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Final Report

SOLAR ELECTRIC PROPULSION

MISSION REQUIREMENTS STUDY

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SUMMARY

This report summarizes the work performed under Contract NAS5-20126, which consisted of a one-year effort to generate optimum low thrust trajectory and performance data which are necessary to define the applicability of solar-electric propulsion as a means of primary propulsion for the unmanned exploration of the solar system.

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INTRODUCTION

This report summarizes the work performed under Contract NAS5-20126. The objective of this work is to generate optimum solar-electric trajectory and performance data required in the definition of solar-electric mission applications for the unmanned exploration of the solar system. The data generation effort was directed toward the investigation of high-energy missions, including solar probes, extra-ecliptic probes, and rendezvous missions to the asteroid Ceres and to the comets D'Arrest and Encke. Mercury orbiter and Pluto flyby missions were also investigated. Concurrent computer program improvements were implemented, which enhanced data generation efficiency.

I. SOFTWARE DEVELOPMENT

A significant portion of the effort involved in the current study consisted of computer program modification and improvement. The modifications and improvements fall into two general categories:

(1) Changes which enhance the efficiency in generating numerical results;

(2) Changes which expand the capability in simulating various types of missions.

The first category includes the program-logic clean-up modifications required to produce a presentable, export-version of the trajectory computer program from the workhorse-version used daily by the authors.

The trajectory computer program is called HILTOP, an acronym for Heliocentric Interplanetary Low-Thrust Trajectory Optimization Program. The program is documented (Reference 1) and is available from NASA upon request. HILTOP is referred to as TOPCAT in References 2 and 3 ; however, its name has been officially changed from TOPCAT to HILTOP so that it may be distinguished from the original computer program developed by Princeton University, which is called TOPCAT and is described in Reference 4. HILTOP, which is felt to be a significantly improved version of the original Princeton-developed TOPCAT, has generated the solar-electric trajectory and performance data in the above-cited References 2 and 3 and also in Reference 5 . HILTOP is a double-precision program written in FORTRAN IV and designed for the IBM 360 computer.

The first category of changes mentioned above also includes the modifications to the single-precision, FORTRAN IV, IBM 360 computer program ADMAP, an acronym for Automatic Data Manipulation Program. This program, which generated the graphical format of References 2 and 3 , required modifications in order to produce the format of Reference 5 . ADMAP is described in Reference 6 and is available from the authors upon request.

A discussion concerning the specific modifications which were made to the trajectory program HILTOP is given in the following paragraphs.

HILTOP was modified to permit the option of using secant (finite-difference) partial derivatives, as opposed to analytical derivatives, of the hunting procedure dependent variables with respect to the independent variables. This permits the verification of the validity of the analytic partials, and also permits alteration of the mathe-

matical model without having to also derive the corresponding analytic partials. This modification was crucial in allowing converged optimal solutions to be obtained for many highly-sensitive solar-electric propulsion missions, such as solar probes and rendezvous missions to the asteroid Ceres and the comet Encke. The analytic partial derivatives were found to be of insufficient accuracy for these missions.

Although HILTOP contains analytic ephemerides for the major planets, it, originally had no capability for specifying the ephemeris of any other astronomical body. In order to simulate missions to asteroids and comets, the capability of specifying the ephemeris of an arbitrary body was added to the program. The arbitrary body is assumed to move in a stationary elliptical orbit which is specified by inputing the standard conic elements a, e, i, Ω, ω, and t_p to the program.

In order to investigate optimum extra-ecliptic missions using low-thrust propulsion, it was necessary to incorporate new trajectory end-conditions into HILTOP. Since some difficulty was encountered while undertaking this task, it is worthwhile to summarize the development of the extra-ecliptic boundary conditions. In the discussion which follows, a basic knowledge of optimum space flight theory is assumed. The relevant portion of the basic transversality equation (evaluated at the final time) for the problem at hand is

$$\Lambda \cdot d\dot{R} - \dot{\Lambda} \cdot dR = 0$$

where R is the spacecraft's heliocentric position vector and Λ is the primer vector, along which the thrust vector is optimally directed. At the final time, the spacecraft has a heliocentric radius r, speed v, and flight path angle γ, and osculating orbital elements consisting of inclination i, ascending node angle Ω, and argument of position ω, which is the angle from the line of nodes to the spacecraft. The variations of the final spacecraft state dR and $d\dot{R}$ may be expressed as

$$dR = (R/r)\,dr + (h \times R)\,d\omega + (k \times R)\,d\Omega + (\ell \times R)\,di$$

$$d\dot{R} = (\dot{R}/v)dv + (h \times \dot{R})\,d\omega + (k \times \dot{R})\,d\Omega + (\ell \times \dot{R})di + (-h \times \dot{R})\,d\gamma$$

where h, k, and ℓ are unit vectors being parallel to, respectively, the spacecraft's final angular momentum, the z-axis, and the line of nodes of the final extra-ecliptic orbit and the ecliptic. The ecliptic is taken to coincide with the x-y plane.

The problem of interest assumes that the following four state end-conditions are specified as follows:

$r = 1$

$v = 1$

$\gamma = 0$

i = specified value of extraecliptic inclination

This implies that two transversality conditions are required in order to make up the complete set of six required end conditions defining the final state of the spacecraft. The final spacecraft state therefore corresponds to extraecliptic ballistic flight in the unit sphere about the sun (implying a periodic Earth-communications-vector). The above four state end-conditions imply that dr, dv, $d\gamma$, and di vanish, so that the variations become

$$dR = (h \times R)\, d\omega + (k \times R)\, d\Omega$$

$$d\dot{R} = (h \times \dot{R})\, d\omega + (k \times \dot{R})\, d\Omega$$

Substituting these relations into the basic transversality equation yields

$$\Lambda \cdot ((h \times \dot{R})d\omega + (k \times \dot{R})\, d\Omega) - \dot{\Lambda} \cdot ((h \times R)d\omega + (k \times R)\, d\Omega) = 0$$

Since $d\omega$ and $d\Omega$ are independent variations, their coefficients in the above relation are required to vanish:

$$\Lambda \cdot (h \times \dot{R}) - \dot{\Lambda} \cdot (h \times R) = 0$$

$$\Lambda \cdot (k \times \dot{R}) - \dot{\Lambda} \cdot (k \times R) = 0$$

Using a handbook vector formula allows these transversality conditions to be expressed as follows:

$$h \cdot ((\dot{R} \times \Lambda) - (R \times \dot{\Lambda})) = 0$$

$$k \cdot ((\dot{R} \times \Lambda) - (R \times \dot{\Lambda})) = 0$$

Recognizing $(\dot{R} \times \Lambda) - (R \times \dot{\Lambda}) = C$ as the well-known vector constant-of-the-motion allows the two extraecliptic transversality conditions to be expressed simply as

$$h \cdot C = 0$$

$$k \cdot C = 0$$

the latter of which is recognized as the transversality condition which arises in two-dimensional problems when the travel-angle is optimized. These two conditions taken together simply say that, for optimum flight, the constant vector C should be parallel to the line of intersection of the initial and final orbit planes, the initial orbit plane being the ecliptic.

In practice, it was discovered that these simple dot-products $h \cdot C$ and $k \cdot C$ strongly resist vanishing on the computer, and therefore a study was undertaken to find an equivalent pair of transversality conditions which possess favorable convergence properties on the computer. After several unsuccessful attempts, the search for such a pair of transversality conditions was abandoned, and instead resort was made to a particular problem formulation in which all six of the end conditions, including the four conditions involving the final spacecraft state, are replaced by an equivalent set of six end conditions in such a manner that the two transversality conditions become absorbed in all six of the new end conditions.

The six new end conditions consist of the position and velocity differences between the actual final spacecraft state (on a guessed trajectory) and the desired final spacecraft state, where the desired final state is not explicitly known but may be expressed implicitly in terms of the transversality result that the constant vector C lie in the (final) line of nodes. The actual final spacecraft state R_a, $\dot{R}_a$ is known on any given nonoptimum trajectory, for which the constant vector $C=(C_x, C_y, C_z)$ points in some arbitrary direction. The desired final spacecraft state is then constructed by considering a fictitious spacecraft which is travelling in a final circular orbit having unit radius and inclined the desired amount to the ecliptic. The ascending node angle Ω of this circular orbit is computed as

$$\Omega = \tan^{-1} (C_y / C_x)$$

since this is the desired transversality result as C_z vanishes. The fictitious spacecraft's unit circular orbit may thus be pictured as being anchored such that its line of nodes with respect to the ecliptic coincides with the projection of C into the ecliptic.

It remains to express the fictitious spacecraft's argument of position ω in terms of the desired transversality result that C lie parallel to the line of nodes of the final orbit. Let the fictitious spacecraft's final state be expressed as R_f and $\dot{R}_f$, where these vectors are orthogonal and have unit magnitudes. For the desired result that C lie along the line of nodes, the angle between C and R_f should be ω, which implies the angle between C and $\dot{R}_f$ would be $\omega + \pi/2$. Then $C \cdot R_f = |C| \cos \omega$ and $C \cdot \dot{R}_f = - |C| \sin \omega$. Substituting the expression for C into these dot-products yields

$$C \cdot R_f = (\dot{R}_f \times \Lambda) \cdot R_f$$

$$C \cdot \dot{R}_f = - (R_f \times \dot{\Lambda}) \cdot \dot{R}_f$$

where two terms have vanished due to self-cross-products. The primer Λ and its derivative $\dot{\Lambda}$ are evaluated at the actual trajectory endpoint, since they are associated with C which corresponds to the actual trajectory. Applying a handbook vector formula to the right-hand side of the above two equations yields

$$C \cdot R_f = (R_f \times \dot{R}_f) \cdot \Lambda = |C| \cos \omega$$

$$C \cdot \dot{R}_f = (R_f \times \dot{R}_f) \cdot \dot{\Lambda} = - |C| \sin \omega$$

The quantity immediately to be solved-for is ω, and the vectors R_f and $\dot{R}_f$ are ultimately sought. To solve for ω, the fictitious spacecraft's angular momentum (at the final time) $R_f \times \dot{R}_f$ may be replaced by any vector h_f normal to the fictitious spacecraft's orbital plane in the direction of $R_f \times \dot{R}_f$. Such a vector is $h_f = (m_x \sin i, m_y \sin i, \cos i)$ where

$$m = (m_x, m_y, m_z) = k \times C / |k \times C|.$$

Then

$$\omega = \tan^{-1} (-h_f \cdot \dot{\Lambda} / h_f \cdot \Lambda)$$

and finally the fictitious spacecraft's state, or desired state, is obtained from

$$R_f = \begin{bmatrix} \cos \omega \cos \Omega - \sin \omega \sin \Omega \cos i \\ \cos \omega \sin \Omega + \sin \omega \cos \Omega \cos i \\ \sin \omega \sin i \end{bmatrix}$$

and $\dot{R}_f = dR_f/d\omega$ since $\dot{\omega} = 1$. The six extra-ecliptic endconditions for optimum flight which are coded into HILTOP are then $R_a - R_f$ and $\dot{R}_a - \dot{R}_f$, which turn out to have very favorable convergence properties.

Another significant modification which was made to HILTOP consisted of incorporating into the program the necessary logic for simulating optimum missions for which the total accrued thrusting time of the low thrust propulsion system is constrained to be less than or equal to a specified value. The analytical theory was implemented by declaring the accrued propulsion time τ as a new state variable having the differential equation $\dot{\tau} = h_\sigma$, where $h_\sigma = 1$ during thrusting and 0 during coasting. The adjoint variable to τ is λ_τ, having the differential equation $\dot{\lambda}_\tau = 0$. Application of Pontryagin's Maximum Principle leads to the result that a term $\nu \lambda_\tau / a_o \gamma$ should be added to the old thrust switching function σ (old) $= \lambda - \nu \lambda_\nu / c$ to produce a more general switching function:

$$\sigma = \lambda - \nu\lambda_v/c + \nu\lambda_\tau/a_o\gamma$$

where the quantities are defined in Reference 2. λ_τ is a constant which is set to zero when the thrusting time is unconstrained and which is chosen by iteration to have a value such that the specified maximum thrusting time is satisfied whenever the unconstrained optimum thrusting time exceeds the specified maximum thrusting time. Due to an abundance of higher priority work, this task was halted while still in the software-checkout stage. A few attempts to obtain an optimum constrained-thrusting-time solar electric propulsion trajectory were unsuccessful, and it is not known at this time if this failure is attributable to faulty program logic or to some difficulty inherent in the problem.

Some minor HILTOP modifications were necessary in order to satisfy the data generation requirements discussed in Section III. The options of inputing launch vehicle coefficients and engine efficiency coefficients were added to the program, as was the option of simulating tilted solar arrays to maintain maximum power near the sun. The option of suppressing the predictor-corrector (PC) numerical integration in favor of Runge-Kutta (RK) numerical integration proved to be very helpful in obtaining solar probes. (When the PC integration is used for highly-sensitive missions, its RK starter introduces a slight numerical inconsistency with respect to the PC formulation, making the iterator's partial derivative matrix inaccurate.) HILTOP was also modified to allow the generation of magnetic tapes containing trajectory information compatible for input to the ASTEA program, which is described in Reference 7.

A major task consisted of preparing and documenting an export-version of HILTOP. Considerable effort was expended in cleaning-up, simplifying, and improving the efficiency of the program logic. Several large subroutines were broken into smaller ones for clarity of operation and to reduce compilation time. The standard printout format was examined and improved, as were all program diagnostic messages. The program was documented in Reference 1, which is being published concurrently with this report.

II. DATA GENERATION

The solar electric mission analysis data generated under Contract NAS5-20126 was devoted mostly to the support of other NASA studies. The information generated is displayed graphically in Section III of this report and in Reference 5. Early in the year (1970), some solar probe mission data generation was begun assuming the launch vehicles investigated in References 2 and 3, but this data, which is too sparse for publication at this time, was shelved as the higher-priority requests discussed in Section III were received. Since the current study of solar-electric mission applications will continue through the coming year (1971), this solar probe study will hopefully be taken off the shelf and brought to completion.

Before the data described in Section III could be generated, starting solutions had to be obtained with the trajectory computer program, HILTOP. As experienced low-thrust trajectory analysts are aware, this type of activity is no trivial matter, especially for difficult or sensitive missions such as those under consideration. Substantial effort was devoted to obtaining starting solutions for solar probe and extra-ecliptic missions, for Mercury orbiter, Ceres rendezvous, and Pluto flyby missions, and for rendezvous missions to the comets D'Arrest and Encke. Mercury orbiter missions remain so difficult to generate, at this time, that only a few converged optimal solutions have been obtained. Several programming improvements, mentioned in Section I, have alleviated some of the difficulty in generating data for other missions, and hopefully the Mercury orbiter missions will yield to attack in the continuing effort.

One particular numerical difficulty is extremely troublesome when it arises during the generation of a given set of data on the computer. The difficulty arises when an infinitesimal coast phase appears along a trajectory, and all subsequent trajectories become locked onto this class of trajectories having a very small coast phase, which prevents convergence to the optimal solution. Some effort has been devoted to solving this problem, but to no avail. Other low thrust trajectory analysts have also experienced this difficulty, using other computer programs. The alleviation of this difficulty would tend to make the generation of optimum low thrust trajectory data somewhat of a straightforward matter, once the necessary starting solutions are obtained.

III. SUPPORT OF OTHER NASA STUDIES

It has been the policy under the contract NAS5-20126 to cooperate in the performance of other NASA funded studies of solar electric propulsion by providing, whenever possible, performance data for specified missions. This policy is felt to be in keeping with the primary objective of the contract to define appropriate applications and requirements of solar electric propulsion. Since it is unrealistic to expect that all applications of interest can be studied in detail in this contract, it is clear that significant advances in the field of solar electric propulsion mission analysis will best be achieved through a cooperative effort of the many analysts involved throughout the country. The following paragraphs and figures describe the work accomplished toward this end during the subject contract.

Prior to the publication of Reference 3, Mr. J. P. Mullin of the Electric Thruster Systems Office, NASA/OART, requested information concerning the effect of sub-optimal power on the performance requirements of selected missions to the outer planets. Specifically, he was interested in 900 day Jupiter, 1800 day Saturn, and 3600 day Uranus orbiter missions. In response to this request, a set of graphs was prepared for each of the three missions showing the behavior of 15 selected parameters as a function of the reference power level. This information is reproduced here in Figures 1 - 9. The definitions of the symbols appearing on the graphs are as follows:

M_o - initial spacecraft mass

M_{ps} - mass of propulsion system

M_p - low thrust propellant

M_r - retro step mass, propellant and tankage

M_N - net spacecraft mass

T_p - propulsion time

P_{MAX} - maximum power generated

c - jet exhaust speed

T_o - thrust generated at 1 AU

R_{max} - maximum distance from sun

R_{min} - minimum distance from sun

θ - travel angle

$V_{\infty d}$ - departure excess speed

$V_{\infty a}$ - arrival excess speed

V_r - retro incremental velocity

Additional information that has been requested by Mr. Mullin throughout the year have included a trajectory profile of a typical fully optimized solar electric Mercury orbiter mission (Fig. 10), a definition of the minimum flight times to Pluto for a fixed spacecraft weight and specified launch vehicle in each launch opportunity that exists in the 1975-1990 time period (Fig. 11), and a performance comparison of the capabilities of the TAT(3C)/Delta/TE 364 vehicle for Mars and Venus orbiter missions with and without a solar electric upper stage (Fig. 12).

Mr. A. L. Friedlander of the Illinois Institute of Technology Research Institute (IITRI) requested solar electric performance data in support of one of his NASA contracts. Specifically, he desired fully optimized performance data for a direct Jupiter orbiter mission and an indirect Neptune orbiter mission as a function of the size and shape of the final orbit about the destination planet. The purpose of the data was to verify analytic scaling equations that had been developed to predict the effect on performance of the selection of the final orbit characteristics. The requested data are presented in Figures 13 and 14, and superimposed over the optimum data are several points evaluated with the scaling equations. The accuracy of the scaling equations is quite remarkable.

One of the specific tasks of the subject contract was to prepare an input for the 1971 edition of the Estimating Factors Book published by the Battelle Memorial Institute

for NASA/OSSA. The data are desired in the form of graphs of net spacecraft mass as a function of flight time with notations of optimum power requirements for each mission investigated. New information provided this year includes data for solar probe missions to 0.05 and 0.1 AU; for 4-burn and 5-burn, 1 AU circular orbit missions inclined 45 degrees and 60 degrees to the ecliptic; for Ceres rendezvous missions with launches in both easy and difficult opportunities; and for the 1982 comet D'Arrest rendezvous mission with arrival at perihelion and at 50 days prior to perihelion. Performance data for two individual launch vehicles were presented. The data are reproduced in Figures 15 - 21.

To ease the trajectory analysis requirements related to a pending NASA contract study, Mr. A. C. Mascy of the Mission Analysis Division, NASA/OART, requested performance and trajectory data for several high energy missions which are important in the exploration of our solar system. During the course of the contract, Mr. Mascy was provided mission data for solar probe, extra-ecliptic, Ceres rendezvous, comet D'Arrest rendezvous and comet Encke rendezvous missions. This data were presented in tabular form as requested. Due to the large quantity of data involved, it will not be reproduced here, but it is presented in graphical form in a companion company report (Ref. 5).

Because an electronic plotter is employed in the preparation of virtually all the graphical data, it was found convenient to prepare a magnetic tape data library. This tape library is available for distribution upon request. To the present time, it has been given to Mr. Don Bartz of JPL and to Mr. Mascy. A multipurpose automatic data manipulation program (ADMAP) has been developed to interrogate this tape and to perform a variety of functions with the data extracted from the tape. Two features presently available with this program are the preparation of the plot tapes that are used by the electronic plotter and the preparation of condensed listings of the optimum trajectory and performance data. This latter feature was employed in the preparation of the tabular data discussed in the preceding paragraph. In addition, requests from Mr. Bartz for these listings have been honored for the sub-optimal power data that has been published in graphical form in Reference 3. The data manipulation program itself has been given to Mr. Mascy for the purpose of interrogating the data tape library on their own computer.

IV. NEW TECHNOLOGY

Reviews of the work carried out under Contract NAS5-20126 were conducted periodically during the course of the contract by the Principal Investigator with particular attention being given to possible contributions to New Technology. A survey of these reviews indicated there is little possibility that applications of the techniques or results exist outside the field of low thrust performance and trajectory analysis. Consequently, it is concluded that no patentable inventions or other contributions to New Technology have been generated under this contract.

$M_{ps} \times 10^2$

$M_r \times 10^2$ $M_o \times 10^3$

$M_p \times 10^2$ $M_N \times 10^3$

Note: Specific mass on all graphs is 30 kg/kw unless otherwise specified.

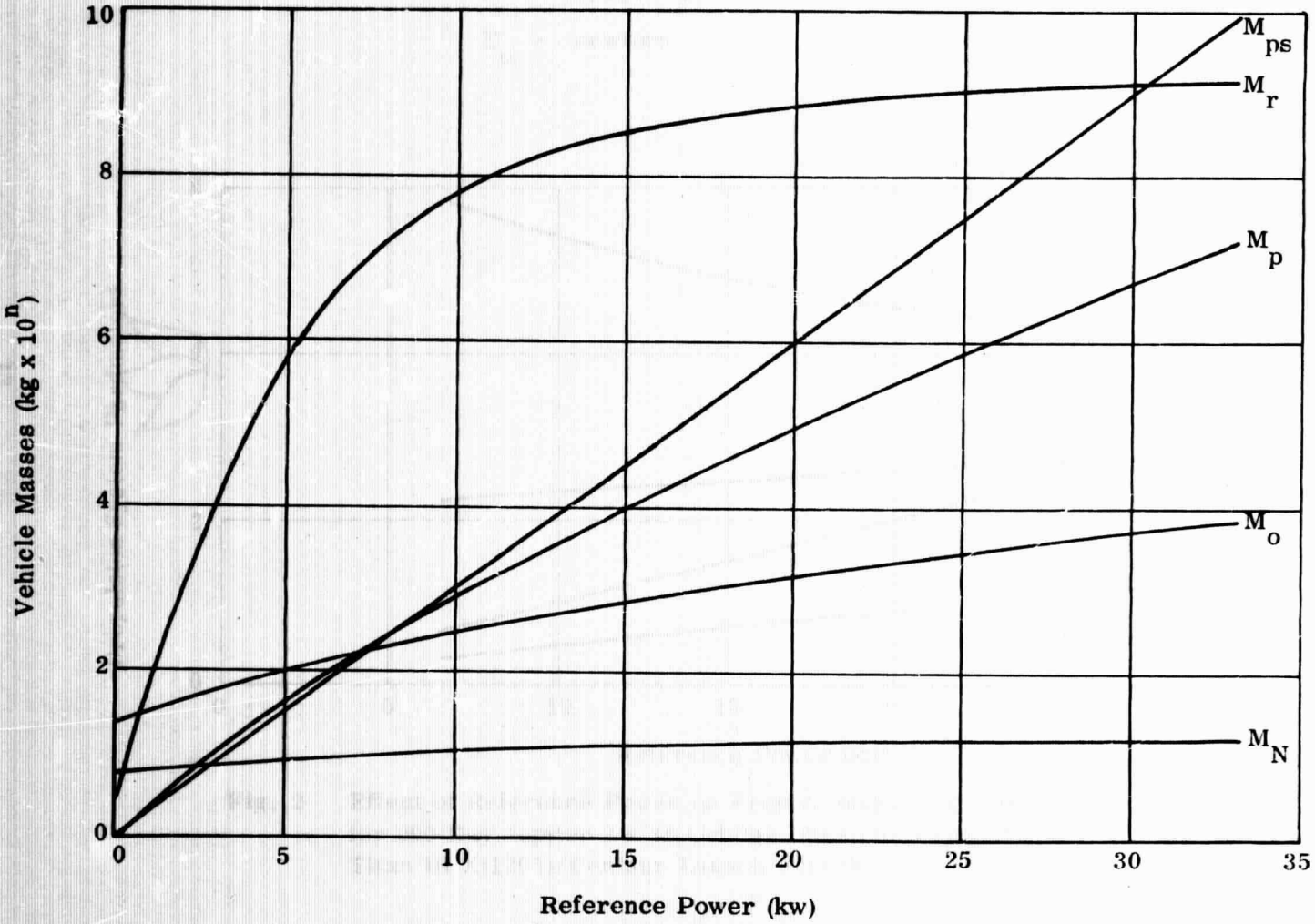


Fig. 1 Effect of Reference Power on Spacecraft Masses for 900 Day Jupiter 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

T_p - days x 10^2

P_{MAX} - kw x 10

c - m/sec x 10^4

T_o - newtons

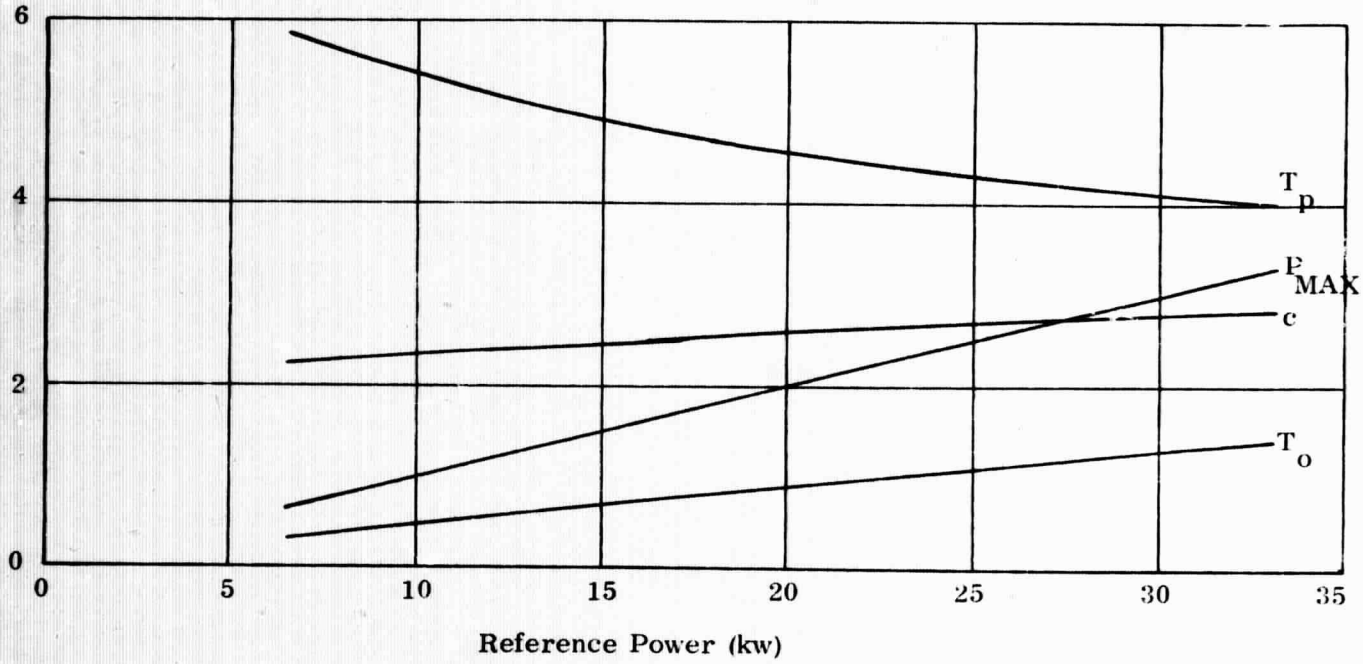


Fig. 2 Effect of Reference Power on Propulsion System Parameters for 900 Day Jupiter 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

$V_{\infty d}$ - m/sec x 10^3

$V_{\infty a}$ - m/sec x 10^3

V_r - m/sec x 10^3

R_{max} - AU

R_{min} - AU

θ - degrees x 10^2

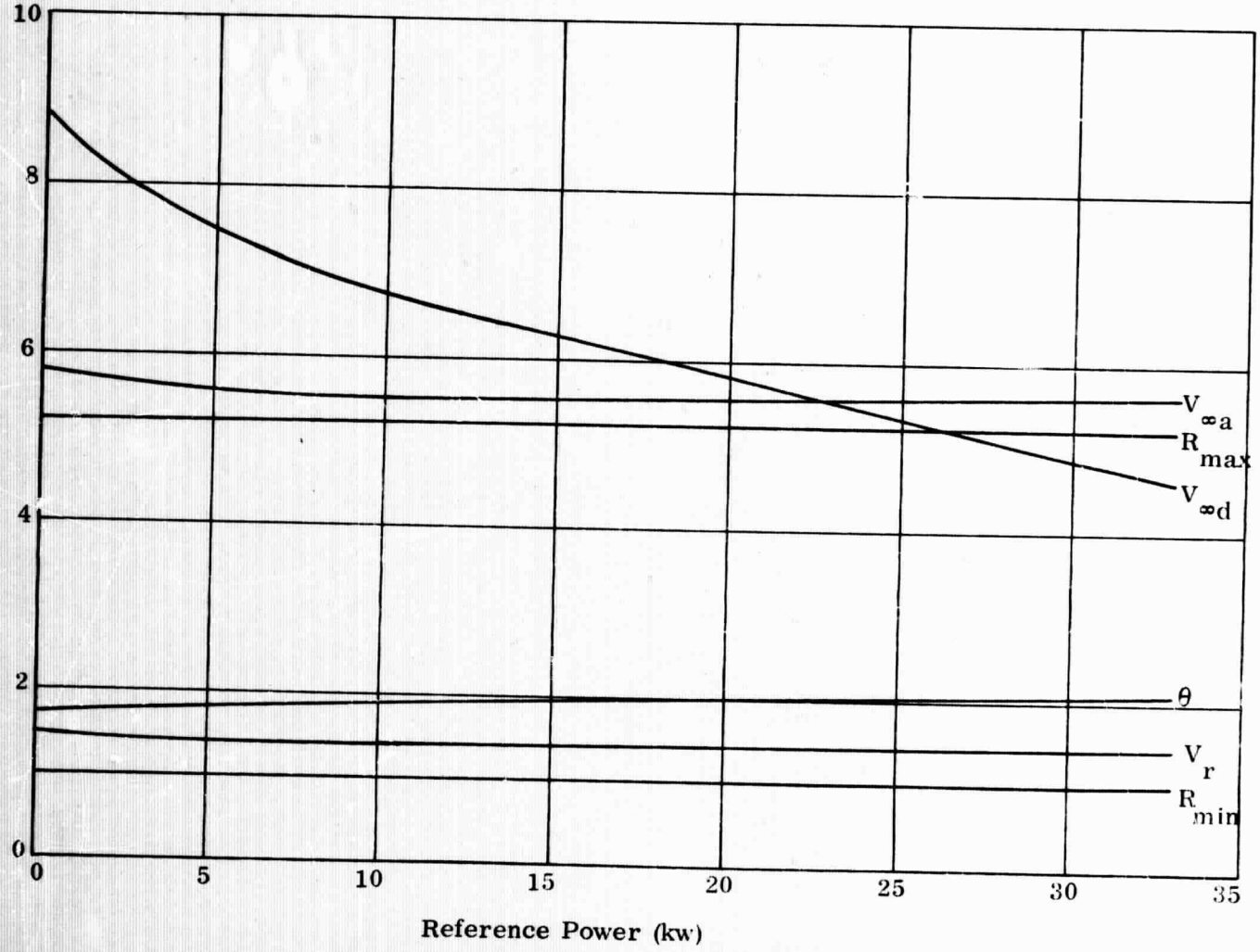


Fig. 3 Effect of Reference Power on Trajectory Parameters for 900 Day Jupiter 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

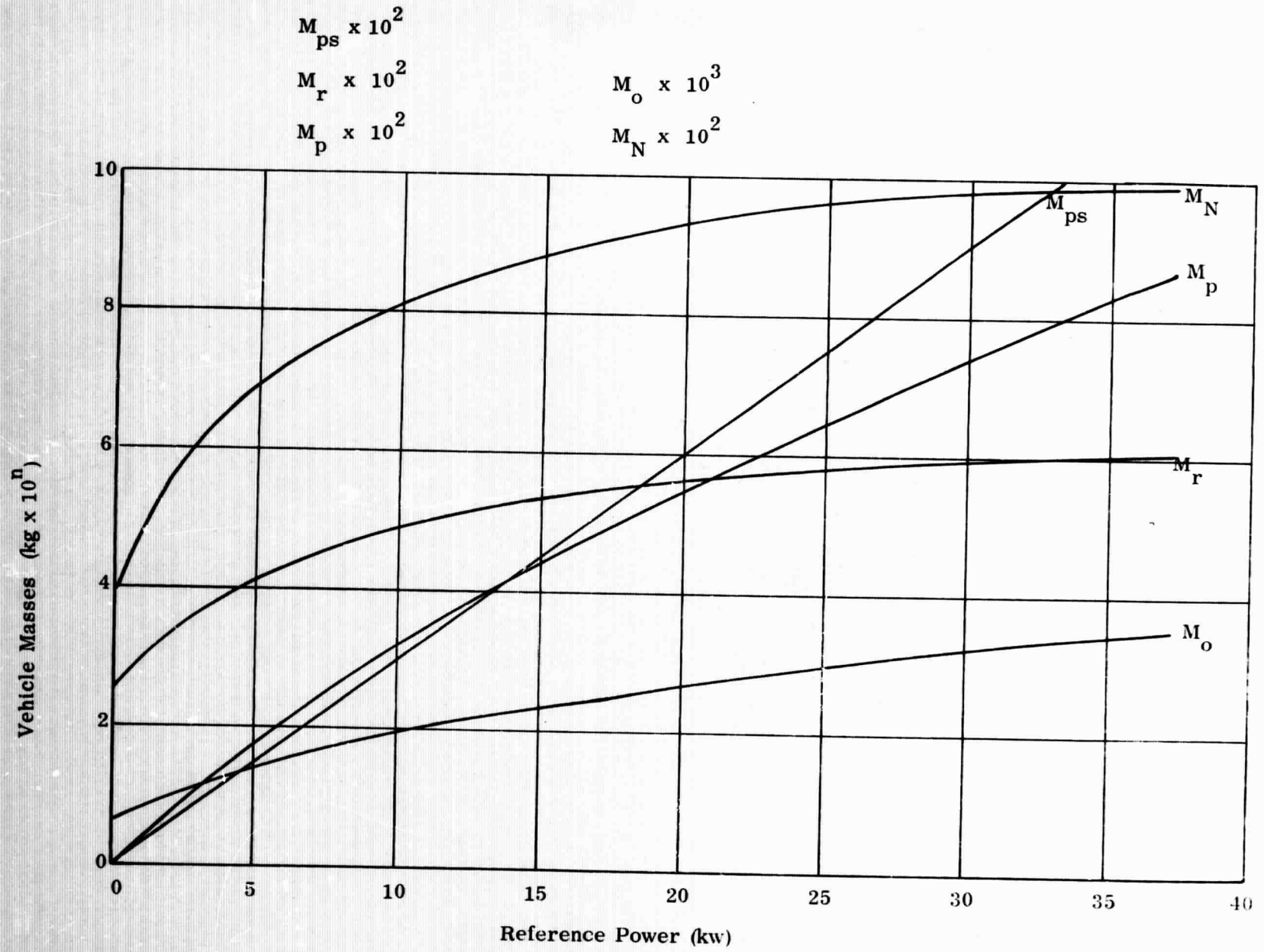


Fig. 4 Effect of Reference Power on Spacecraft Masses for 1800 Day Saturn 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

T_p - days x 10^2

P_{MAX} - kw x 10

c - m/sec x 10^4

T_o - newtons

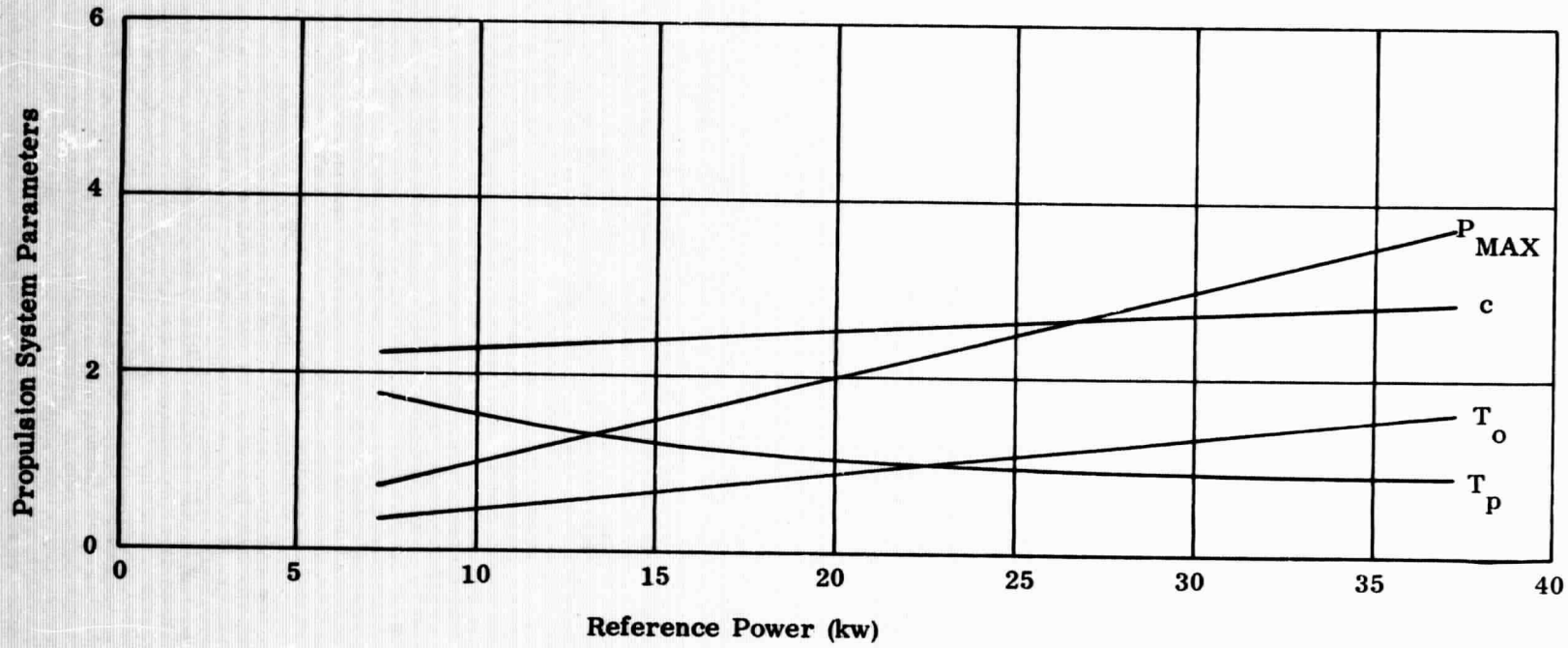


Fig. 5 Effect of Reference Power on Propulsion System Parameters for 1800 Day Saturn 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

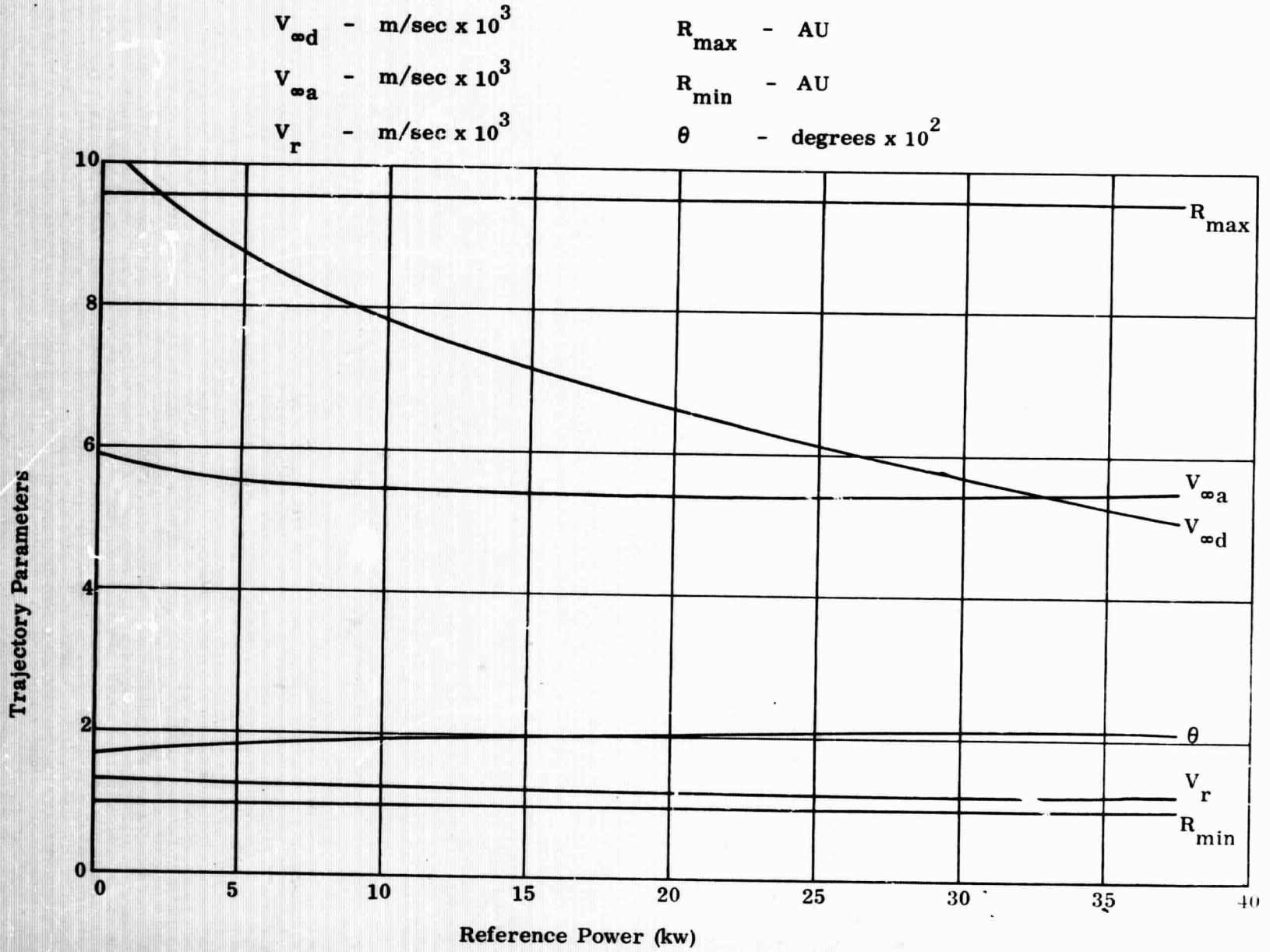


Fig. 6 Effect of Reference Power on Trajectory Parameters for 1800 Day Saturn 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

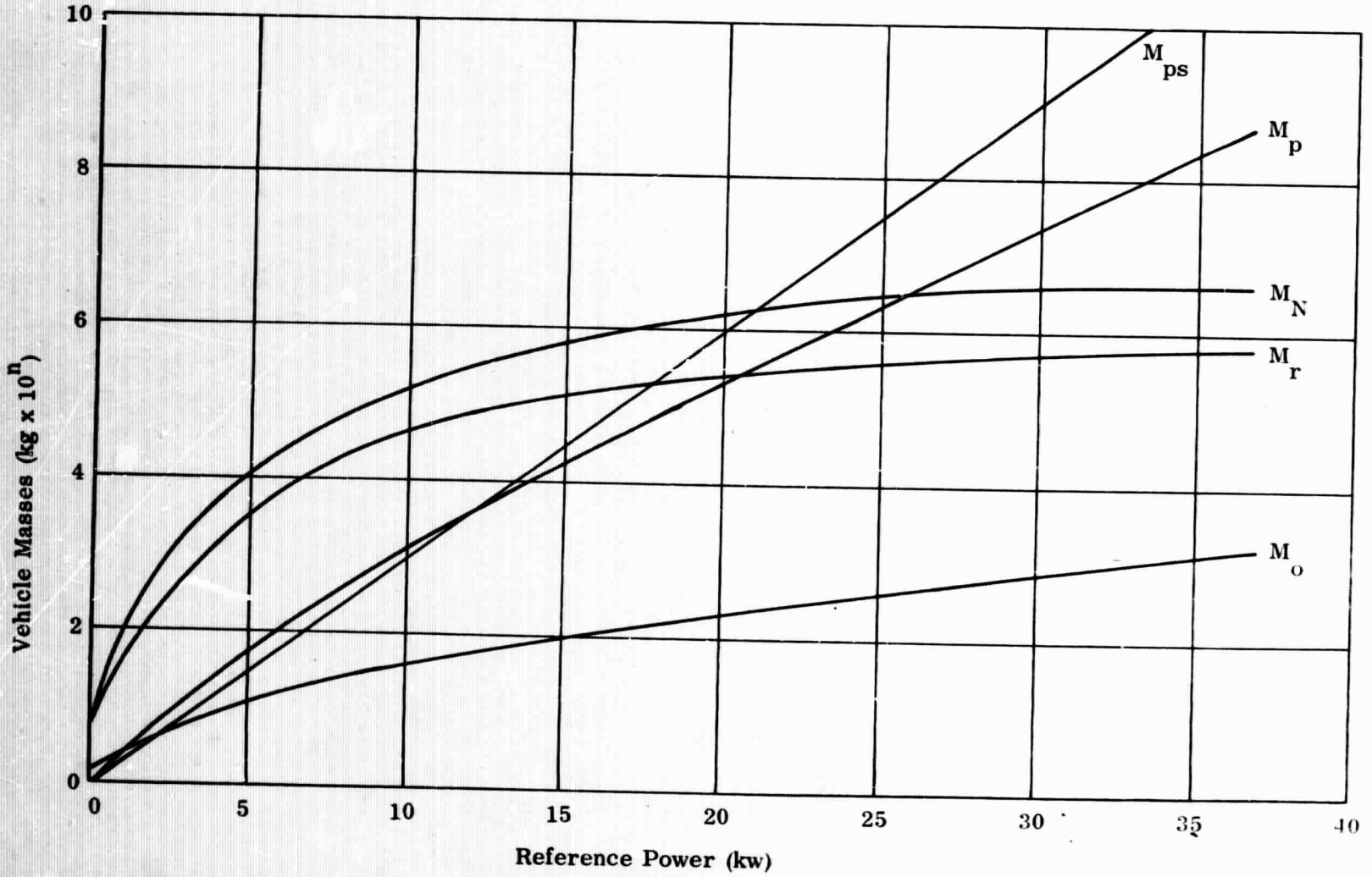


Fig. 7 **Effect of Reference Power on Spacecraft Masses for 3600 Day Uranus 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.**

T_p - days x 10^3

P_{MAX} - kw x 10

c - m/sec x 10^4

T_o - newtons

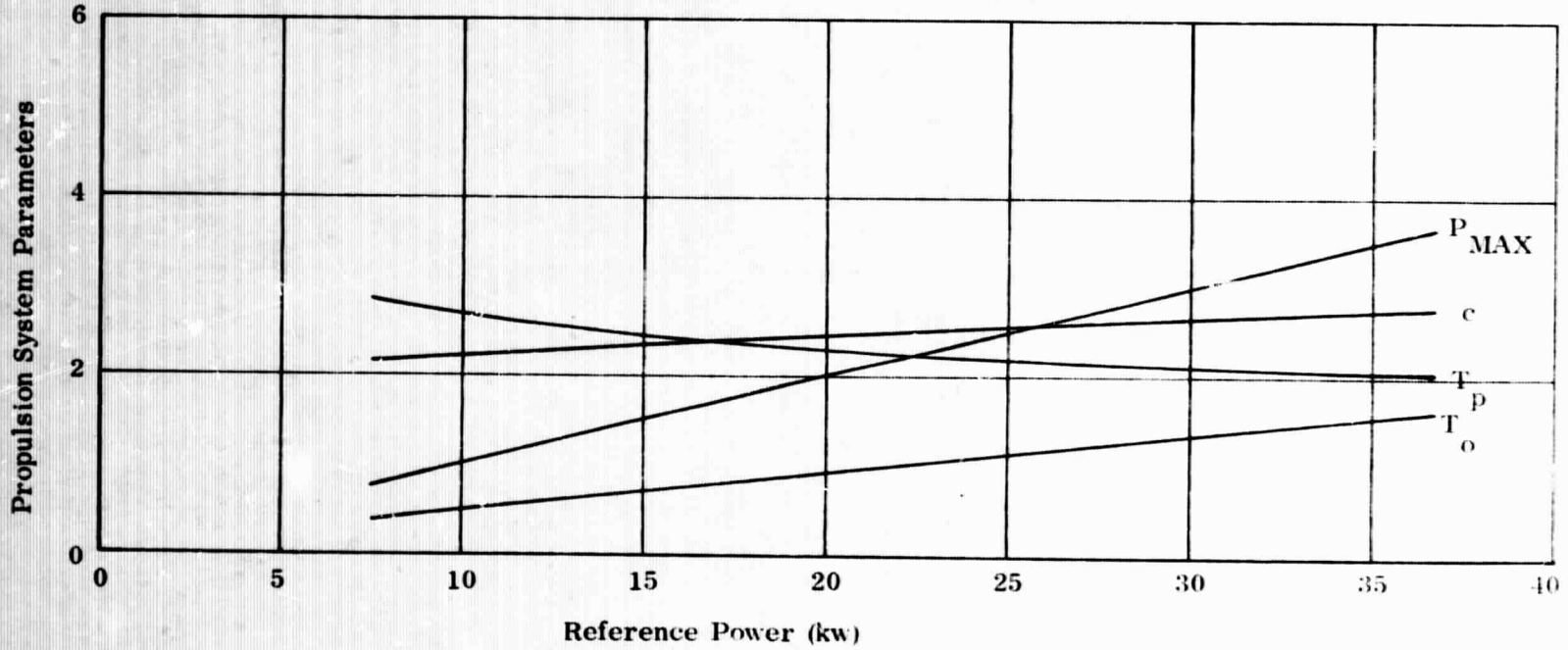


Fig. 8 Effect of Reference Power on Propulsion System Parameters for 3600 Day Uranus 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

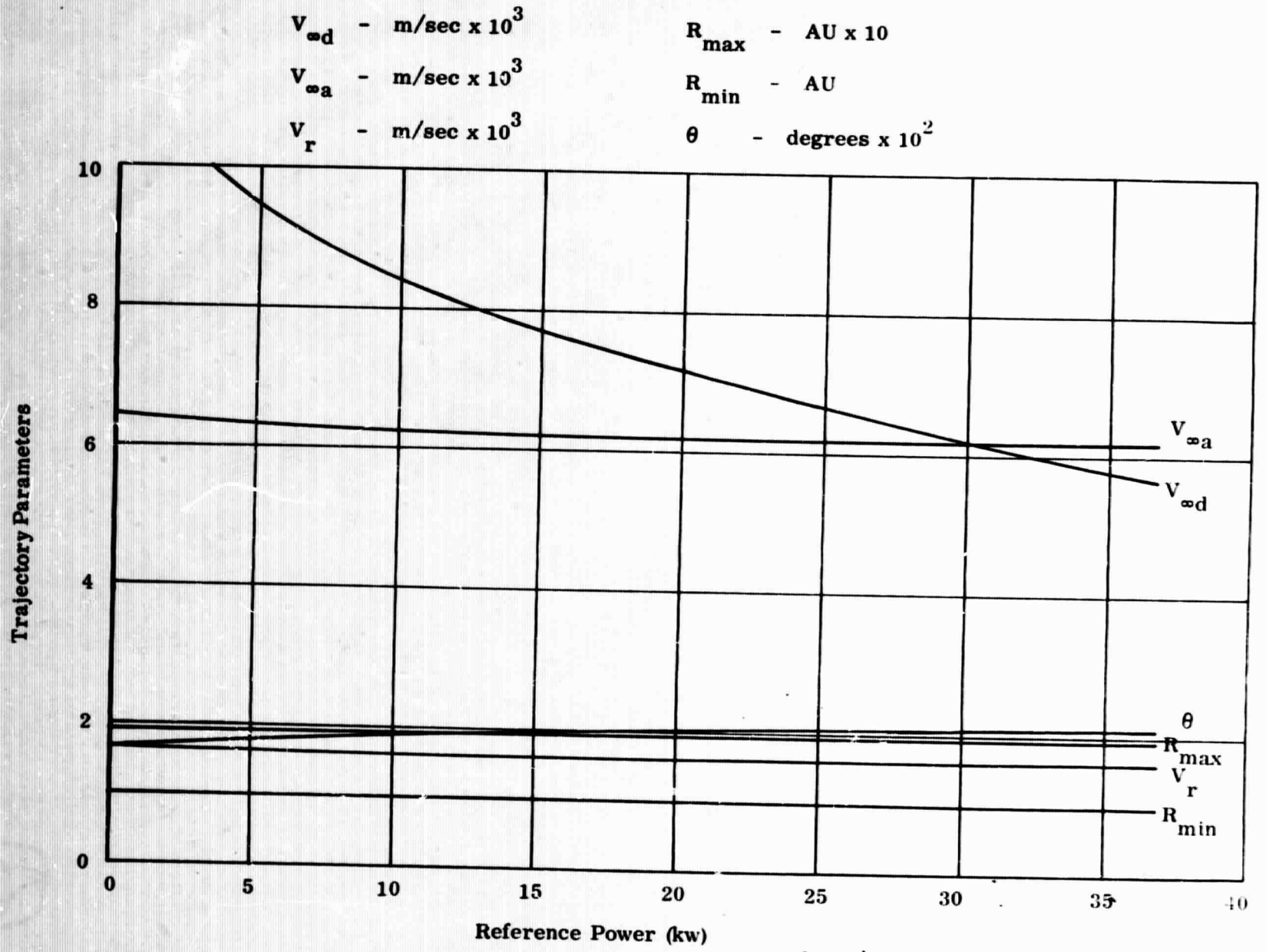


Fig. 9 Effect of Reference Power on Trajectory Parameters for 3600 Day Uranus 2 x 38 Orbiter Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

Numbers represent days from launch.

Arrows represent thrust direction.

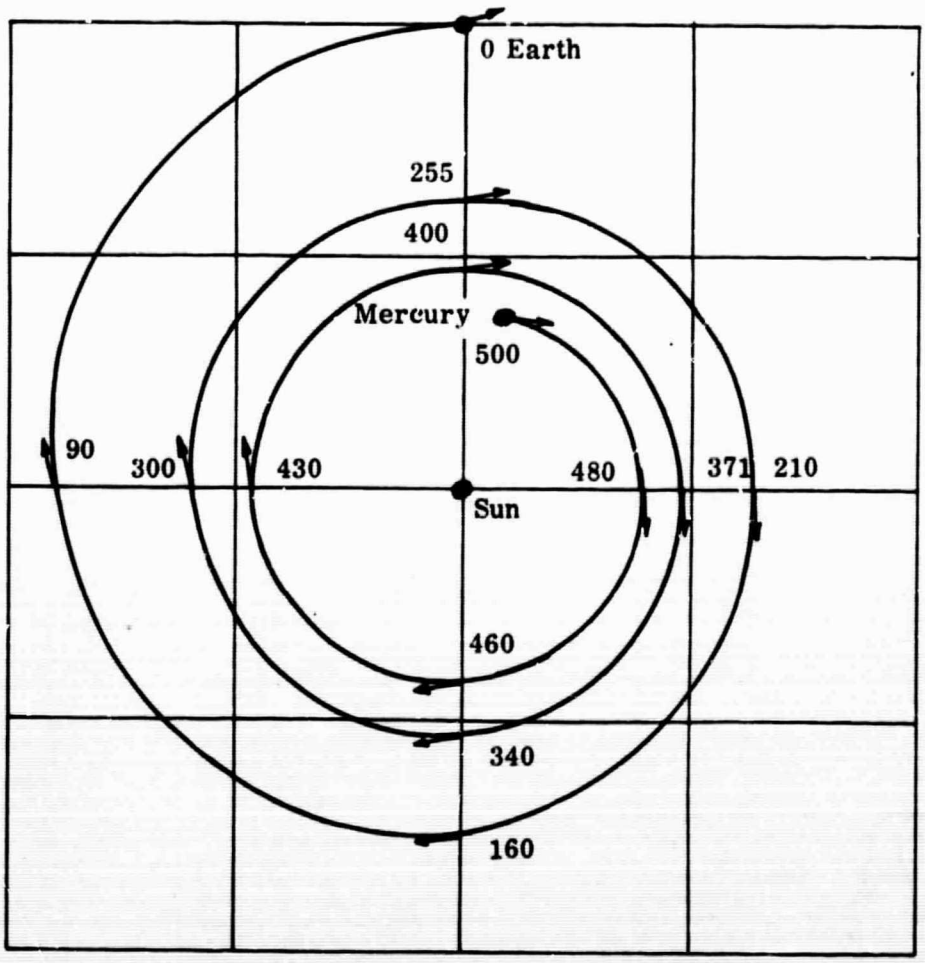


Fig. 10 **Trajectory Profile of a Typical Fully Optimized Solar Electric Propulsion Mercury Orbiter Mission (Ecliptic Projection).**

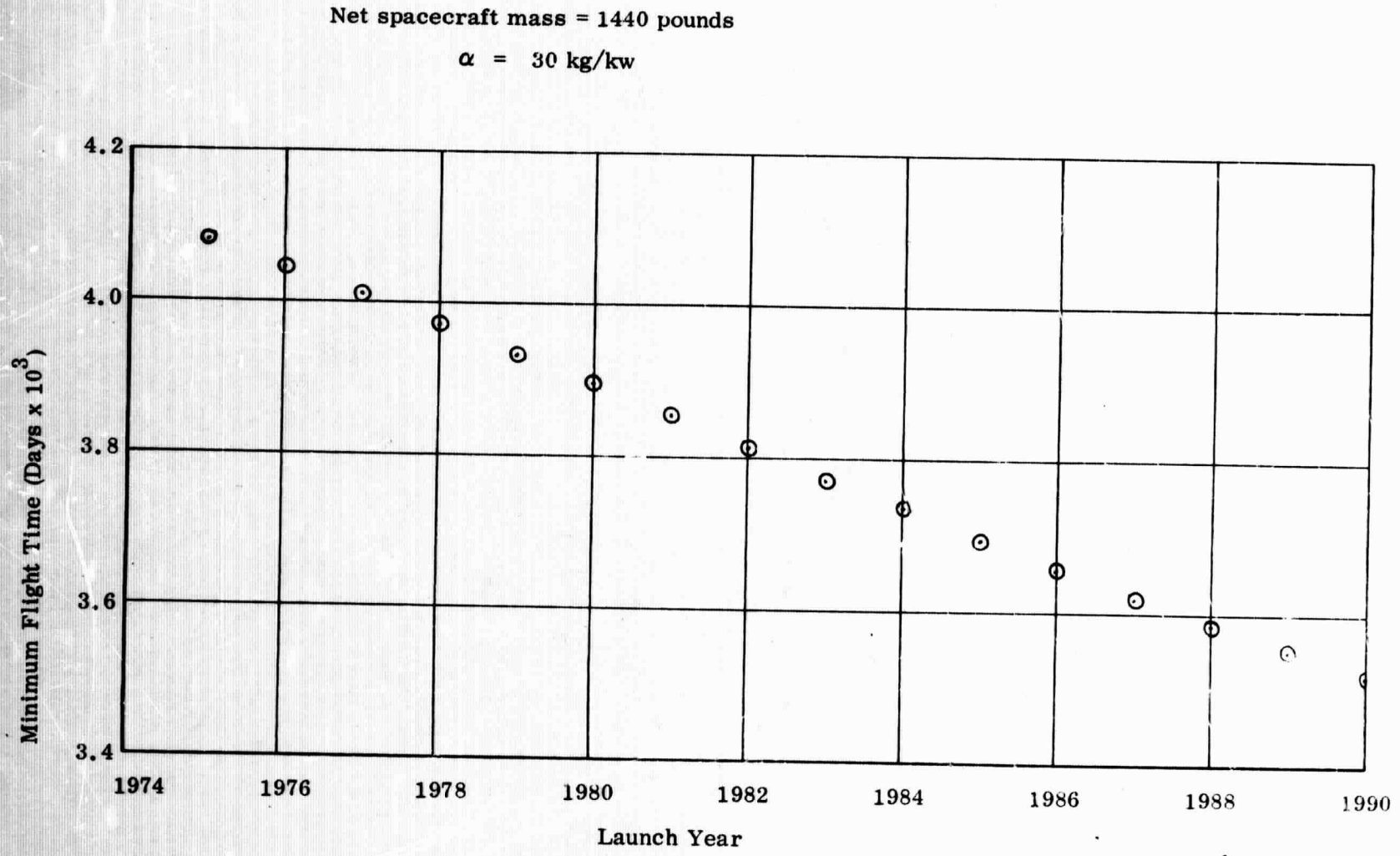


Fig. 11 Minimum Flight Times for Pluto Flyby Missions Using the Titan III X(1205)/Centaur Launch Vehicle.

Capture Orbit

Periapse distance = 1.2 planet radii

Apoapse distance
dashed lines = 1.2 planet radii
solid lines = 22.8 planet radii

α = 30 kg/kw

Low thrust propulsion system jettisoned prior to retro burn.

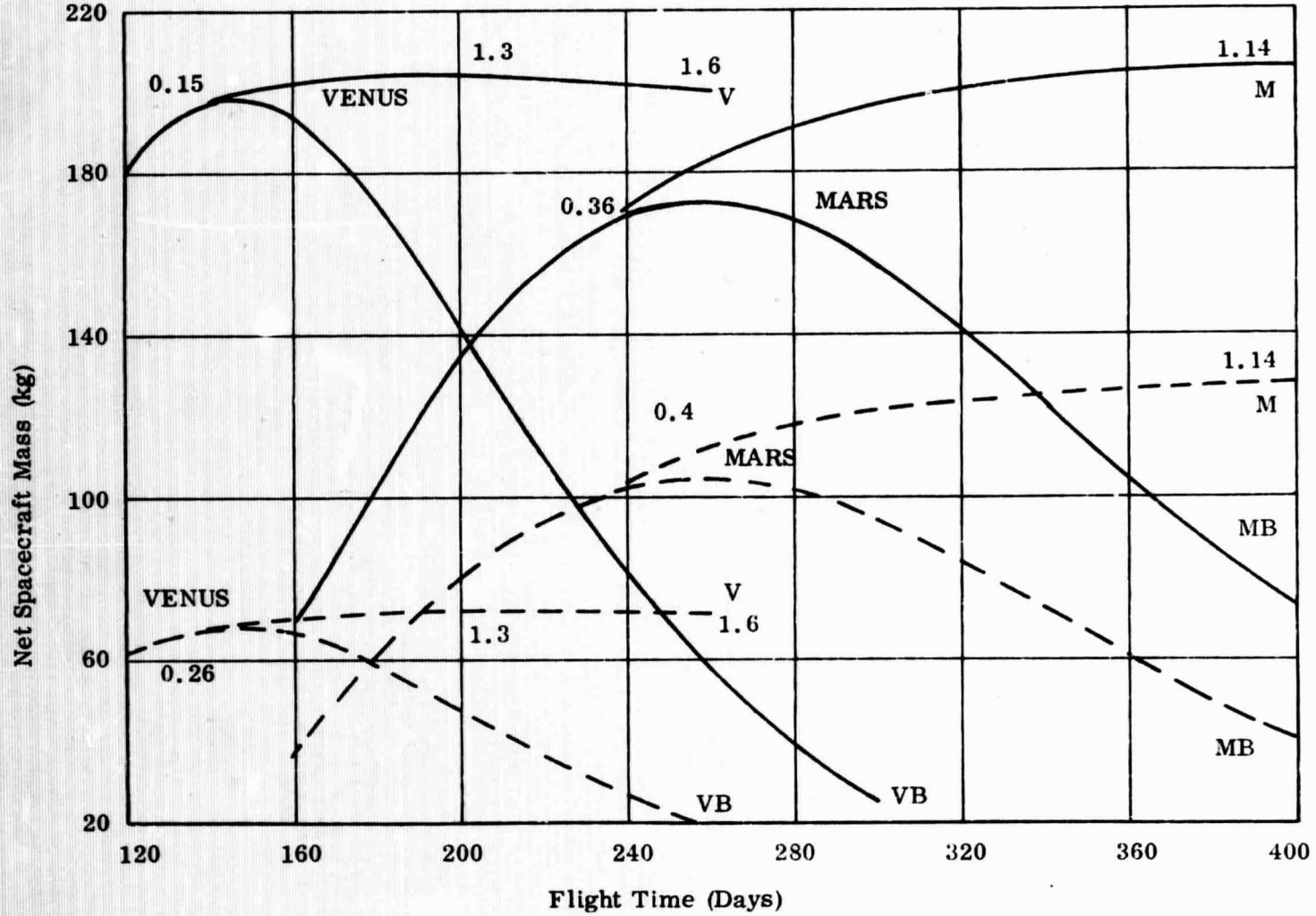


Fig. 12 Performance Comparison for Mars and Venus Orbiter Missions With and Without a Solar Electric Upper Stage Using the TAT(3C)/Delta/TE364 Launch Vehicle.

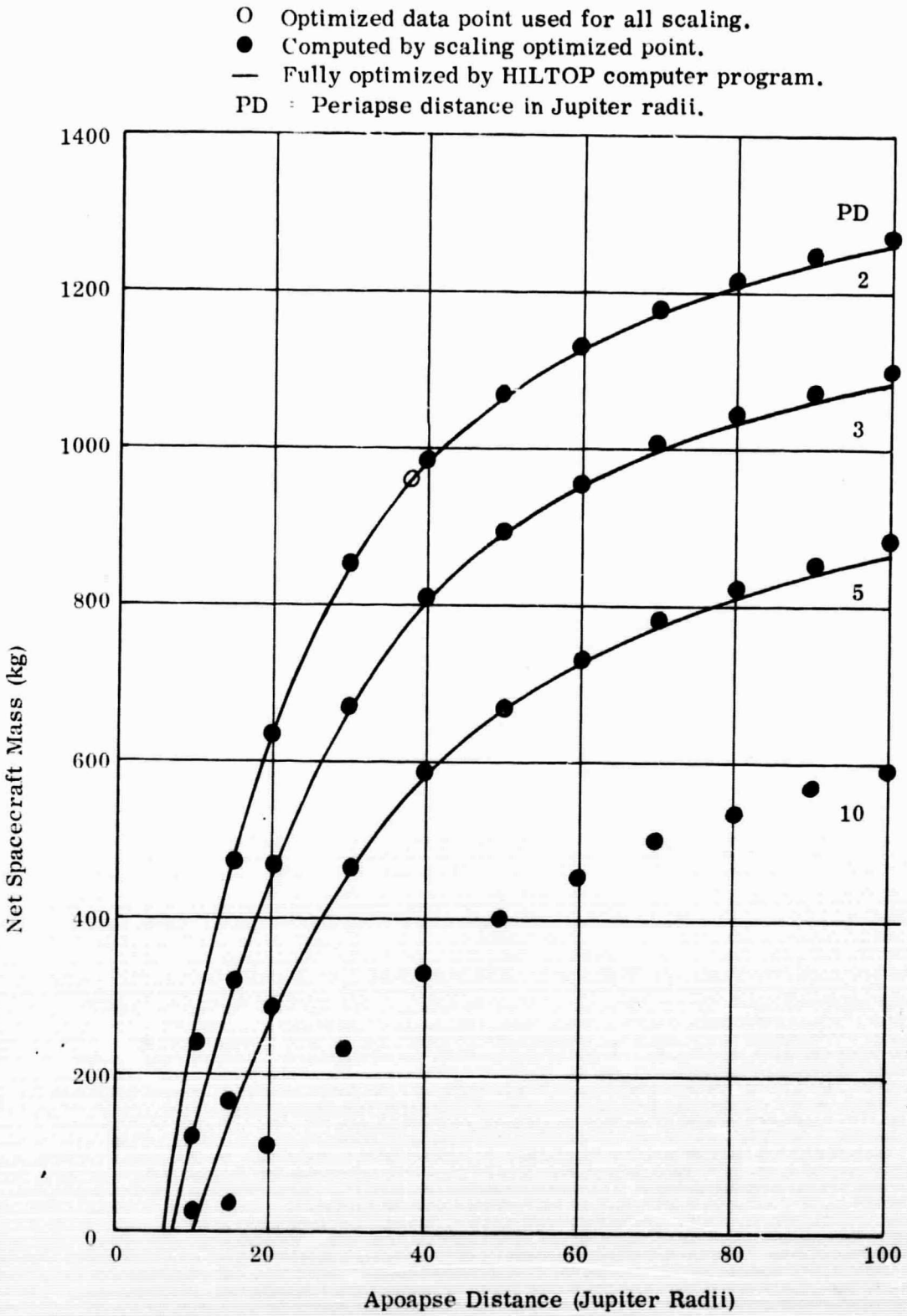


Fig. 13 Net Spacecraft Mass Scaling for 600 Day Mode A Jupiter Orbiter Missions (Powerplant Jettisoned).

● Computed by scaling optimized point.
— Fully optimized by HILTOP computer program.

Titan III X (1207)/Centaur launch vehicle
PD = Periapse distance in Neptune Radii

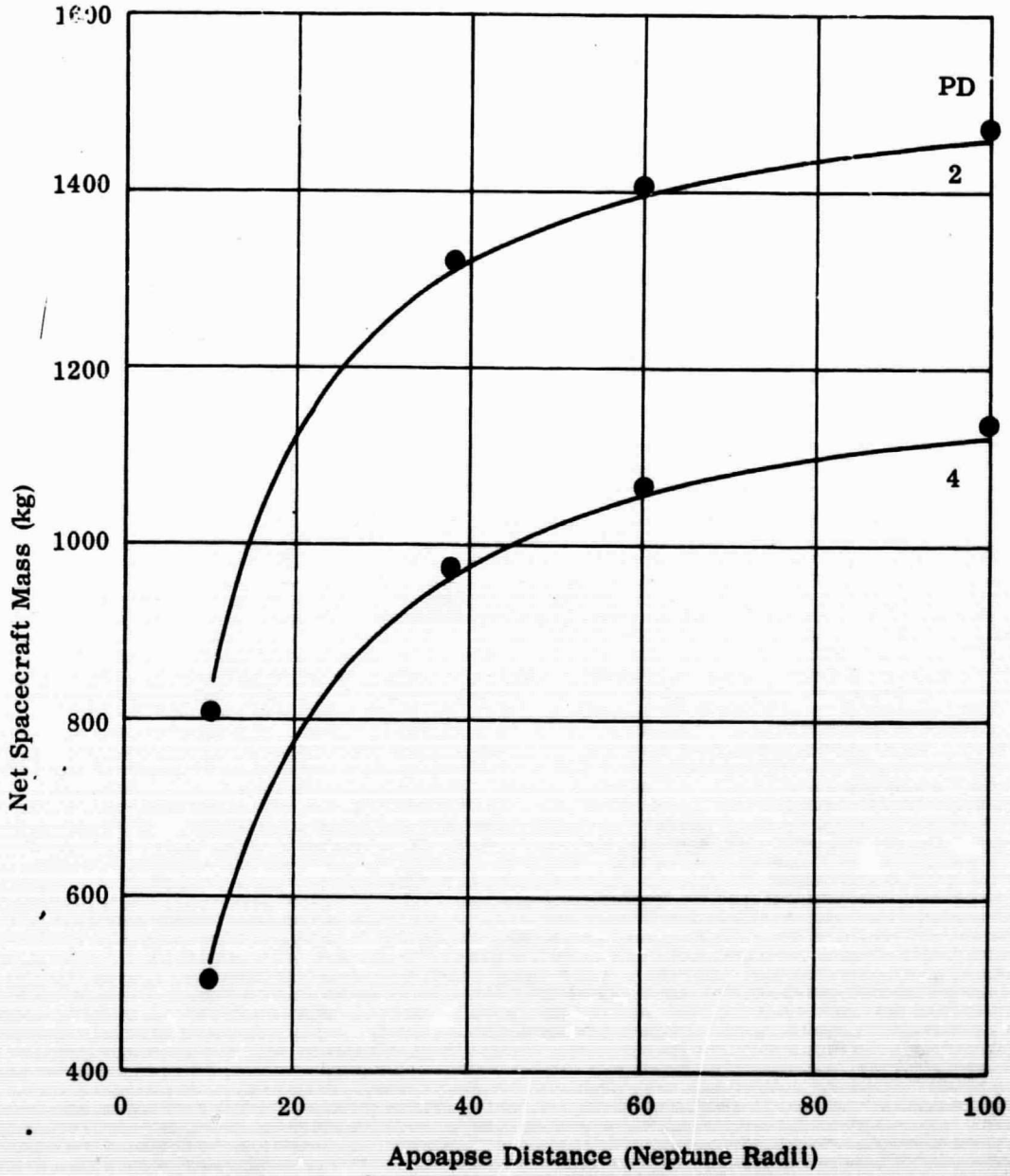


Fig. 14 Net Spacecraft Mass Scaling for 6000 Day Mode B Neptune Orbiter Missions (Powerplant Jettisoned).

α = 30 kg/kw

Numbers represent reference power in kw.

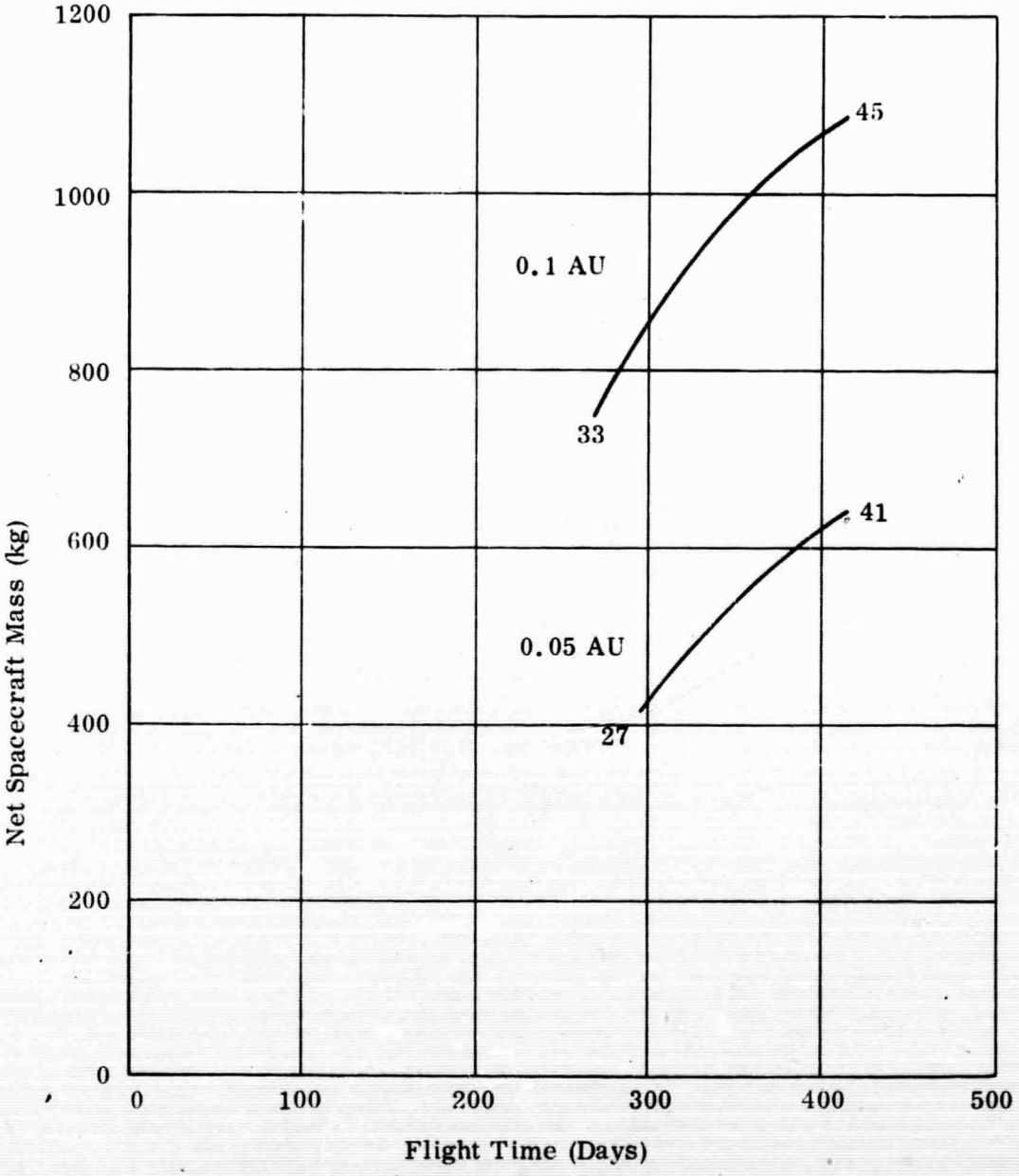


Fig. 15 Performance Requirements for Mode B Solar Probe Missions Using the Titan III D/Centaur Launch Vehicle

α = 30 kg/kw

Numbers represent reference power in kw.

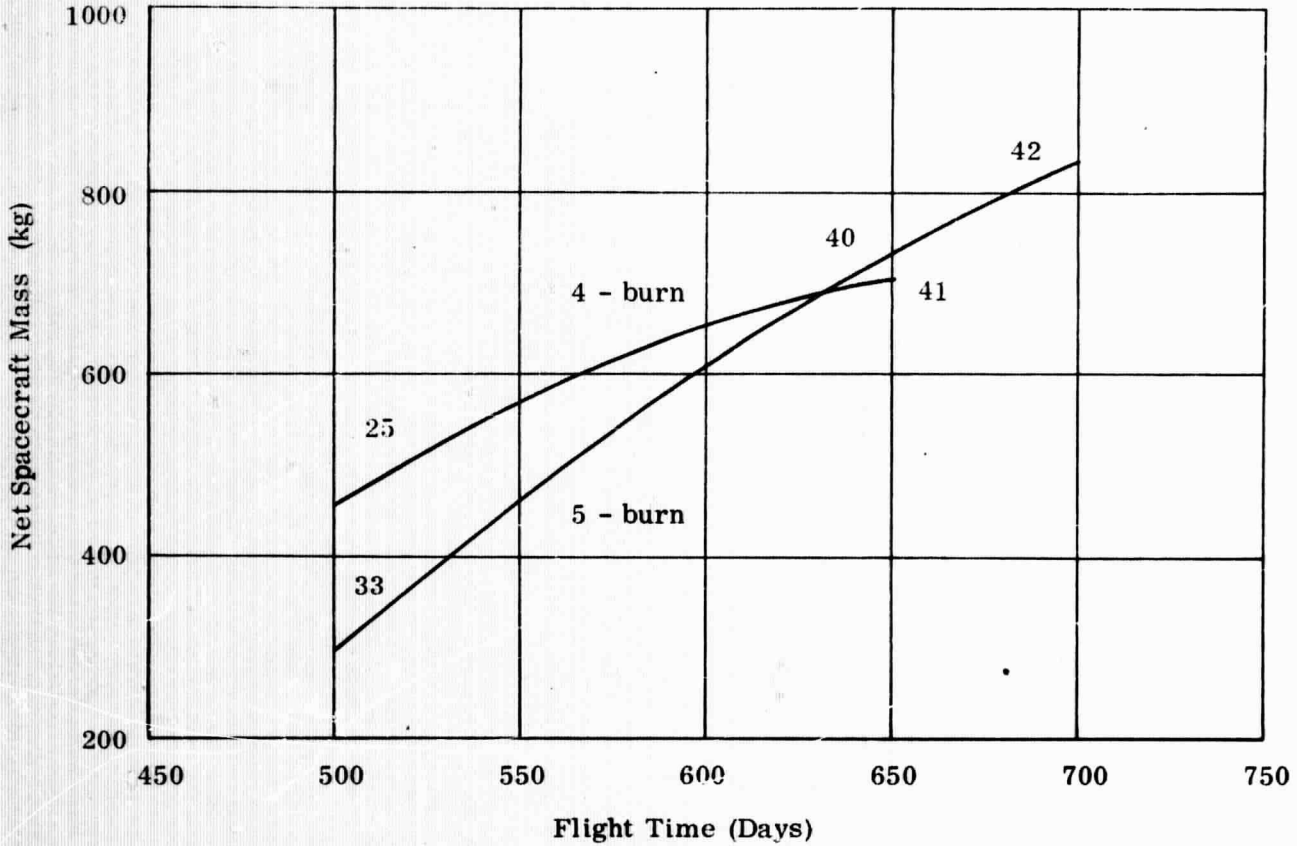


Fig. 16 Performance Requirements for 45° Extra-Ecliptic Missions Using the Titan III D(1205)/Centaur Launch Vehicle.

α = 30 kg/kw

Numbers represent reference power in kw.

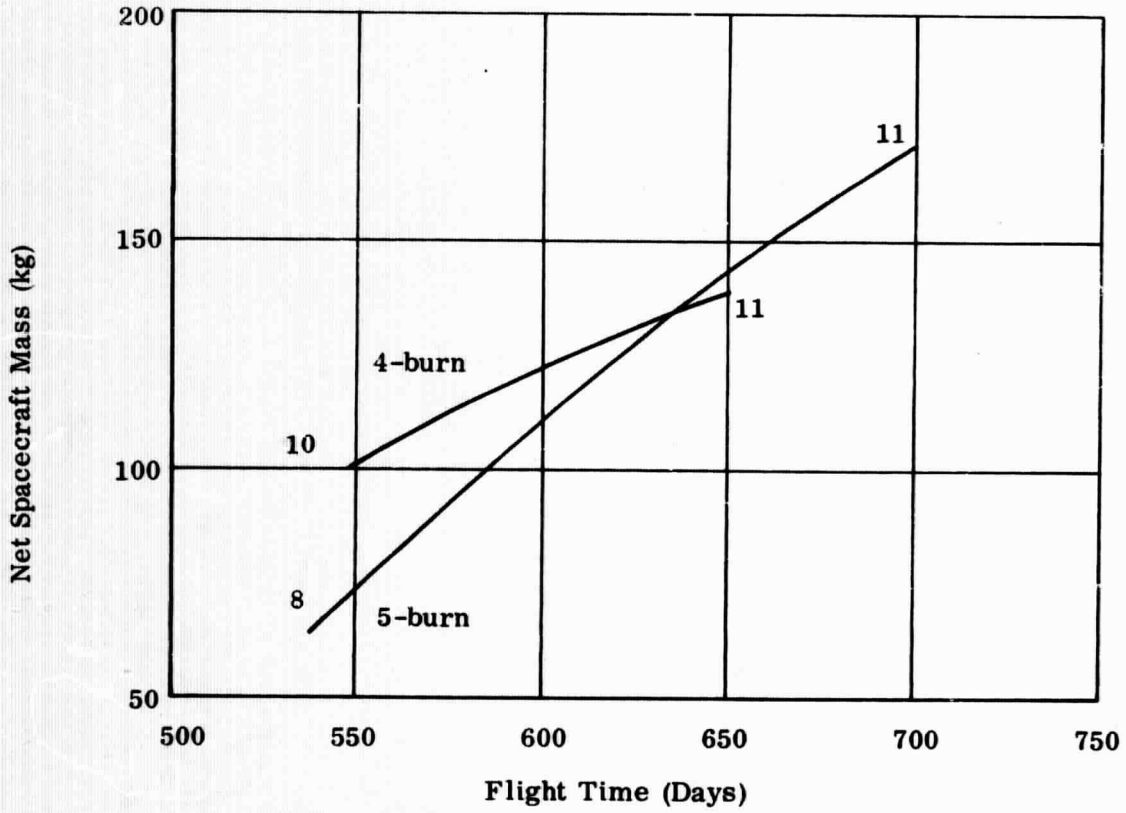


Fig. 17 Performance Requirements for 45° Extra-Ecliptic Missions Using the Titan III B(core)/Centaur Launch Vehicle.

α = 30 kg/kw

Numbers represent reference power in kw.

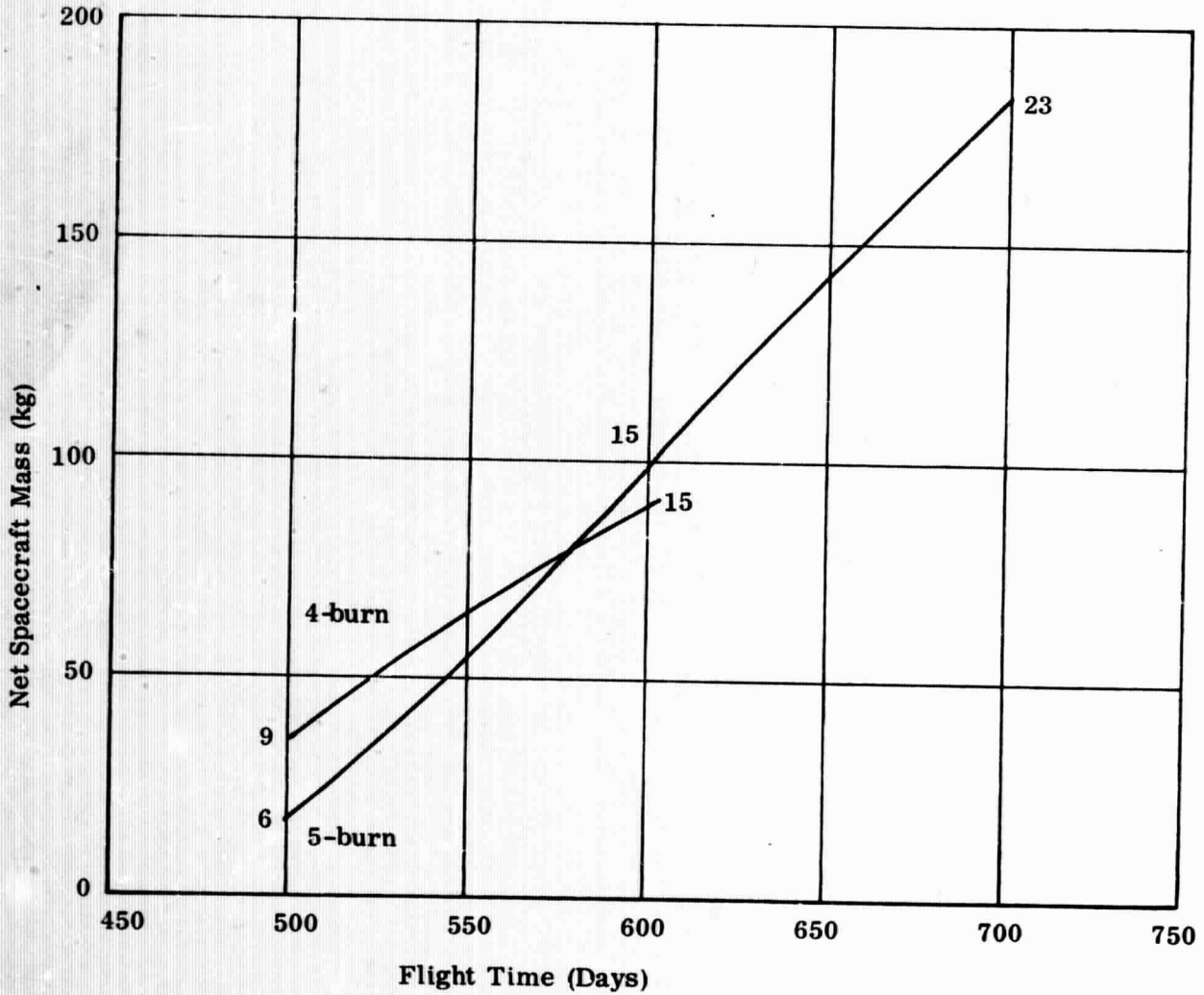


Fig. 18 Performance Requirements for 60° Extra-Ecliptic Missions Using the Titan III D(1205)/Centaur Launch Vehicle.

α = 25 kg/kw

Numbers represent reference power in kw.

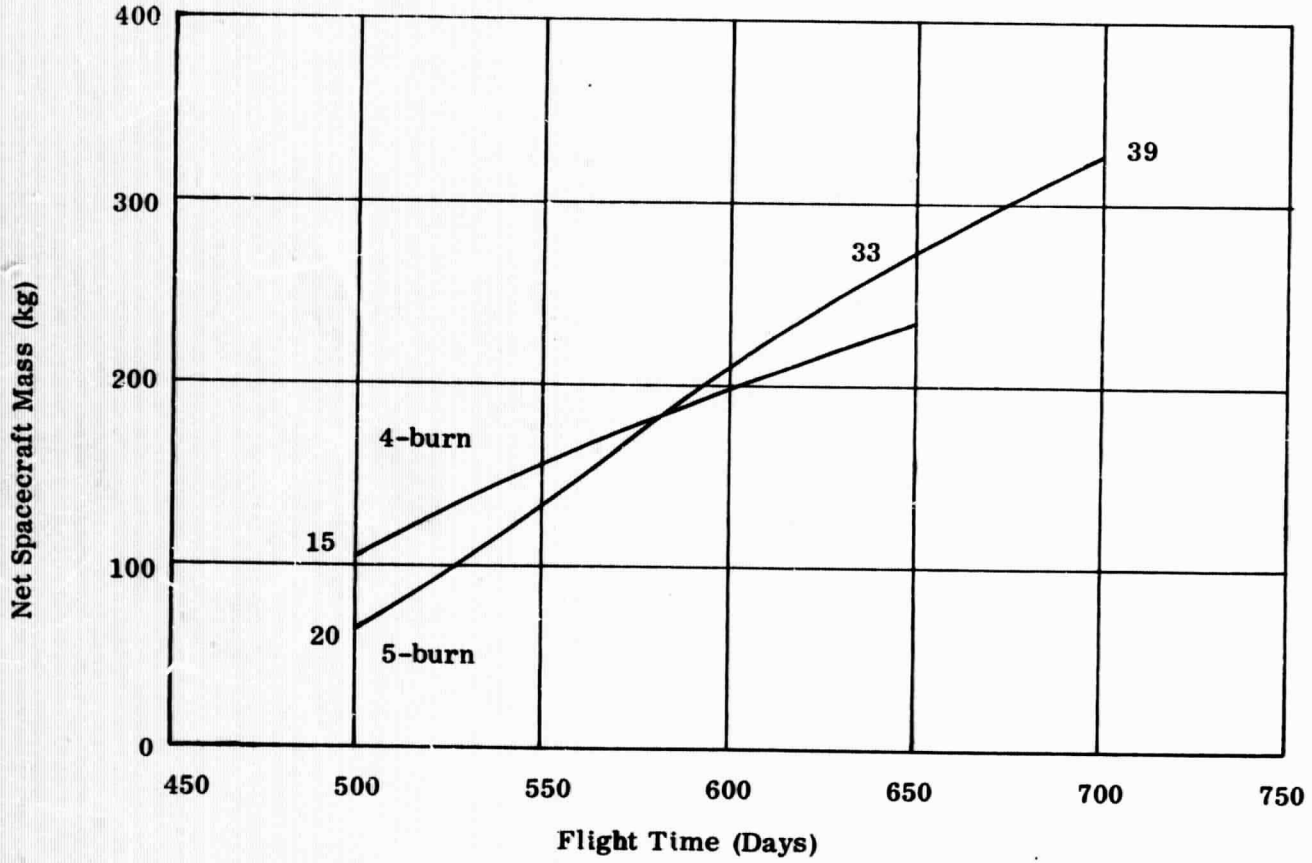


Fig. 19 Performance Requirements for 60° Extra-Ecliptic Missions Using the Titan III D(1205)/Centaur Launch Vehicle.

α = 30 kg/kw

A = Titan III D/Centaur launch vehicle.

B = Titan III B/Centaur launch vehicle.

Solid lines = 1976 launch opportunity.
Dashed lines = 1979 launch opportunity.

Numbers represent reference power in kw.

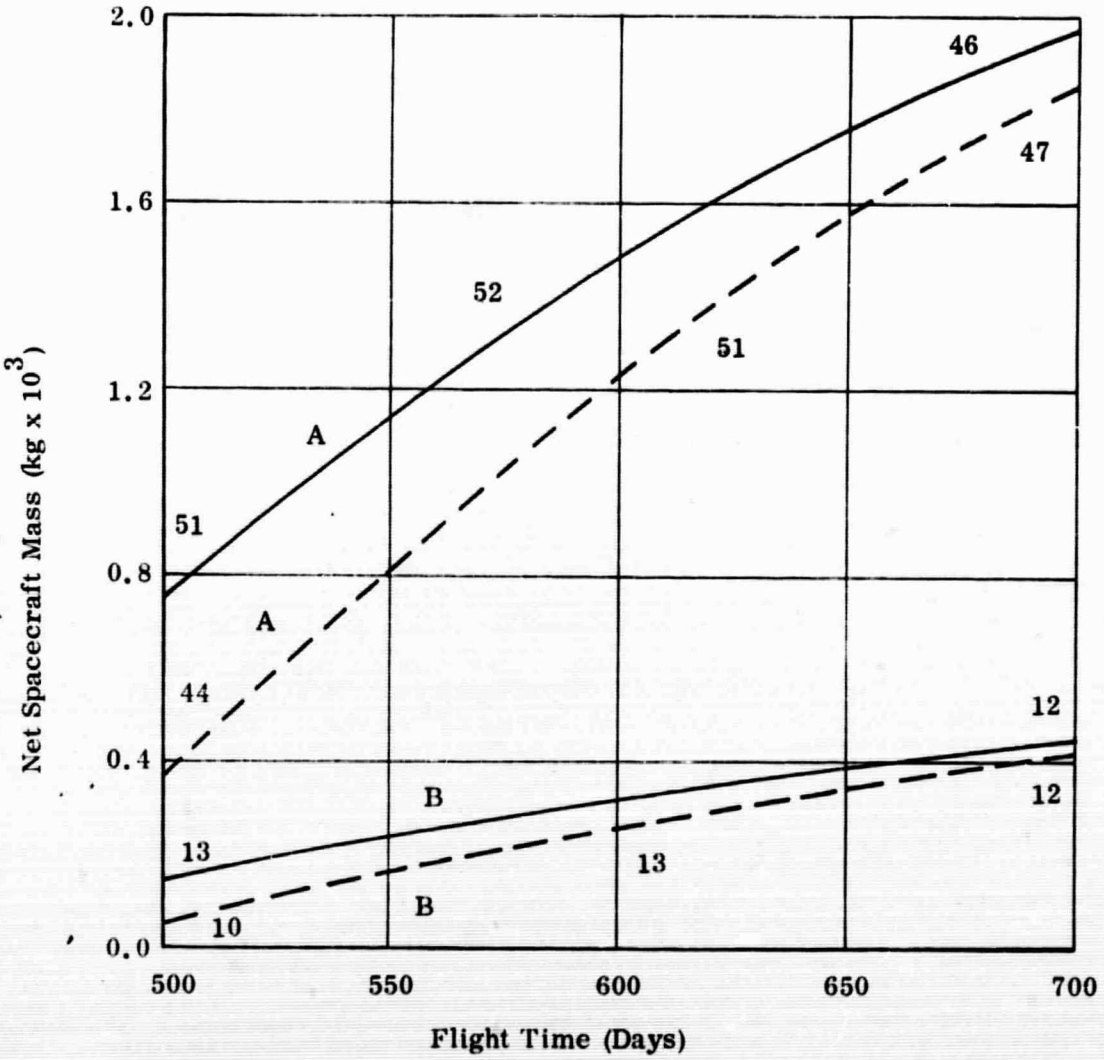


Fig. 20 Performance Requirements for Ceres Rendezvous Missions.

α = 30 kg/kw

A = Titan III D/Centaur launch vehicle.

B = Titan III B/Centaur launch vehicle.

Solid lines = perihelion rendezvous.
Dashed lines - rendezvous 50 days before perihelion.

Numbers represent reference power in kw.

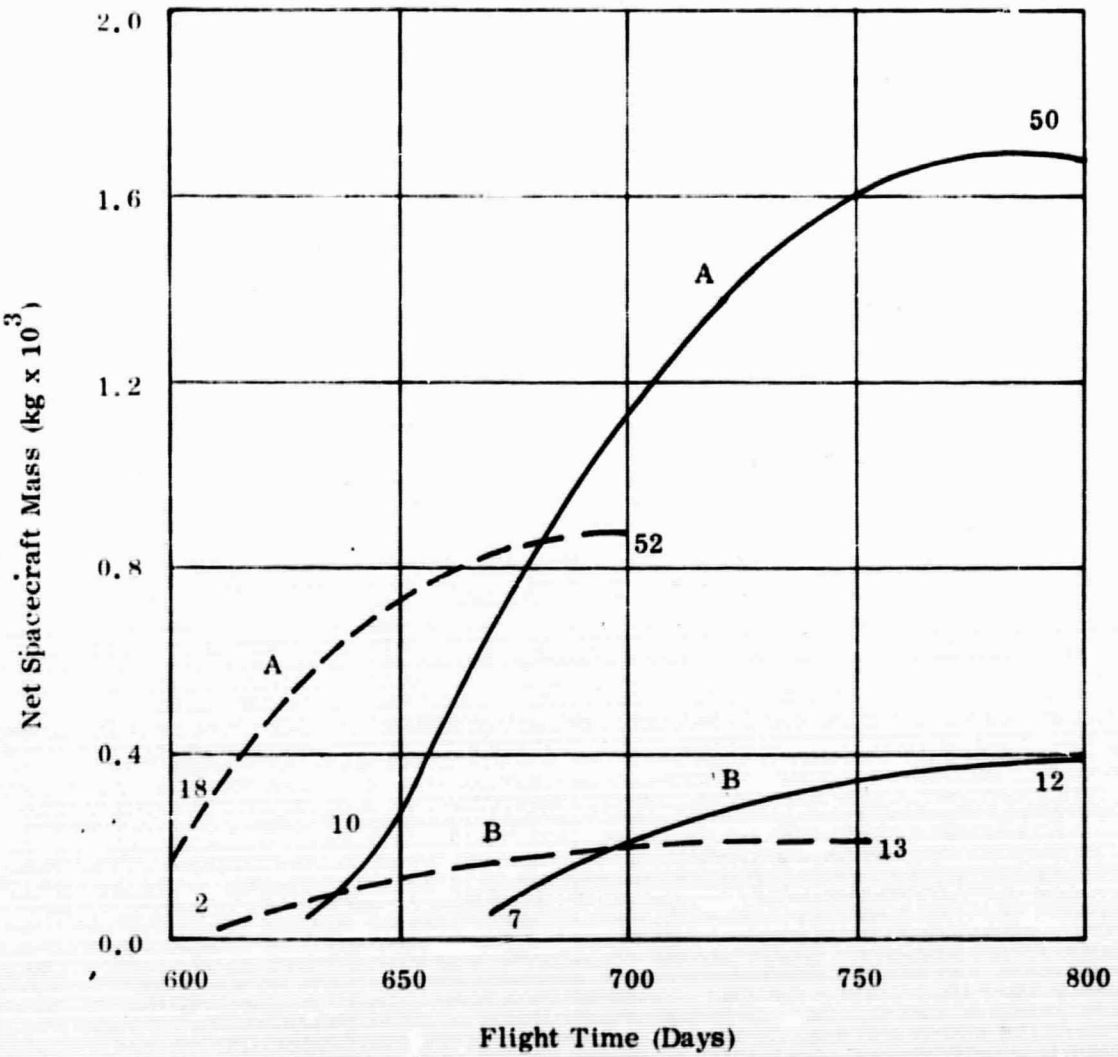


Fig. 21 Performance Requirements for Comet D'Arrest Rendezvous Missions (1982 Apparition).

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