Extraecliptic 1.0 a.u. Constant Power Electric Propulsion Missions

Alfred C. Mascy*
NASA Mission Analysis Division, Moffett Field, Calif.

Performance of electric propulsion systems using constant thrust with coast trajectories is presented for the 1.0 a.u. out-of-the-ecliptic missions. Ranges of powerplant specific mass, mission times, and planar inclinations are explored. All mission payloads have been maximized on the basis of optimized coast times, specific impulse, powerplant mass and thrust vector orientation, as well as optimum velocity supplied by the Earth departure stage. Results indicate that through the proper combination of high and low thrust, missions with extremely high inclinations to the ecliptic may be accomplished. The power requirements for these missions are delineated and appear to be modest.

Introduction

WITHIN the scientific community there is a growing interest in conducting unmanned missions out of the ecliptic plane.[1,2] Scientists desire to know what phenomena exist in space at different latitudes and along planes oriented at angles to our Earth-sun plane. For ease of communication and for monitoring of extraecliptic phenomena, there would be some advantages of maintaining the orbit at the same Earth-sun distance. Since the preponderance of existing in situ data is at 1.0 a.u. within our Earth-sun or ecliptic plane, data accumulated at 1.0 a.u. but at other latitudes would be invaluable in determining the spherical symmetry or asymmetry of solar space phenomena. Whether for magnetic field or micrometeoroid measurements, or solar flare observance or solar plasma experiments, there is a great interest in obtaining 1.0-a.u. extraecliptic trajectories.

Because of the high velocity requirements, chemical propulsion systems alone are not capable of transporting reasonable payloads efficiently to 1.0-a.u. orbits inclined much higher than about 20°. A close passage of the planet Jupiter may be used to increase a spacecraft's inclination to be ecliptic.[3] However, trip times to Jupiter are on the order of 500 days with an additional 500 days for the return trip to the vicinity of 1.0 a.u. Notwithstanding two passages through the asteroid belt and the uncertainties in the environmental parameters of Jupiter, the resultant highly elliptical trajectory, although inclined to the ecliptic, has only a relatively brief observation time at the desired high latitudes in the vicinity of 1.0 a.u. Of course, the trajectory might be circularized at 1.0 a.u. on its return from Jupiter, but the velocity requirements are prohibitive.

Low-thrust propulsion systems, however, permit performance of such high-velocity missions.[4] By the use of an electric propulsion system, circular 1.0-a.u. orbits of high inclinations may be obtained. Solutions of these low-thrust trajectories have been difficult to obtain in the past, requiring a search on Lagrange operators for the optimization of the thrust vector in three-dimensional space.[5] Switch times have been difficult to guess, and over-all optimization of system parameters, such as specific impulse (I_{sp}), power plant mass and initial acceleration, has been tedious. Often, to facilitate the obtaining of solutions, thrust has been assumed to take place only near the ascending and descending modes. The thrust vector orientation has also been constrained to be normal to the instantaneous orbit plane. These restrictions lead to very long mission times with decreased performance.

Presented as Paper 68-546 at the AIAA 4th Propulsion Joint Specialist Conference, Cleveland, Ohio, June 10–14, 1968; submitted June 6, 1968; revision received September 10, 1969.

* Research Scientist, Office of Advanced Research and Technology. Associate Member AIAA.

It is the purpose of this paper to present the performance characteristics of electric propulsion systems as applied to the 1.0-a.u., out-of-the-ecliptic missions using constant-thrust-with-coast trajectories. Through the use of new computer programs, all mission payloads have been maximized on the basis of optimized coast times, I_{sp}, powerplant mass, and thrust vector orientation, as well as optimum hyperbolic excess velocity V supplied by the Earth departure stage. The electric propulsion stage is optimally combined with a chemical propulsion stage for Earth departure. Results are shown both for departure from low Earth orbit using generalized high-thrust systems and for specific launch vehicles and upper stages.

Analysis

The electric propulsion system is assumed to operate after a high-thrust stage has given it an optimal V, which is used to provide an initial inclination to the ecliptic i_{initial}, and the electric stage provides the additional plane change to achieve the desired total inclination i_{final}. The transition between i_{initial} with the high-thrust system and start-up inclination with the low-thrust system is accomplished by asymptotic matching of the high-thrust departure velocity with the electric propulsion system initial velocity.[6] The low-thrust system is assumed to operate with constant power and constant I_{sp} commencing at the same longitude at which the high-thrust stage provided the initial plane change. A view of the heliocentric trajectory is shown in Fig. 1. It should be noted that the trajectory is not necessarily restricted to a 1.0-a.u. sphere during the trip. As a result of the optimization procedure, less energy is required if the trajectory is allowed to vary outward. At the end of the trip time, nevertheless, a 1.0-a.u. circular orbit has been established with the proper inclination to the ecliptic plane. Terminal conditions are

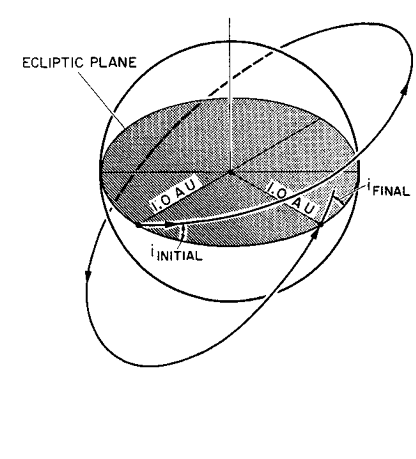

Fig. 1 View of typical heliocentric extraecliptic trajectory.

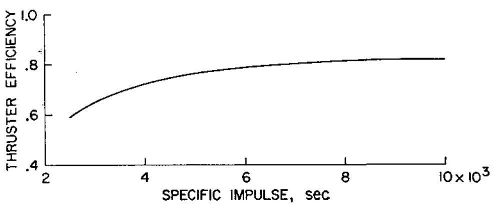

Fig. 2 Efficiency of electric thrustors.

selected such that the longitude of ascending node of the final orbit is close to the initial longitude. Hence, optimal trajectories are obtained which have approximately one solar revolution.

Data for the heliocentric portion of these extraecliptic missions which utilize thrust/coast/thrust trajectories were computed by means of a computer program developed by United Aircraft Corporation under contract to NASA.[7] This computer code employs an implicit finite-difference Newton-Raphson algorithm[8] for solving the system of differential equations that describes the heliocentric trajectory and system optimization problem. The program optimizes I_{sp}, powerplant mass fraction, powered time, and coast time while maximizing the payload mass fraction. Terminal conditions are input for a dummy planet with the desired change in inclination. The semimajor axis of the planet is taken at 1.0 a.u. and the eccentricity equals zero.

The thruster subsystem performance assumed for this study was based on mid-1970 estimates[9,10] and includes a power-conditioning efficiency of 90%. In Fig. 2, the efficiency of the over-all thruster subsystem is shown as a function of I_{sp}. The mass of the tanks for the electric propulsion propellant was assumed to be 3% of the propellant mass. The mass of the primary power, conversion equipment, power conditioning, thrusters, and structures necessary for support and attachment of these items is included in the definition of low-thrust propulsion system, and α is the ratio of this mass to the power delivered to the input terminals of the power-conditioning

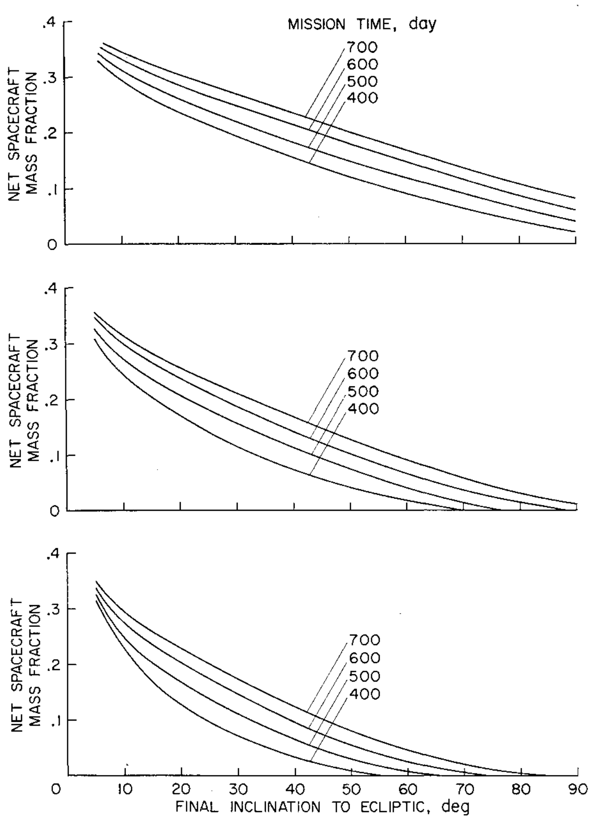

Fig. 3 Performance of combined high- and low-thrust systems a) $\alpha = 10$ kg/kw$_e$; b) $\alpha = 20$ kg/kw$_e$; c) $\alpha = 30$ kg/kw$_e$.

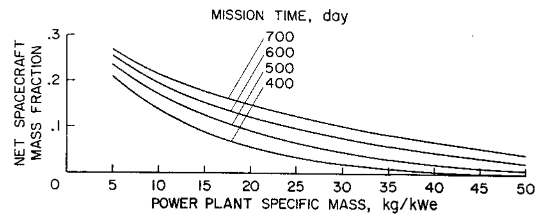

Fig. 4 Effect of powerplant specific mass on net spacecraft mass fraction of combined high- and low-thrust systems for 45° extraecliptic missions.

assembly. The net spacecraft mass presented in this study includes the science payload, data handling, computational, and communications equipment, attitude and thermal control, guidance, and the necessary support structures for these items; it is often referred to simply as payload mass.

Results and Discussion

Extraecliptic trajectories were computed for changes in planar inclinations up to 90° using a low-thrust stage with constant power. In Fig. 3, the net spacecraft mass fraction μ_n (ratio of net spacecraft mass to mass in low circular Earth orbit of 278 km altitude) is shown for final inclinations to the ecliptic plane up to 90°. The high-thrust stage assumed for the Earth departure and initial inclination maneuver was based on an I_{sp} of 450 sec and an inert mass fraction of 0.137. It may be seen that with the longer trip times, greater μ_n may be realized, and for a given trip time, μ_n decreases exponentially with increasing change in inclination. If powerplants were available with $\alpha = 10$ kg/kw$_e$, a 90° out-of-the-ecliptic mission could be accomplished in 700 days with $\mu_n \simeq 0.09$, that is, 9% of the mass originally in Earth orbit before high-thrust start-up. The sun's equatorial plane is inclined about 7° to the ecliptic, so that it may not be necessary to acquire such a high latitude for some observations such as magnetic field measurements the intensity of which might be maximum at about 83° inclination. Even with $\alpha = 30$ kg/kw$_e$, which is indicative of near-term state-of-the-art solar-powered systems, one could obtain $\mu_n = 0.025$ to 70° inclination in 700 days or a like amount to 60° in 600 days, or to 50° in 500 days. These are very high inclinations and, of course, lead to the same high latitudes at 1.0 a.u.

A mission that has received a good deal of interest is a 45° extraecliptic probe. Figure 4 shows μ_n vs α for this mission, with combined electric and chemical stages, for four mission times t_m. While μ_n appears to decrease exponentially with increasing α, it increases almost linearly with increasing t_m for the 400–700 day range considered here.

Of interest is the time history of the 45° extraecliptic mission using a chemical system combined optimally with an electrical system of $\alpha = 30$ kg/kw$_e$. In Fig. 5 are shown the distance R and the latitude vs the longitude positions along a 600-day mission. The time shown on the lower abscissa should be taken as approximate since the longitude is not a linear func-

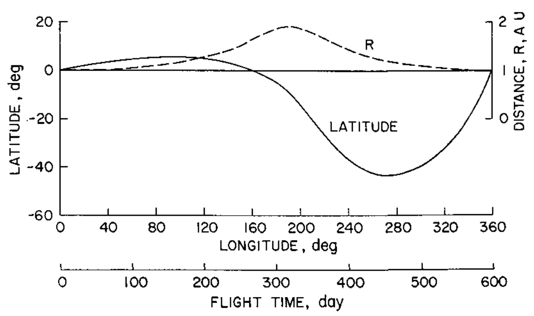

Fig. 5 Time history of heliocentric trajectory for 45° extraecliptic mission.

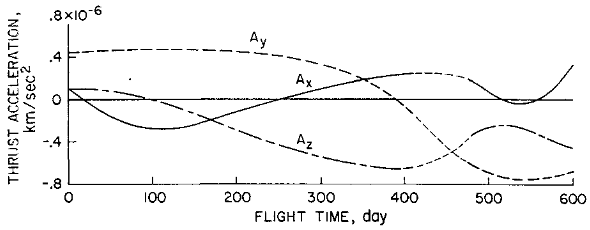

Fig. 6 Time history of thrust acceleration for 45° extraecliptic mission.

tion of time. It is seen that R reaches $\sim$ 1.8 a.u. at the midspan of the mission time and then returns to 1.0 a.u. after 600 days. The out-of-the-ecliptic component, latitude, is above the ecliptic during the first half of the mission and then dips down reaching almost $-45°$ at about 475 days. After 600 days, the trajectory has established a 1.0-a.u. circular orbit with $i = 45°$, and has traveled one solar revolution. The deviation from 1.0 a.u. during the trajectory is not too great, and it may be possible to extend the results contained herein to the approximate performance of solar-powered electric powerplants. As a consequence of optimization, solar-powered trajectories have some portions which deviate inward to smaller a.u. distances. They thereby gain advantage of the increased power available, offsetting the lower power at larger distances where plane changes are more effective. Preliminary results indicate that although the trajectories differ, the propulsion requirements are relatively close for both constant-power and solar-power systems for these missions. The time history of the thrust acceleration vector for the 45° mission is shown in Fig. 6. The magnitude of the Z component increases constantly until the coast phase (shown by the dotted lines). The spacecraft coasts to nearly 45° latitude, whereupon the thrusting resumes to return the trajectory to a 1.0-a.u. circular orbit.

A comparison of results of previous methods indicates that the optimization procedures employed in this analysis allow a large increase in payload over those cases which constrained the radius vector during flight. Likewise, t_m's are reduced by a factor of two or more compared with the constrained radius case, which allows thrusting only as the ascending and descending nodes. The computer program utilized herein optimized the length of the thrust phases as well as the level of acceleration. The code also had the capability of selecting one or two coasts in addition to optimizing the length of each coast phase. The highest performance was obtained when the propulsion system was operated almost continually. A single coast phase of short duration was found to be optimal. With a slight dependence on α, coast phases were generally less than 10% of the mission time.

The power requirements for a 45° extraecliptic mission were investigated for a range of α. In Fig. 7, the power required in kilowatts divided by the payload in kilograms is shown for a range of mission times. This factor, termed $\bar{P}$, appears to be not too sensitive to variations in α and therefore provides a good indicator of power requirements that depend more strongly on mission characteristics such as t_m and i. For example, for a 600-day mission, $\bar{P} \simeq 0.05$ kw$_e$/kg and is insensitive to α; if the payload weighed 240 kg, the primary power necessary for electric propulsion would be about 12 kw.

Figure 8 shows $\bar{P}$ vs inclinations for a 600-day mission, with combined propulsion systems as described previously, for three α's. Again, the insensitivity of $\bar{P}$ to α is noted for inclinations up to $\sim$60°. For higher inclinations, the power requirements become great for missions restricted to 600 days, and the payloads are very low, as indicated by Fig. 3. Hence, there is some benefit to extending t_m to 700 days for the higher inclinations. For a 700-day, 75° extraecliptic mission, $\bar{P} \simeq 0.09$ kw$_e$/kg, within the range of α examined.

Up to this point, the optimal combination of high and low thrust included a chemical stage which boosted the electric stage from a low circular parking orbit. To access the performance of electric propulsion in combination with some specific launch vehicle system, Fig. 9 was developed using the same method of asymptotic velocity matching to optimally combine the high- and low-thrust systems. These calculations are for an electric propulsion system which has $\alpha = 30$ kg/kw$_e$ and which is combined with an Atlas/Centaur launch vehicle. The characteristics of the launch system are shown by the dotted curve and achieve zero payload at about 14°. The addition of another chemical stage would increase the zero-payload inclination to $\sim$20°. However, the addition of an electric stage greatly improves the performance of this launch system, allowing the zero-payload inclination to increase to about 80° to the ecliptic plane for 700-day missions. The combined system can deliver 220 kg to 45° in 600 days, or 120 kg to 60° in 700 days. Longer mission times or larger launch vehicles would allow spacecraft to navigate over the celestial north pole with reasonable payloads.

In Fig. 10, are detailed the power requirement P, thruster beam velocity C, and hyperbolic excess velocity V with mission time for a 45° mission. The electric stage has $\alpha = 30$ kg/kw$_e$ and uses an Atlas/Centaur launch vehicle for

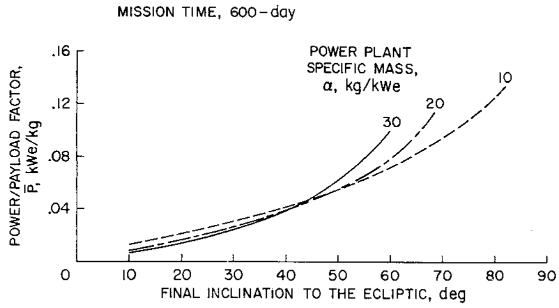

Fig. 8 Power requirements for combined high- and low-thrust systems for 600-day extraecliptic missions.

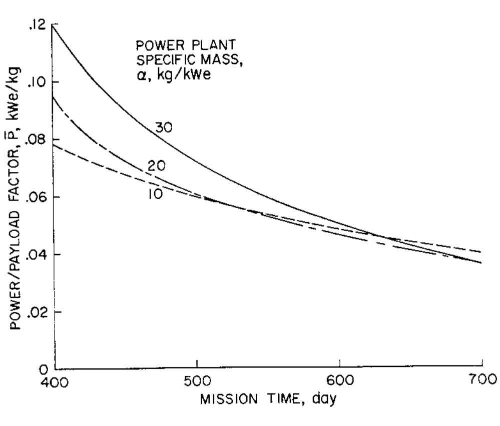

Fig. 7 Power requirements for 45° extraecliptic missions.

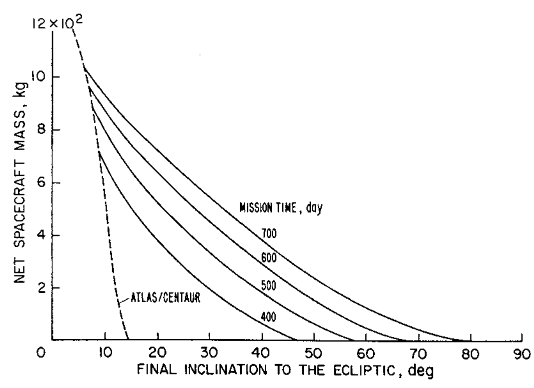

Fig. 9 Performance of selected launch vehicle in combination with electric propulsion; α = 30 kg/kw$_e$.

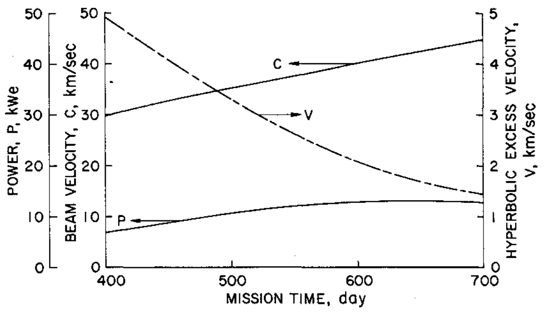

Fig. 10 System requirements for 45° extraecliptic missions; $\alpha = 30$ kg/kw$_e$.

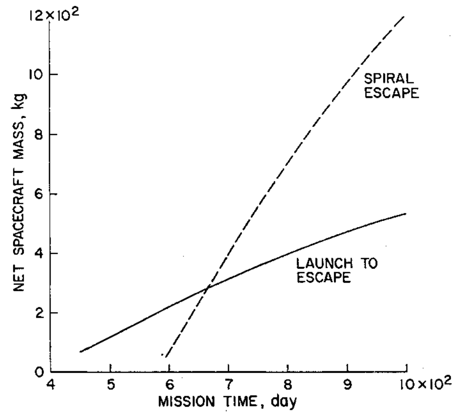

Fig. 11 Performance comparison of low-thrust spiral escape and high-thrust launch to escape for 45° extraecliptic mission; $\alpha = 30$ kg/kw$_e$.

Earth departure. The thruster velocity increases almost linearly with t_m, as less propellant is consumed; it is 40 km/sec for the 600-day mission. With increasing t_m the dependence on the launch system is eased, and V decreases. The 600-day mission requires a departure velocity of about 2 km/sec, which provides an initial inclination of about 4°. The power level remains fairly constant with mission time and is ~12 kw$_e$ for the 600-day trip; P appears to depend mostly on the characteristics of the launch vehicle.[11] Although there is an optimal P for a particular launch vehicle, P may be constrained to a lower value without a large decrease in net spacecraft mass. The 600-day 45° extraecliptic mission constrained to $P = 7$ kw$_e$ still provides 180 kg of net spacecraft.

The possible advantages of using the electric stage for Earth departure were also investigated. In Fig. 11, the net spacraft mass is shown for the 45° out-of-the-ecliptic mission for an optimally combined Atlas/Centaur launch vehicle and an electric stage of 30 kg/kw$_e$. The solid curve shows the performance for the system which uses the launch vehicle to provide Earth escape. The dotted curve shows the performance for a system which uses the launch vehicle to place the electric stage in a low parking orbit, from which it commences a low-thrust spiral away from Earth. For $t_m < 670$ days, the high-thrust launch to escape provides greater payload. Beyond 670 days, significant increases in payload may be realized through the use of low-thrust spiral escape for the departure maneuver. As an example, for an 800-day mission, the spiral-escape mode provides 700 kg of net spacecraft mass compared to 400 kg for the high-thrust escape mode. The power requirements, however, have increased markedly from 12 kw$_e$ for high-thrust escape to over 50 kw$_e$ for low-thrust escape.

Concluding Remarks

Electric propulsion extraecliptic missions of up to 90° inclination have been examined for powerplant specific masses (α's) up to 50 kg/kw$_e$ and missions times from 400 to 700 days. Through the optimal combination of high thrust to provide the initial inclination, reasonable payloads may be delivered to high inclinations. For example, for a 45° extraecliptic 600-day mission, with near term power plants, the payload is 8% of the initial weight in Earth orbit. Using an Atlas/Centaur launch vehicle, a payload of 220 kg (500 lb) may be realized. Similar payloads may be delivered to higher inclinations using larger launch vehicles, or longer mission times. The use of a low-thrust spiral for escape greatly increases the payload of missions in excess of about 700 days.

The power requirements for these missions are reasonable. A parameter $\bar{P}$, the primary power divided by the payload mass, is rather insensitive to variations in system parameters such as α, but retains a dependency on mission difficulty such as time and inclination. For example, the power required to propel a payload of 240 kg to 45° inclination in 600 days from Earth orbit is 12 kw, whether $\alpha = 10$, 20, or 30 kg/kw$_e$. The power may be constrained to lower levels without a large decrease in net spacecraft mass.

Through the optimization process, the trajectory departs slightly from 1.0 a.u. during the flight returning to a circular 1.0-a.u. inclined orbit at the end of the trip time. A brief investigation indicates that the optimization procedure employed in this analysis allows a large increase in payload over methods which constrain the radius vector. Trip times are also reduced by a factor of two over the constrained radius case, which allows thrusting only at the ascending and descending nodes.

References

[1] Biermann, L., "Some Aspects of the Physics of Interplanetary Space Related to Out-of-the-Ecliptic Studies," *Advances in Space Science and Technology*, Vol. 7, Academic, New York, 1965.

[2] Matthews, H. F. and Erickson, M. D., "The NASA Advanced Pioneer Mission," Paper 857D, April 1964, Society of Automotive Engineers.

[3] Minovitch, M., "Utilizing Large Planetary Perturbations for the Design of Deep-Space, Solar-Probe, and Out-of-Ecliptic Trajectories," TR 32-849, Dec. 1965, Jet Propulsion Lab., Pasadena, Calif.

[4] Mascy, A. C., Dugan, D. W., and Pitts, S. W., "Applications of Combined Electric, High-Thrust Propulsion Systems," *Journal of Spacecraft and Rockets*, Vol. 5, No. 7, July 1968, pp. 785–791.

[5] Hrach, F. J., "Out-of-the-Ecliptic Plane Probe Mission Employing Electric Propulsion," TN D-4455, April 1968, NASA.

[6] Melbourne, W. G. and Sauer, C. G., Jr., "Performance Computations with Pieced Solutions of Planetocentric and Heliocentric Trajectories for Low-Thrust Missions," *Jet Propulsion Laboratory Space Programs Summary No. 37–36*, Vol. IV, Pasadena, Calif., Dec. 1965.

[7] Mascy, A. C., "A Compilation of Current Computer Programs for Low-Thrust Trajectory Analysis," TM X-1824, July 1969, NASA.

[8] Van Dine, C. P., Fimple, W. R., and Edelbaum, T. N., "Application of Finite-Difference Newton-Raphson Algorithm to Problems of Low-Thrust Trajectory Optimization," AIAA Paper 65-698, Monterey, Calif., 1965; also *AIAA Progress in Astronautics and Aeronautics: Methods in Astrodynamics and Celestial Mechanics*, Vol. 17, edited by R. L. Duncombe and V. G. Szebehely, Academic, New York, 1966, pp. 377–400.

[9] Richley, E. A., "Bombardment Thrustor Investigations at the Lewis Research Center," AIAA Paper 68-542, Atlantic City, N. J., 1968.

[10] Masek, T. D., "Thrust System Technology for Solar Electric Propulsion," AIAA Paper 68-541, Atlantic City, N. J., 1968.

[11] Mascy, A. C., "Low-Thrust Mission Simulation—Feedback to Hardware Definition," *7th Electric Propulsion Conference*, AIAA, New York, 1969.