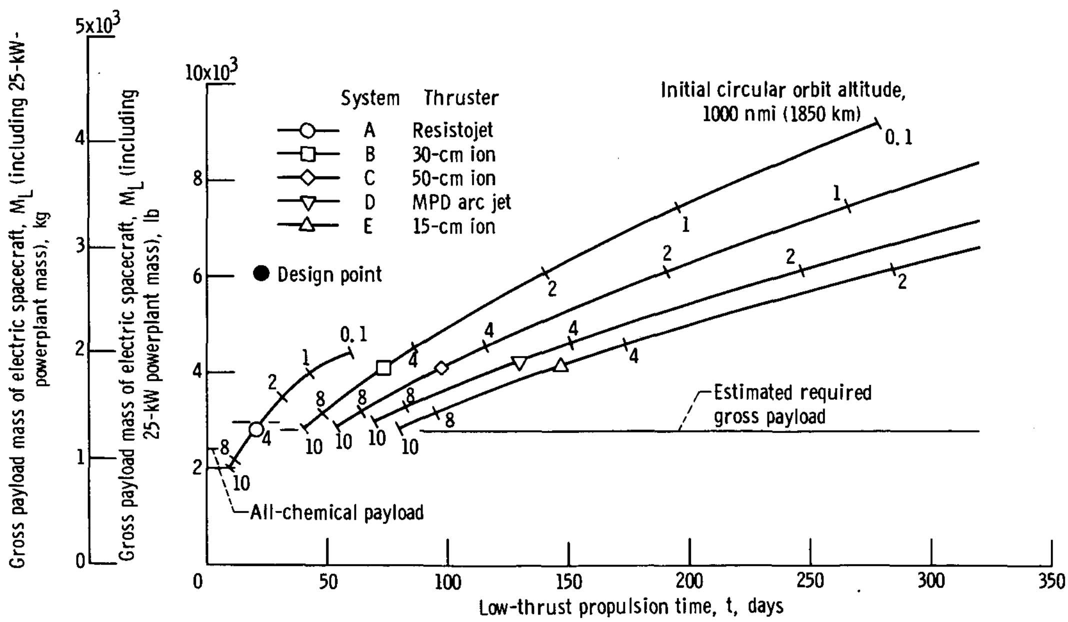

Figure 4. - System performance comparison for synchronous equatorial satellite-raising mission. Hypothetical chemical launch vehicle in Atlas-Centaur weight class plus a solar-electric propulsion system; inclination change by electric system, 28.5°; electric power, 25 kilowatts; no solar-cell degradation; near-optimal thrust steering.

chronous equatorial orbit can be increased by starting electric propulsion from a lower initial orbit. When this is done, the propulsion time required for the mission is increased. The values of estimated required gross payload mass are indicated on the figure. The value for all the systems, except the resistojet system, is 2806 pounds (1273 kg). The value for the resistojet system is 2963 pounds (1344 kg). For comparison, the all-chemical payload of 2400 pounds (1090 kg) is indicated in the figure. In this case, the launch vehicle is used to accomplish the perigee maneuver required to transfer from parking orbit to synchronous altitude. The apogee maneuver is then accomplished with a solid rocket propulsion system with a specific impulse of 300 seconds and sized to deliver maximum payload to synchronous equatorial orbit within the capabilities of the assumed launch vehicle.

For ease of comparison, the values of propulsion time required to perform the mission with the estimated required values of gross payload mass are taken from figure 4 and listed in table III, along with the associated values of specific impulse and initial circular orbit altitude. It should be emphasized that the values in table III are for a near-optimal thrust steering program resulting in a near-minimum low-thrust ΔV requirement, and with no allowance for a possible increase in the mass of the various systems. As indicated in table III, the optimum value of specific impulse for the 30-centimeter-diameter insulated-grid ion thruster is approximately 2100 seconds, and that for the MPD arc jet thruster approximately 1900 seconds for the estimated required gross payload. The specific impulse for the 50-centimeter-diameter ion thruster would have optimized at a value below 2000 seconds for the required gross payload but, because it is believed that this

TABLE III. - SYSTEM PERFORMANCE COMPARISON (FIG. 4)

System	Thruster	Estimated required gross payload, M_L		Propulsion time, t, days	Specific impulse, I, sec	Initial circular orbit altitude (approx), h_0	
		lb	kg			nmi	km
A	Resistojet	2963	1344	22	840	4 000	7 410
B	30-Centimeter-diameter insulated grid ion	2806	1273	39	2100	10 000	18 530
C	50-Centimeter-diameter ion	↓	↓	52	2000	10 000	18 530
D	MPD arc jet	↓	↓	62	1900	11 000	20 390
E	15-Centimeter-diameter ion (SERT II)	↓	↓	78	4770	10 000	18 530

value is the lowest at which this thruster can operate, the 2000-second value was used. It was observed that for the cases in which the specific impulse was optimized, the optimum value of specific impulse increases slightly as the required gross payload is increased. As indicated in table III, the initial circular orbit altitude for each system, except the resistojet system, is approximately 10 000 nautical miles (18 530 km). The value for the resistojet system is about 4000 nautical miles (7410 km).

All the systems can perform the mission if sufficient propulsion time is allowed. Certainly the best system would be the one which is able to perform the mission in the least time, if no other considerations were important. From table III it can be seen that the resistojet thruster system can perform the mission in the least time, approximately 22 days; however, this system has associated with it the following two disadvantages which the other systems do not have:

The first is that the growth potential of the resistojet thruster system is limited. At the initial circular orbit altitude of 100 nautical miles (185 km), the value of gross payload mass for the resistojet thruster system is approximately 4400 pounds (2000 kg), as can be seen from figure 4. The maximum value of gross payload mass for the other systems is considerably higher (e.g., over 9000 lb (4080 kg) for the 30-cm-diam insulated-grid ion thruster system). Of course, the consideration of radiation damage from the Van Allen belts may restrict the range of useful initial altitudes to values much higher than the 100-nautical-mile (185-km) point cited.

The second disadvantage is the need for a hydrogen propellant tank. The large propellant requirement of the resistojet thruster system and low density of the hydrogen propellant demand a considerably larger tank than is needed for the mercury or ammonia propellants of the other systems. Furthermore, the liquid-hydrogen propellant requires sufficient insulation that the boiloff rate does not exceed the required mass flow rate. The other systems do not have as severe a tank insulation problem. Consequently, the propellant tank for the resistojet thruster system would be heavy, unwieldy, and expensive compared to those for the other systems. (The propellant tank for the resistojet thruster system is discussed again in the next section.)

The system requiring the next lowest amount of propulsion time is the 30-centimeter-diameter insulated-grid ion thruster system. As can be seen in table III, the time required is 39 days. Even though this system requires almost twice the time for the resistojet thruster system, it has some compensating advantages. The mercury propellant of this system is easily stored and because of its high density does not require a large propellant tank. The relatively high specific impulse results in a low propellant requirement, which, in turn, results in a small required initial spacecraft mass, and also a relatively high value of initial circular orbit altitude. A small spacecraft in both size and mass generally costs less. Also, the high initial orbit altitude minimizes the problems of radiation damage and shadow entry, and allows for spacecraft growth by permitting the start of electric propulsion at a lower altitude.

The 15-centimeter-diameter ion thruster system also uses mercury as a propellant and has the same advantages from using this propellant that the 30-centimeter-diameter ion system has. The propulsion time required by this system is longer, 80 days, about twice that for the 30-centimeter-diameter ion thruster system. However, the 15-centimeter-diameter ion thruster system has the additional advantage of using hardware presently being developed for the SERT II mission.

Preliminary Spacecraft Designs

The system performance comparison of the preceding section allows the selection of a typical example case for each system so that preliminary spacecraft designs can be drawn. An initial circular orbit altitude of 5000 nautical miles (9270 km) was selected for all the systems except the resistojet thruster system for which an altitude of 4000 nautical miles (7410 km) was chosen. The lower altitude was chosen for the resistojet thruster system because it cannot deliver the required mass to the desired orbit starting from the 5000-nautical-mile (9270-km) orbit. An orbit lower than 4000 nautical miles (7410 km) was not selected because the initial mass of the spacecraft would be much larger than that for the other systems. Starting from these altitudes, the Van Allen radiation damage is slight because the high-density regions of both the protons and electrons trapped in the Earth's magnetic field lie below these initial orbits. It can be seen in figure 4 that these choices, indicated by the data points on the individual curves, allow a margin of over 1000 pounds (454 kg) for each system, except for the resistojet thruster system for which no margin is allowed. For these values of initial orbit altitude, preliminary designs for spacecraft employing the various electric propulsion systems were drawn.

Figure 5 shows a preliminary design of a spacecraft employing a 15-centimeter-diameter ion thruster propulsion system in orbit about the Earth with its solar panels deployed. (Designs for spacecraft employing the other systems, except that for the resistojet thruster system, would be similar.) In this design, the main spacecraft body is a tube frame. The electric thrusters are mounted on one end of the frame and the positionable parabolic antenna is mounted on the other. The solar panels are attached to the sides of the frame and are deployed by means of a pantograph linkage similar to that used on the Pegasus satellite. A fin-and-tube radiator for dissipating the waste heat from the television transmitter and the other electronic equipment is located on the shadow side of the spacecraft. Although not shown in the figure, the transmitter and other electrical equipment, as well as the propellant and tankage for the low-thrust propulsion system, would be mounted inside the tube frame.

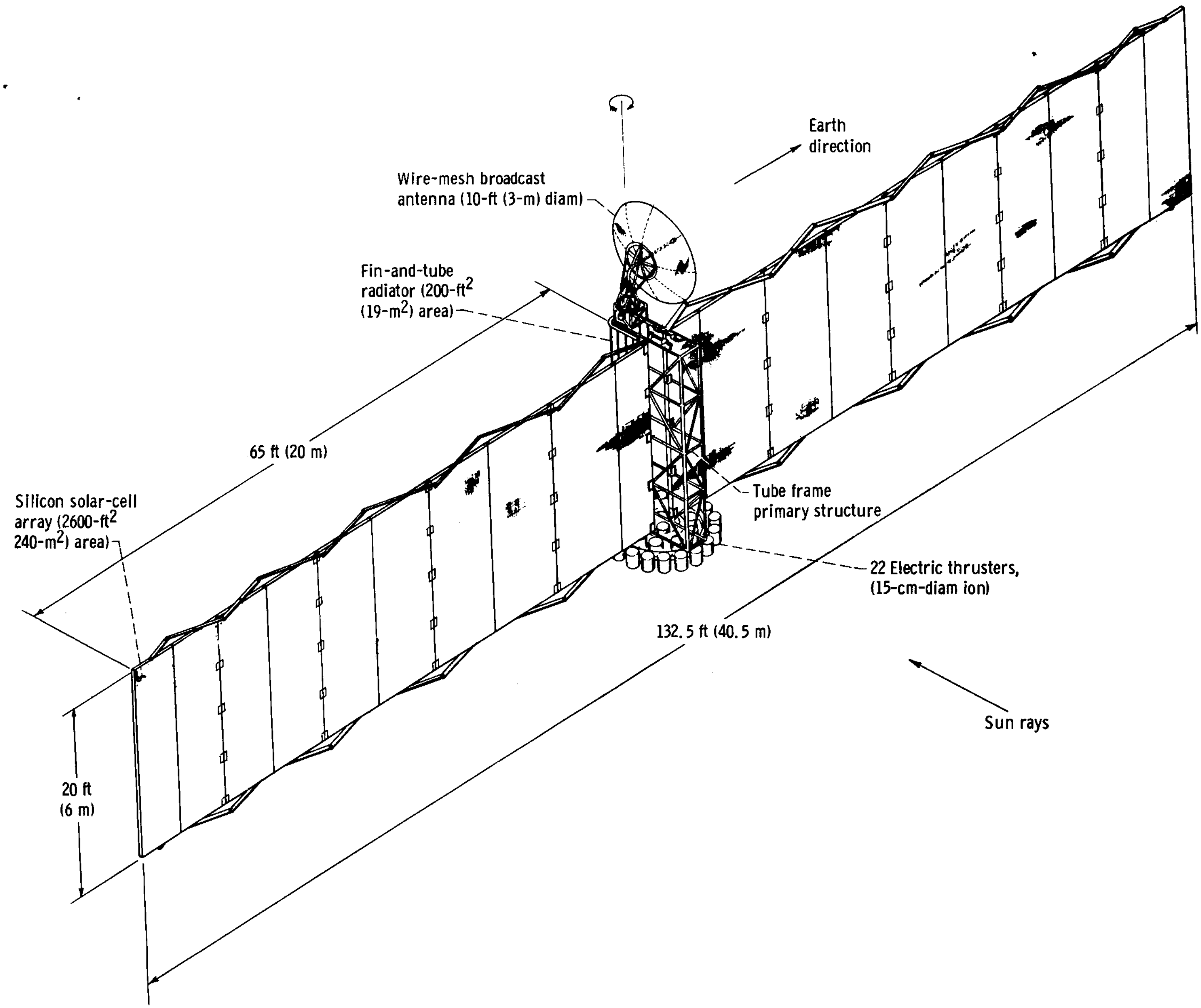

Figure 5. - Preliminary design of spacecraft employing 15-centimeter-diameter ion thruster propulsion system, shown in Earth orbit with its solar panels deployed. Electric power, 25 kilowatts.

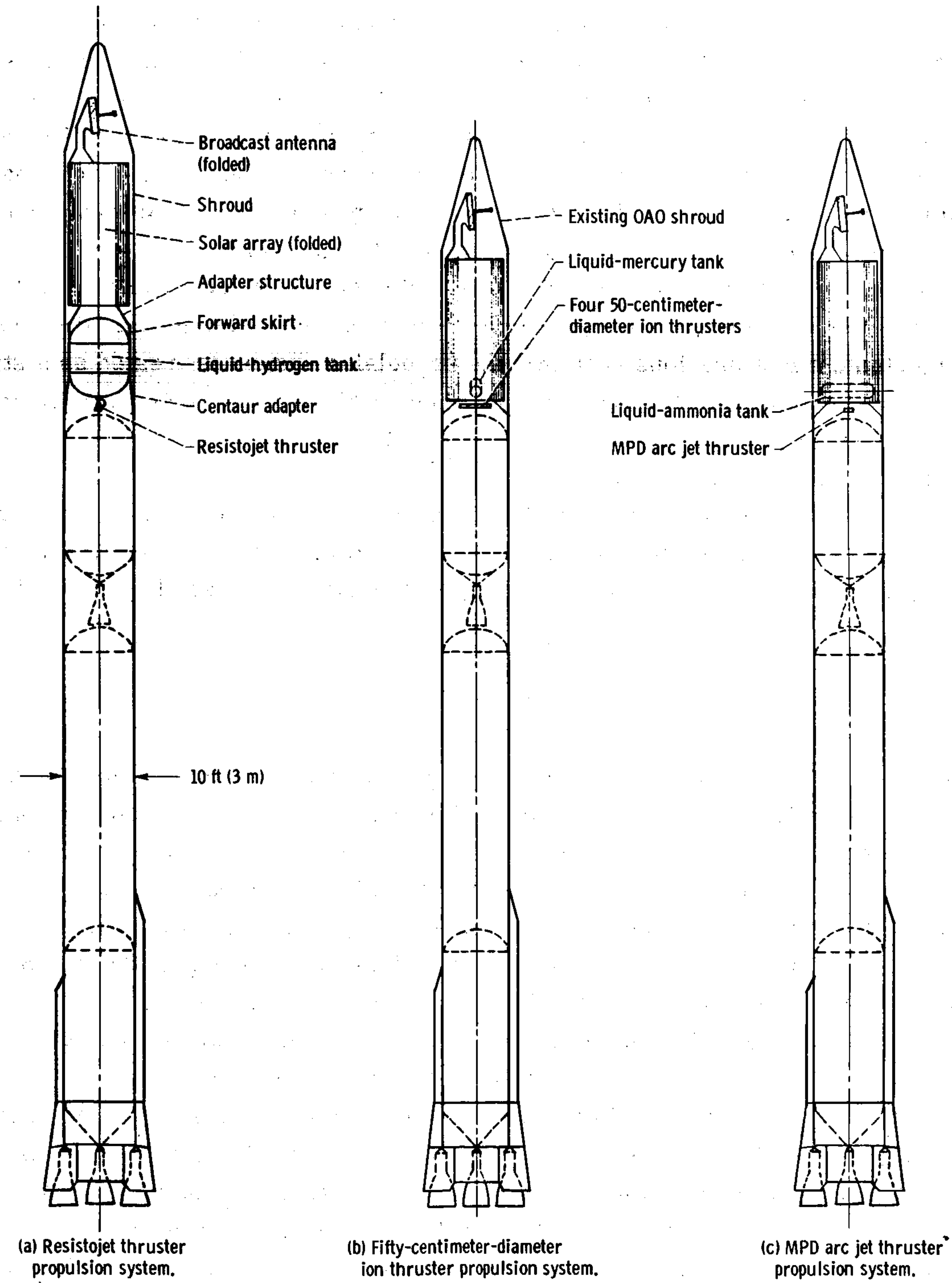

Figure 6. - Preliminary designs for spacecraft employing a resistojet thruster, a 50-centimeter-diameter ion thruster, and an MPD arc jet thruster propulsion system, shown in folded configuration on hypothetical chemical launch vehicle in Atlas-Centaur weight class. Solar-electric power, 25 kilowatts.

Figure 6 presents three spacecraft designs, each for a different low-thrust propulsion system, shown in the folded configuration on the launch vehicle. The spacecraft on the left employs a resistojet thruster propulsion system; the center one, a 50-centimeter-diameter ion thruster system; and the one on the right, an MPD arc jet thruster system. Spacecraft with propulsion systems employing the other electrostatic thrusters (i.e., the 30- and 15-cm-diam ion thrusters) would be similar in design to that of the 50-centimeter-diameter ion thruster system, and are therefore not shown in the figure. Note that spacecraft employing the electrostatic thruster propulsion systems and the MPD arc jet thruster system can fit beneath the existing OAO (Orbiting Astronomical Observatory) shroud. The spacecraft employing a resistojet thruster propulsion system would require a new, longer shroud. Because of the large propellant tank needed for the resistojet thruster system (as was pointed out in a previous section), the propulsion system is treated as a stage; that is, it is separated from the gross payload mass when the final orbit conditions are reached. Consequently, the electric propulsion system cannot be used for station keeping, and a separate system must be provided.

A mass tabulation for each spacecraft is presented in table IV. The negative margin for the resistojet thruster system means that the initial circular orbit altitude would have to be lowered slightly and the amount of hydrogen propellant increased by a small amount.

Thrust Steering

In order for a spacecraft of the design shown in figures 5 and 6 to follow the near-optimal thrust steering program assumed for the system performance comparison, it would be necessary for the solar panels to move relative to the spacecraft main body so as to always face the Sun, while the main body moved in a plane perpendicular to the radius vector from the earth (if the thrusters were rigidly attached to the spacecraft). The mechanism required to accomplish this was not included in the designs shown. Such a device would certainly complicate the problem of spacecraft control and would require that additional mass be allowed for the propulsion system. Although no estimate was made of the mass penalty, it is believed that it would be small. An additional problem would be that of transferring a large amount of electric power across a revolving joint. For simplicity, it might be better to rigidly attach the solar panels to the main body of the spacecraft (as indicated in figs. 5 and 6) and employ a thrust steering program that a spacecraft with this constraint could follow. Such a thrust steering program increases the low-thrust ΔV requirement, which, in turn, reduces the electric spacecraft performance.

A possible thrust steering program for a spacecraft with rigidly attached solar panels is shown in figure 7 and is described as follows:

(1) The spacecraft rotates about an axis perpendicular to the solar panels once every revolution about the earth.

TABLE IV. - TOTAL SPACECRAFT MASS TABULATIONS

System	Thruster	Initial circular orbit altitude, h_0		Vehicle mass														Available initial vehicle mass, M_0		Margin	
				Electric propulsion system mass								Adapter structure, M_{adp}		Estimated required gross payload, M_L		Total estimated required initial vehicle mass, M_0					
				Fixed, M_{fix}		Propellant, M_P		Tankage and insulation, M_t		Subtotal											
		nmi	km	lb	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb	kg
A	Resistojet	4000	7410	186	84	2212	1004	456	207	2854	1295	195	88	2963	1344	6012	2727	5900	2676	-112	-51
B	30-Centimeter-diameter insulated-grid ion	5000	9270	283	128	846	384	43	19	1172	532	---	--	2806	1273	3978	1804	5270	2390	1292	586
C	50-Centimeter-diameter ion			270	122	885	402	45	20	1200	544	---	--			4006	1817			1264	573
D	MPD arc jet			96	44	927	421	63	29	1086	493	---	--			3892	1765			1378	625
E	15-Centimeter-diameter ion (SERT II)			358	162	391	178	391	178	1140	517	---	--			3946	1790			1324	601

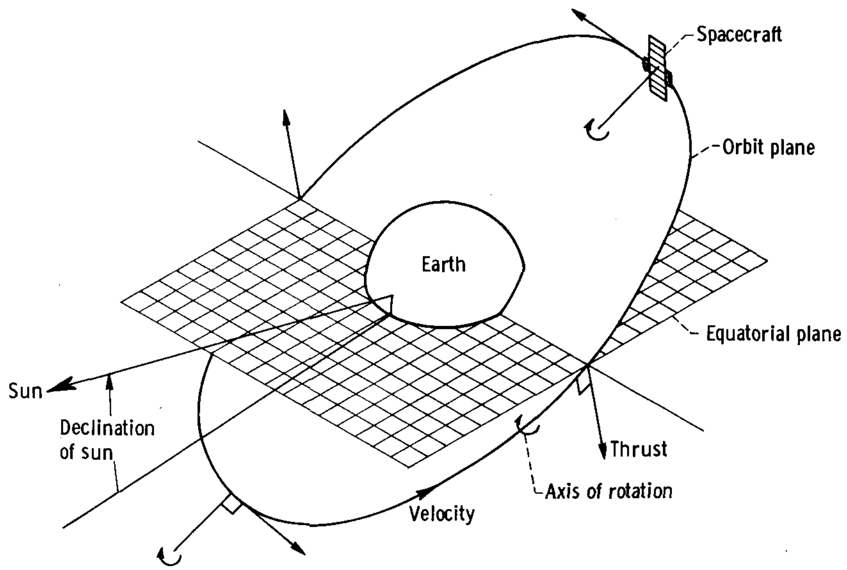

Figure 7. - Possible steering program for spacecraft with rigidly attached solar panels and thrusters.
and thrusters.

(2) The axis of rotation lies in the orbit plane which results in the thrust being confined to a plane perpendicular to the orbit plane.

(3) The axis of rotation is positioned in the orbit plane to allow the most sunlight to fall on the panels.

A penalty in electric power is incurred because the solar panels are not perpendicular to the sunlight and hence do not receive the maximum amount of input solar power. At any instant of time, the thrust vector can be resolved into three components: tangential (along the velocity vector), normal (perpendicular to the orbit plane), and radial (along the radius vector), and as the vector rotates the magnitude of each component varies with time. The tangential component increases the altitude of the orbit, the normal component reduces the inclination, but the radial component of thrust cancels itself over each half revolution about the Earth and does no good. This last effect is the principal reason for the inefficiency of this thrust steering program.

The thrust steering program for the simplified vehicle was incorporated into a computer code which calculates the motion of a spacecraft in Earth orbit under the influence of low thrust. Trajectory calculations were performed to determine the performance penalties incurred by using the nonoptimal thrust steering program. Performance with this thrust steering program depends upon the time of year the mission is performed (which determines the declination of the Sun) and the time of day at which the mission is begun (which determines the initial orientation of the orbit plane relative to the Sun-Earth line). For a particular launch date, a best initial orientation of the orbit plane can be found; that is, one that will allow the mission to be performed in the least time. The results for

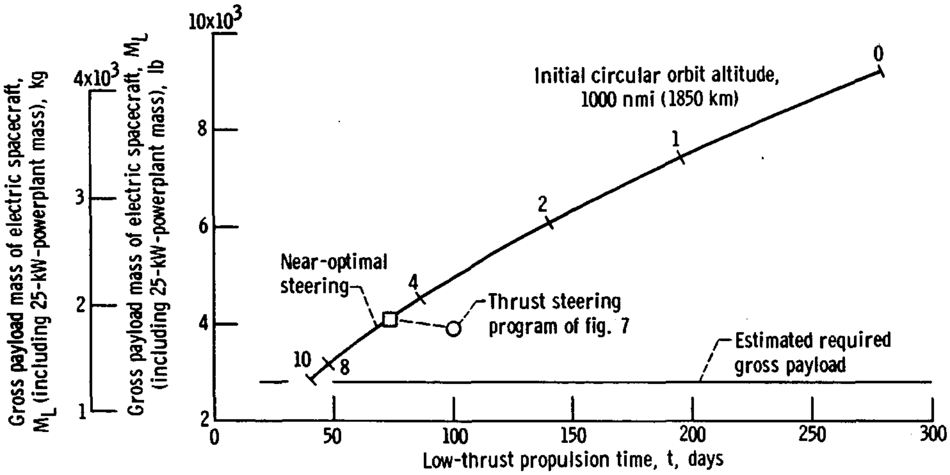

Figure 8. - Effect of thrust steering program (fig. 7) upon spacecraft performance. Hypothetical chemical launch vehicle in Atlas-Centaur weight class plus a solar-electric propulsion system employing 30-centimeter-diameter insulated-grid ion thrusters; inclination change by electric system, 28.5°; electric power, 25 kilowatts; no solar-cell degradation.

the nonoptimal thrust steering program which follow are for the best initial orbit for the particular launch date.

Figure 8 shows the effect of the nonoptimal thrust steering program upon the performance of a spacecraft employing a 30-centimeter-diameter insulated-grid ion thruster propulsion system. The curve shown in the figure is the same as the one in figure 4 and represents the performance with near-optimal thrust steering. The point on the curve indicates the performance for an initial circular orbit altitude of 5000 nautical miles (9270 km). For this same value of initial circular orbit altitude and for a March equinox launch date, the performance obtained by employing the nonoptimal thrust steering program is indicated by the circled point to the right of the curve. It can be seen that the required propulsion time is increased by 28 days, or by 39 percent. The additional propellant mass required, which subtracts from the margin, is 160 pounds (72 kg), an increase of 19 percent. This propellant increase corresponds to a low-thrust ΔV increase of 22 percent. These penalties have to be weighed against those for the movable solar array system which were previously discussed.

The value of specific impulse used in this comparison is the optimum value for near-optimal steering (~2100 sec). The specific impulse for the nonoptimal steering case was optimized and found to be only slightly higher (~2200 sec). This small change in specific impulse has a negligible effect upon the values of gross payload mass and propulsion time shown in figure 8.

To examine the effect of launch date on spacecraft performance, the computer calculations described above were repeated for a series of launch dates over the year. The variations in both propellant mass and propulsion time for these cases are small, approximately 3 percent for propellant mass and 15 percent for mission time. The variation in mission time includes the penalty associated with the spacecraft passage through the Earth's shadow cone.

SUMMARY OF RESULTS

Five different solar-electric propulsion systems are considered for the mission of raising a direct-broadcast television satellite from a subsynchronous Earth orbit to a synchronous equatorial orbit. A comparison is made to determine advantages and disadvantages of each system for this application. Using a hypothetical booster in the Atlas-Centaur weight class to establish the initial orbit, all of the electric propulsion systems studied can deliver the estimated required payload of approximately 2800 pounds (1270 kg) to a synchronous equatorial orbit. The propulsion system employing a resistojet thruster is able to perform the mission in the least time, 22 days for the assumed 25-kilowatt power level and with a near-optimal thrust steering program. The propulsion system employing 30-centimeter-diameter insulated-grid ion thrusters requires somewhat more time, 39 days. The propulsion time for the 15-centimeter-diameter ion thruster system is 80 days, and the times for the other systems lie between these latter two values.

Although the propulsion system utilizing a resistojet thruster has the advantage of minimum time over the other systems, it has the following two disadvantages associated with it: (1) the growth potential of a spacecraft with this propulsion system is limited; and (2) the hydrogen propellant of the resistojet system presents problems which the propellants for the other systems do not. Specifically, the hydrogen propellant tank must be considerably larger than those for the other systems and must be insulated to maintain the propellant at a cryogenic temperature. The next system, the 30-centimeter-diameter insulated-grid ion thruster system has the following advantages: (1) the growth potential of a spacecraft with this propulsion system is high; and (2) the size and mass of the spacecraft is small, primarily because of the high density of the mercury propellant. The 15-centimeter-diameter ion thruster system also has these advantages, and in addition, has the advantage of employing hardware which will soon be tested in space. This latter advantage has to be weighed against the longer propulsion time of the 15-centimeter ion thruster system.

In order for a spacecraft to follow the near-optimal thrust steering program (upon which the performance comparison is based), it would be necessary that the solar array move relative to the spacecraft (if the thrusters were fixed to the vehicle). This results in a complex spacecraft. A simpler design can be adopted at the cost of an increase in low-thrust ΔV and, hence, a reduction in electric propulsion system performance. As an example, by allowing the solar panels (as well as thrusters) to be rigidly attached to the spacecraft body, the performance of the 30-centimeter-diameter insulated-grid ion thruster propulsion system starting from an initial circular orbit altitude of 5000 nautical miles (9270 km) is affected in the following way: The propulsion time is increased by 39 percent, and the propellant requirement is increased by 19 percent. These penalties have to be weighed against those associated with a spacecraft having a movable solar array.

Launch date has a small effect upon spacecraft performance. For an initial orbit altitude of 5000 nautical miles (9270 km), a 30-centimeter-diameter insulated-grid ion thruster propulsion system, and a thrust steering program for a rigid spacecraft, the propellant mass varies by about 3 percent and the propulsion time varies by about 15 percent for launch dates throughout the year.

The use of electric propulsion is especially advantageous when the payload mass contains a high-level power source so that little or no mass penalty is incurred in providing electric power needed by the thrusters. This situation occurs in the proposed mission of raising a direct-broadcast television satellite to synchronous orbit. Whether the use of electric propulsion in combination with a small booster is cheaper than the use of a large booster requires more refined studies. In any event, the electric system has associated with it a time penalty of several months, the seriousness of which cannot be evaluated here.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, September 25, 1969,
124-09.

APPENDIX - SYMBOLS

e	Napierian base
$F_1, F_2, \ldots F_5$	auxiliary functions defined in section Basic Equations
g	acceleration due to gravity at Earth's surface, 9.80665 m/sec^2
h	circular orbit altitude, nmi (km)
I	specific impulse, sec
i	inclination, rad
K	conversion factor, 0.4536 kg/lb
M	mass, lb (kg)
P	electric power, W
t	propulsion time, days
V	circular orbit velocity, m/sec
ΔV	low-thrust characteristic velocity increment, m/sec
η	efficiency

Subscripts:

adp	adapter structure
f	final
fix	fixed
har	electrical harness
L	gross payload
ov	overall
p	propellant
pc	power conditioning
ps	propulsion system
sup	thruster support structure
th	thruster
tk	tankage and insulation
0	initial

Superscript:

$\cdot$	derivative with respect to time

REFERENCES

1. Schreitmüller, K. R.; and Wichmann, H. G.: Electric Thrustors for Communication Satellites. Paper presented at the Seventh European Space Symposium, Bordeaux, France, May 22-24, 1967.
2. Godwin, R. C.; and Rees, T.: Electric Propulsion for ELDO Vehicles. Paper presented at the Technical Meeting on Research and Development in Propulsion Technology, British Interplanetary Society, London, England, Oct. 31 - Nov. 1, 1967.
3. Edelbaum, Theodore N.: Propulsion Requirements for Controllable Satellites. ARS J., vol. 31, no. 8, Aug. 1961, pp. 1079-1089.
4. Page, Russell J.; Halbach, Carl R.; and Short, Robert A.: 3-kW Concentric Tubular Resistojet Performance. J. Spacecraft Rockets, vol. 3, no. 11, Nov. 1966, pp. 1669-1674.
5. Richley, Edward A.; and Kerslake, William R.: Bombardment Thruster Investigations at the Lewis Research Center. Paper 68-542, AIAA, June 1968.
6. Reader, Paul D.: Experimental Performance of a 50 Centimeter Diameter Electron-Bombardment Ion Rocket. Paper 64-689, AIAA, Sept. 1964.
7. Moeckel, W. E.: Promises and Potentialities of Electric Propulsion-Status of Thrustor Performance. Paper 66-1024, AIAA, Dec. 1966.
8. Byers, David C.; and Staggs, John F.: Sert II Flight-Type Thruster System Performance. Paper 69-235, AIAA, Mar. 1969.
9. Hrach, Frank J.: Optimum Specific Impulse for Electrostatic Engines Used for Satellite Raising to Synchronous Equatorial Orbit. NASA TM X-52568, 1969.