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Summary Report
Study of Low-Acceleration Space
Transportation Systems
Contract NAS8-11309

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Prepared for the
George G. Marshall Space Flight Center
National Aeronautics & Space Administration
Huntsville, Alabama


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# FOREWORD

This document contains the Summary Report for the Study of Low-Acceleration Space Transportation Systems. The study effort was sponsored by the NASA G. C. Marshall Space Flight Center, Huntsville, Alabama, under Contract No. NAS8-11309. The results of the studies reported herein were obtained by the Systems Analysis Section of the United Aircraft Research Laboratories, with support from the Advanced Power Systems Organization of Pratt & Whitney Aircraft.

The complete results of the study are contained in the following volumes:

Volume I - Summary
Volume II - Technical Report

The initial period of performance began in July 1964 and ended in the latter part of June 1965. Additional tasks were added to the original statement of work and the supplemental period of performance covered July 1965 through June 1966.

Although the current document summarizes the results of the entire study insofar as the general study objectives are concerned, further detailed information showing the chronological development of data obtained in the first study phase may be found in United Aircraft Research Laboratories Report D-910262-3, Study of Low-Acceleration Space Transportation Systems (July 1965), Interim Report.

## ACKNOWLEDGEMENTS

Significant contributions to this report were made by the following United Aircraft personnel:

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# Study of Low Acceleration Space Transportation Systems

Summary Report
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## SECTION I
## INTRODUCTION

This report summarizes the results of a two-year study on manned missions to Mars in the 1980 time period using combined high- and low-thrust space transportation systems. The analysis was performed for the National Aeronautics and Space Administration, G. C. Marshall Space Flight Center. The first year's study was executed under Contract NAS8-11309 which was subsequently enlarged to include another year's study of supplemental tasks. Report D-910262-3 (July 1965) details the results of the first year's study effort. The information presented herein pertains to the major results obtained during the second year of effort.

### Objectives and Scope

The basic objective of the entire study was to determine whether a useful manned mission to Mars could be accomplished during the 1980 time period utilizing high- and low-thrust propulsion systems in combination. The principal specific objective was to investigate means for minimizing the vehicle mass placed on Earth parking orbit. During the initial period of performance it became evident that the conduct of the study relied almost entirely upon the low-thrust trajectory data needed to perform the mission studies. A promising approach was developed, and, accordingly, the initial tasks were supplemented by additional work. The prime objective of this additional effort was to further develop and check the simplified trajectory model for low-thrust systems, including calculation of near-optimum trajectories for combined high-low acceleration systems as applied to the manned Mars mission.

The underlying philosophy in the conduct of the study was oriented primarily toward integrating the requirements of the hybrid-thrust flight mode with the operating characteristics of the major vehicle subsystem, the powerplant. The study concentrated on manned trips to Mars occurring in the opposition of 1980. The prime mission of the hybrid-thrust interplanetary vehicle was to deliver a Mars excursion module onto a Martian parking orbit, loiter in the parking orbit for 30 days, rendezvous with the surface exploration team, and return the crew and scientific materials and data to Earth. The major indicator of mission capability was considered to be the total vehicle mass required on Earth parking orbit as a function of the flight and powerplant parameters and in relation to the payload capability of the Saturn V.

The development of criteria and requirements for the low-acceleration propulsion system was limited to two major considerations. First was the establishment of desired operating characteristics for the primary subsystem of interest, i.e., the powerplant. The second consideration was the determination of the optimum mix of high- and low-thrust propulsion which minimizes gross vehicle mass. The two considerations together, interpreted in terms of technology, constituted the comparison data employed to judge the usefulness of hybrid-thrust vehicle systems for manned exploration of Mars.

## Basic Assumptions

For study purposes, certain basic assumptions were made regarding the low-thrust trajectory, the powerplant, and the vehicle system model. It is felt that these assumptions, while they may narrow the scope of the study, do not affect the conclusions made, which are based on relative comparisons, and the evaluation of the worth of hybrid-thrust systems for manned missions.

The hybrid-thrust mode used in the mission studies consisted of a high-thrust nuclear system for departure from Earth parking orbit, an electric system for the heliocentric transfer to Mars, another nuclear system for capture onto a Martian parking orbit, a nuclear propulsion unit for leaving the parking orbit, an electric system for transfer to Earth, and finally an atmospheric entry system for direct capture at Earth. In special instances the high-thrust propulsion systems for Mars capture and departure were eliminated for study purposes, the capture and departure operation then being effected by the electric propulsion system.

All of the low-thrust trajectories employed in the study were of the variable-thrust mode wherein the specific impulse of the electric propulsion system is allowed to vary in a manner such that propellant consumed is a minimum. The use of variable rather than constant thrust leads to slightly lower trajectory requirements and, consequently, less vehicle mass. The differences, however, do not materially affect the results of this study insofar as the general conclusions derived herein are concerned. The general favored characteristics of the powerplant, the basic flight profile, the mix of high- and low-thrust systems, and, therefore, the usefulness of hybrid-thrust manned Mars mission could be just as well delineated under either type of thrusting program. In view of the objectives of this study and the timeliness of the study results, the assumption of variable-thrust operation appeared justified.

Although the specific impulse was allowed to vary, the corresponding efficiency of the thruster was kept constant at an average value. In general, the output of the power system was permitted to vary with time in a probabilistic sense in order to assess the effects of discrete failures in components of the powerplant. The powerplant considered was an alkali-metal, nuclear Rankine cycle system utilizing a lithium-cooled fast reactor. The specific weight of this system was considered to be a parameter varying between 5 and 20 kg/kw throughout the parametric phases of the study.

The vehicle system model employed in the mass calculations consists of a basic spacecraft (containing a solar flare shelter, life support system, etc.), the Mars excursion module, the Earth capture system, nuclear propulsion steps for Earth departure and planetary arrival or departure, and the electric propulsion system for the outbound and inbound heliocentric transfer. The nuclear propulsion system was considered to be of the NERVA/PHOEBUS type ($I_{sp}$ = 800 sec). The computation of vehicle mass utilized weight scaling laws for the life support system, the Earth capture system, and the high-thrust nuclear propulsion systems.

## Study Approach

It was believed that the basic study objectives would be fulfilled by an integrated analysis of the major characteristics of the flight profile, the mixed-thrust trajectories, and power system parameters which are peculiar to the combined high- and low-thrust interplanetary vehicles. The low-thrust trajectory requirements reflect the influence of the mission duration and the corresponding distribution in leg times, the hyperbolic excess speeds at the terminals of the trajectory, and the powerplant's probable decreasing power output. The powerplant operating characteristics directly affecting the vehicle system mass are specific weight and decreasing available power. By computing the total vehicle mass as a function of the different parameters in the trajectory requirements and powerplant characteristics, it is possible to relate the effectiveness of changing the trip time, the powerplant specific weight, and the decreasing available power.

The probable decrease in available power with time is a key parameter in evaluating the effectiveness of technological approaches to enhancing powerplant performance. Accordingly, the majority of the study effort was spent in not only determining the optimum hybrid-thrust trajectory requirements but also in investigating methods of decreasing powerplant specific weight and of maintaining the power output at its original rating. This dichotomous approach merges when the various parameters are integrated by the mission effectiveness studies. In these studies the trajectory characteristics and the power system technology areas are evaluated in terms of the corresponding vehicle mass requirements. This over-all method of analysis allows firm identification to be made of critical powerplant technology areas, their influence on the mission requirements, and the effectiveness of technological development to reducing such requirements.

# SECTION II
# MISSION ANALYSIS AND REQUIREMENTS

The basic purpose of the mission studies was to identify the sensitivity of mission requirements to postulated changes in the trajectory and powerplant parameters and to determine favored flight profiles, propulsion modes, and desired power system operating characteristics. The approach entails integrating the different parameters in a mass computation procedure to determine the vehicle mass required on Earth parking orbit and its corresponding breakdown, and suitably varying these parameters over a range of expected values. The results are displayed in terms of mass on Earth orbit (MEO), the parameter indicating the mission requirements.

The primary mission of the manned Mars flight was assumed to be a surface exploration in 1980 for a period of 30 days, with a capability of returning 454 kg of scientific samples and information. A crew of 8 astronauts was fixed throughout the study with the number of men descending to the surface taken as not more than four.

The outbound leg high-thrust departure from Earth, low-thrust heliocentric transfer, and capture at Mars were optimized between the high- and low-thrust systems to provide maximum payload-to-gross weight ratio; similarly for the return leg. In addition, the decrease in powerplant output as a function of time was included in the trajectory analysis, thereby providing a means of relating postulated technological improvements in power system performance to the vehicle mass requirements. The level of output delivered by the power system depends on the assumed component failure rates and the probability level desired (see Section IV). In order to provide a link with presently known technology, the initial failure rates were assumed to be those experienced by commercial aircraft gas turbines. Improvements in component technology were postulated which correspond to failure rates about a factor of ten lower than those of aircraft gas turbines.

## Scaling Law Assumptions

The solid-core nuclear high-thrust propulsion step is sized using an empirical equation for the inert weight fraction, $\beta$, as a function of the impulse propellant, $m_p$ (kg).

$$\beta = 0.296 \left(\frac{m_p}{42200}\right)^{-0.264}$$

The propellant required by the mission is given by

$$m_p = \frac{(1-\beta)}{1-\mu\beta} (\mu-1)\ m_L$$

where $\mu$ is the stage mass ratio and $m_L$ is the mass accelerated by the step. The two equations are solved iteratively in the mass computation program.

In the preliminary analyses of the Mars mission the Mars excursion module (MEM) mass was kept constant at 45 metric tons. For the analysis of spacecraft concepts in which the parking orbit operations strongly affect the MEM, the mass of the excursion module was computed wherein the different velocity requirements were accommodated.

The mass of the life support system is a function of the trip duration and the crew size and is given by

$$m_{Ls} = [1.48\ (n-4) + 7.1]\ (T-200) + 500\ (n-4) + 2370$$

where $m_{Ls}$ is the life support and environmental control system mass (kg), n is the number of crewmen, and T is the trip duration (days).

The Earth entry system was considered to be an advanced ablative type capable of atmospheric entry speeds up to 20 km/sec (65,000 ft/sec). The growth of such a system with both entry speed and crew size is given by

$$m_e = 2360 + 167\ n + [239 + 91\ (V_e - 11)]\ n^{2/3}$$

where $m_e$ is the entry system mass (kg), n is the crew size, and $V_e$ is the atmospheric entry velocity (km/sec).

The basic spacecraft exclusive of life support system, MEM, and Earth entry system was estimated to weigh about 45 metric tons. The total mass on Earth parking orbit is found by appropriately adding to the 45 tons the masses of the MEM, the high- and low-thrust propulsive steps, the life support system, and the Earth entry system.

## Mission Requirements

The MEO required for different mission durations and powerplant specific weights and optimum nuclear-plus-electric operation is given in Fig. II-1. These values of mass represent the most optimistic cases considered, since the output of the powerplant was assumed to be constant at 100% of the initial power rating. The dotted curve indicates the mission requirements under a rendezvous Earth-return mode. In this case a rendezvous vehicle must be launched from Earth to retrieve the crew and scientific materials which are at parabolic conditions with respect to Earth. The mass required for the rendezvous mode is significantly higher than the system employing an ablative entry system. The rendezvous mode is even more unattractive because an additional launch of at least a Saturn V is required whose payload delivered on Earth parking orbit should be charged against the mass requirements shown by the dotted curve of Fig. II-1. In subsequent studies the ablative entry mode at Earth was utilized for the foregoing reasons.

The powerplant specific weight has slight effect on the mass required provided the mission duration is greater than 500 days and ablative Earth capture is employed. The insensitivity of mass to specific weight is important because of the currently contemplated powerplant specific weight values.

A comparison of different propulsion system capabilities is shown in Fig. II-2. The all high-thrust vehicle mass requirements for three different levels of nuclear rocket technology are compared against the hybrid-thrust requirements. On the basis of mass, it is seen that the hybrid-thrust propulsion system (solid-core nuclear plus electric) is essentially equivalent to both advanced nuclear propulsion systems. For the Mars trip at least, and under the attendant assumptions, it is conceivable that, in terms of the technological advances required, the hybrid-thrust system could be developed earlier and would yield about the same advantages.

A closer comparison of the mass required of the hybrid (solid-core nuclear and electric) mode against all high-thrust operation is shown in the accompanying table for trip times of 370 and 485 days and a staytime of 30 days.

| Propulsion System | Mass, metric tons 370 days | Mass, metric tons 485 days |
|---|---|---|
| Solid Core Nuclear ($I_{sp}$ = 800 sec) | 4030 | 1633 |
| Hybrid (Solid Core Nuclear + Electric) | | |
| 15 kg/kw | 1650 | 680 |
| 5 kg/kw | -- | 530 |
| Advanced Nuclear ($I_{sp}$ = 2000 sec) | 728 | 535 |

The hybrid-thrust system requires less mass compared to the nonmixed solid-core nuclear system for both trip times. In fact the hybrid mode is almost competitive with a later generation of nuclear rockets, i.e., propulsion systems having a specific impulse on the order of 2000 sec. In the case of the 485-day mission, the hybrid system using a 5 kg/kw specific-weight powerplant requires about the same mass as the 2000-sec nuclear system.

Introduction of the power profiles for different maintenance and reliability levels allows a comparison of the trade-off between these two levels. The results are presented in Figs. II-3 and II-4, respectively, for the 530- and 630-day missions. A "desirable" system consisting of infinite repair capability (except for reactors) and a probability level of 0.999 requires almost the same mass as a technologically "early" system made up of a non-maintained powerplant operating at a probability of 0.99, assuming specific weight is the same. The effect of powerplant specific weight on MEO is seen to be less for the maintained powerplant compared to the nonmaintained plant. This effect indicates that maintaining the powerplant and operating at lower probability levels aids in mitigating the influence of specific weight. This aspect becomes important when consideration is given to the effective distribution of development effort towards achieving desired power system operating characteristics.

The desirability of an onboard maintenance activity was investigated further by selecting different maintenance levels and including the accompanying effect on powerplant specific weight in the mass computations. The results are shown in the following table for the 530-day round trip. The maintenance level corresponds to the number of spares.

| Number of Spares | Powerplant Specific Weight, kg/kw | Mass on Earth Orbit metric tons |
|---|---|---|
| 29 | 14.28 | 638 |
| 42 | 14.55 | 606 |
| 84 | 15.45 | 579 |
| 126 | 16.37 | 571 |
| 168 | 17.28 | 572 |

As would be expected, increasing the maintenance activity increases the power availability at the expense of specific weight. Some point in the maintenance level should be reached where the advantages of the higher available power are more than offset by the increase in specific weight. For the specific trip shown in the table, the minimum mass is reached at about a maintenance level of 126 spares where the corresponding specific weight is just above 16 kg/kw. The obvious maintenance level to employ would be the lowest level giving practically 100% power throughout the mission.

The effect of optimum hybrid-thrust operation on the required powerplant rating (initial power) is illustrated in Fig. II-5 for the 530-day mission. Power required to perform the mission decreases with trip time but does not increase as the specific weight becomes higher because as specific weight does increase the hybrid-thrust optimization process utilizes more of the high-thrust systems (especially Earth-departure) rather than the low-thrust system. Hence the required powerplant mass does not increase as fast as the specific weight, thereby resulting in lower initial powerplant ratings.

The operating points for a power system applied to the missions studied herein can be found by superimposing the candidate power system's specific weight versus power curve over plots of the mission power requirements. The power system operating characteristics strongly influence the vehicle mass which in turn identifies favored technological levels and operating modes. The implications of the mission studies to the evaluation of the power system are discussed in Section IV.

## Spacecraft Concepts

Judging from the results of the mission studies discussed previously, there are trips which can be performed with a single power module under certain assumptions of maintenance and reliability levels. For mission durations greater than 500 days, the initial powerplant ratings are less than 4 Mw for hybrid-thrust vehicle operation (see Fig. II-5). The basic power module analyzed in Section IV is rated at 4 Mw (single reactor); hence, it appears that a vehicle design concept may be developed employing one power module, rather than two as previously required. The conceptual design, formulated for a 530-day mission, illustrates the placement of the various transportation system components and indicates general problem areas associated with using a nuclear-electric powerplant in conjunction with high-thrust systems.

An important study parameter affecting the design of the spacecraft is the type of orbit established about Mars. One selected for study is a 926-km circular orbit with a period of about 2 hours. The other orbit is highly elliptic with an eccentricity of about 0.98, a 30-day period, and a periapsis distance of 926 km above the Martian surface. If a circular orbit is desired, the spacecraft must retro-brake to circular velocity from some fraction of the hyperbolic excess speed remaining from the low-thrust braking. The propulsion required for the spacecraft to establish orbit is relatively large while the MEM propulsion is minimized. Conversely, if a highly elliptical orbit is required, the propulsive braking required for the spacecraft is very small, since the orbit is nearly parabolic and the initial conditions are just above parabolic. The corresponding MEM propulsion is large, since the de-orbit from the ellipse to the surface requires a large incremental velocity.

These operational modes have a profound effect on the mass of the MEM vehicle as shown in the accompanying table. Each vehicle consists of a basic 3.17-metric ton command module, which includes arresting gear, landing structure, and $O_2/H_2$ propellant ($I_{sp}$ = 430 sec).

| | Mass, metric tons | |
|---|---|---|
| | Circular Orbit | Elliptic Orbit |
| Basic 4-Man Module | 3.17 | 3.17 |
| Rendezvous Step | 0.16 | 1.95 |
| Mars Ascent Step | 7.59 | 11.65 |
| Mars Landing Step | 24.74 | 38.10 |
| De-Orbit Step | 1.04 | 33.63 |
| TOTAL MEM WEIGHT | 36.70 | 88.50 |

Figure II-6 illustrates the general layout of the spacecraft conceptual design in its interplanetary configuration. The spacecraft is essentially of modular construction. The electric powerplant, thrusters, solar shelter, Earth entry module (ERM), and the manned modules comprise a basic section of the spacecraft which is independent of the type of high-thrust propulsion. As shown in Fig. II-6, the Mars capture and escape stages can be varied according to the requirements for orbital capture and are jettisonable after use.

Only the electric powerplant, solar shelter, and manned modules rotate. The compartment housing the ERM and the MEM, and the high-thrust propulsion systems are nonrotating. The electric powerplant occupies one extremity of the spacecraft. Radiation shielding, located at the reactor, provides a 30-deg half-angle shadow for the spacecraft and lies in the plane of the manned modules. Conceivably, only the manned modules need rotation. However, shadow shielding would be required for the entire 360-deg sweep of the rotating modules which results in heavy shield weights. Therefore, for this spacecraft concept, only small "ears" are required for the shadow shield, since the shield rotates with the manned modules.

The thrusters are located in four panels which are erected radially from their stowed position in the outer vehicle skin surrounding the solar shelter. The size of the thruster array will vary considerably, depending on the type of thrusters used. It is assumed that 4 megawatts (electric) are supplied to the thrusters. If cesium contact thruster modules are used, the total required area of the thruster array is about 60 square meters. The use of mercury electron bombardment modules would increase the total thruster panel area requirement to about 650 square meters. These areas assume that about 30% of the thrusters are spares.

The solar shelter is used primarily for protection against solar radiation. During these solar flares, the rotation of the spacecraft is stopped. The crew enters the shelter through the central hub. The shelter contains basic control and communication instrumentation, and life support requirements for continuous periods up to 3 or 4 days duration. This module is also occupied during high-thrust nuclear operation as a protective measure against the accompanying radiation.

The manned modules are located on extendable arms which telescope inward during major high-thrust maneuvers to minimize structural bending. The two manned modules are extended to a radius of 25 meters during the phases of the mission in which artificial gravity is required. The vehicle rotates at a rate of 0.4 rad/sec, providing a force of 0.4 g's at the floor of each module. One module is the primary command and control center for the spacecraft and is normally occupied by four men. The scientific module serves as the surface exploration team's living and recreation area. Also included in this module is scientific equipment as required.

The ERM is stowed in an interior location and is protected from the flight environment. It is capable of returning eight men to Earth with atmospheric braking, at speeds up to 20 km/sec. This vehicle also delivers scientific data, equipment, and materials to Earth.

The preliminary spacecraft mass summary (interplanetary configuration) is given in the following table for two types of high-thrust propulsion systems and two types of orbits about Mars.

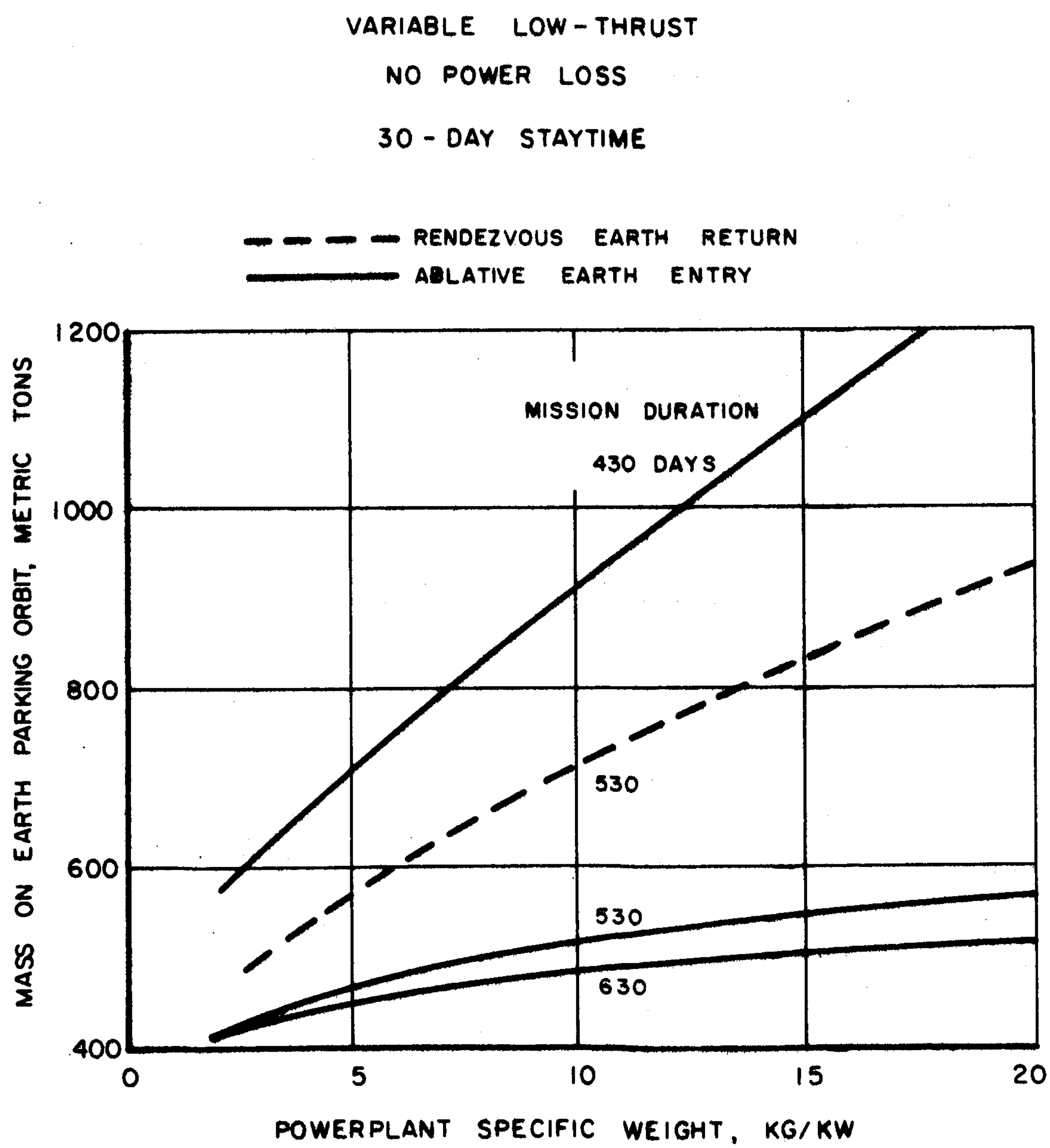
| | Circular Orbit | | Elliptic Orbit | |
|---|---|---|---|---|
| | Nuclear | Chemical | Nuclear | Chemical |
| Basic Spacecraft, metric tons | 58.09 | 53.56 | 58.09 | 53.56 |
| Electric Propulsion System | 119.89 | 119.89 | 119.89 | 119.89 |
| Mars Excursion Module | 36.70 | 36.70 | 88.50 | 88.50 |
| Mars Escape Step | 72.40 | 85.40 | 17.40 | 14.70 |
| Mars Capture Step | 103.00 | 139.53 | 25.06 | 18.16 |
| TOTAL, metric tons | 390.08 | 435.08 | 308.94 | 294.81 |

A contingency of 4.5 metric tons is included for radiation shielding with high-thrust nuclear systems. The minimum Earth-escape payloads occur when the elliptic Mars orbit is utilized, with the chemical Mars stages resulting in slightly less over-all mass required. The circular orbit mission increases the mass requirements of the Earth escape payload by about 27% for solid-core nuclear propulsion and about 47% for chemical. Further analysis is required to account for the orbital operational mode in the over-all mission optimization.

# EFFECT OF SPECIFIC WEIGHT ON VEHICLE MASS

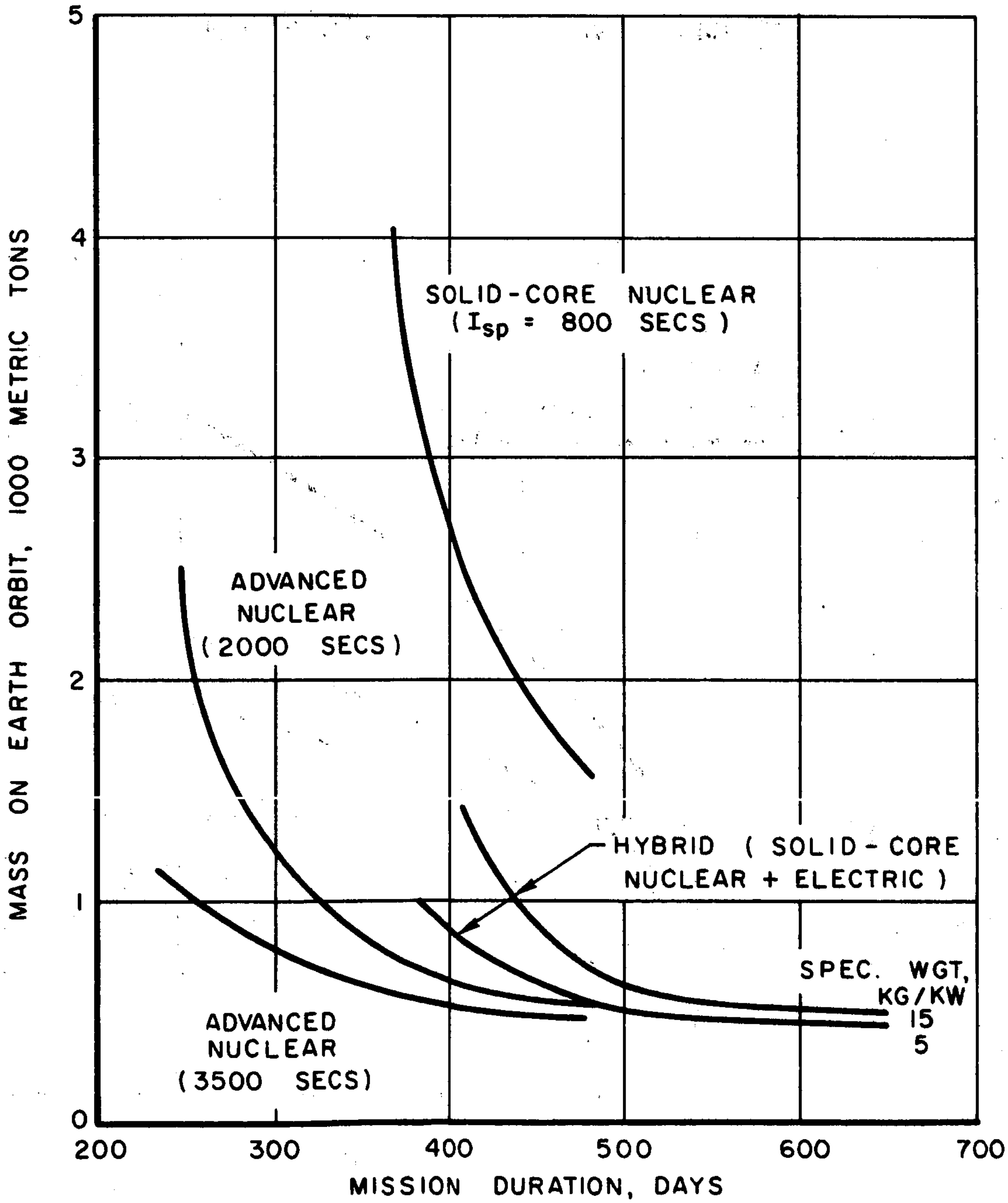
## MARS ROUNDTRIP MISSION

## OPTIMUM HYBRID-THRUST OPERATION



# COMPARISON OF MISSION REQUIREMENTS FOR VARIOUS PROPULSION SYSTEMS

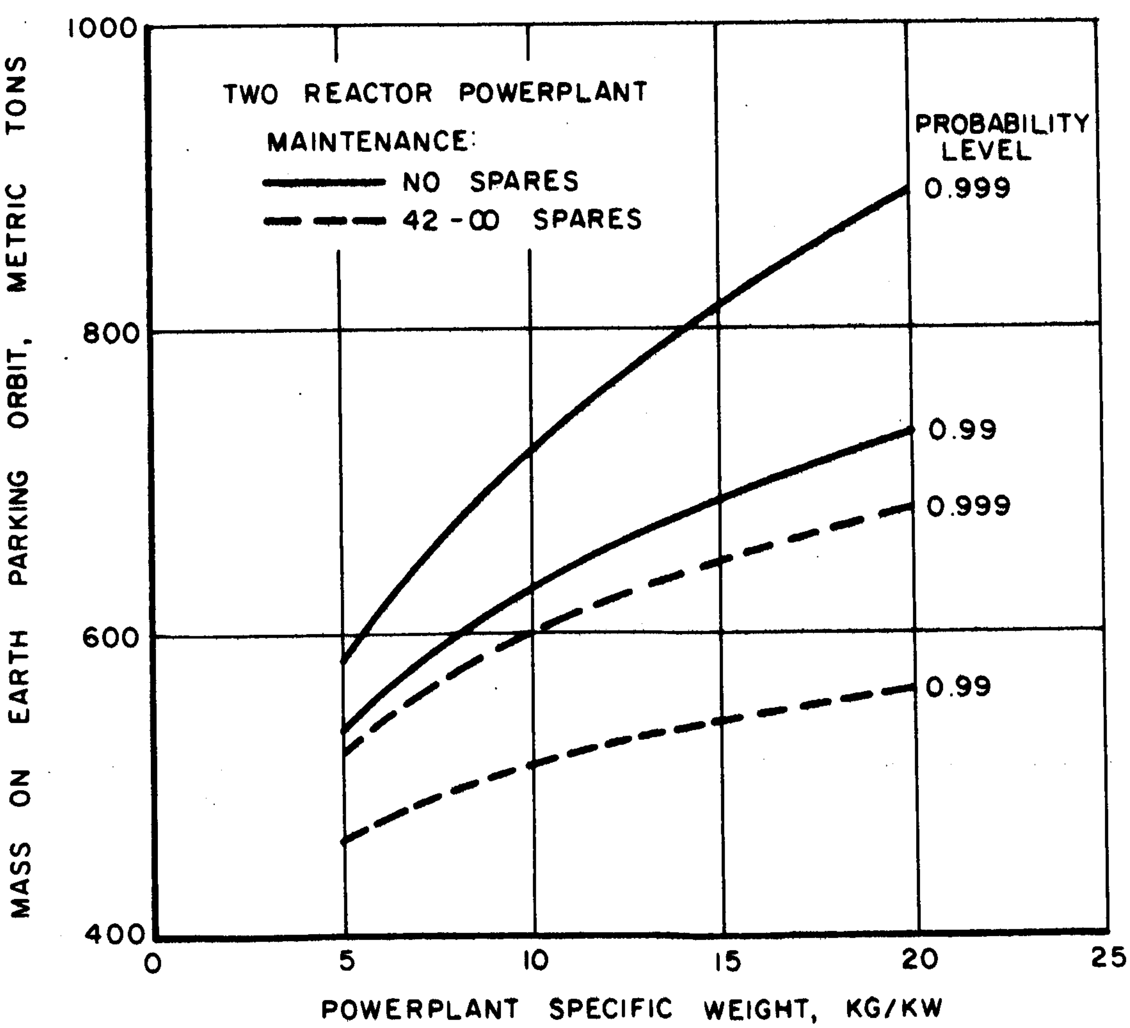
## 1980 MARS MISSION 30-DAY STAYTIME



# EFFECT OF POWERPLANT ON VEHICLE MASS

## 530 DAY MISSION

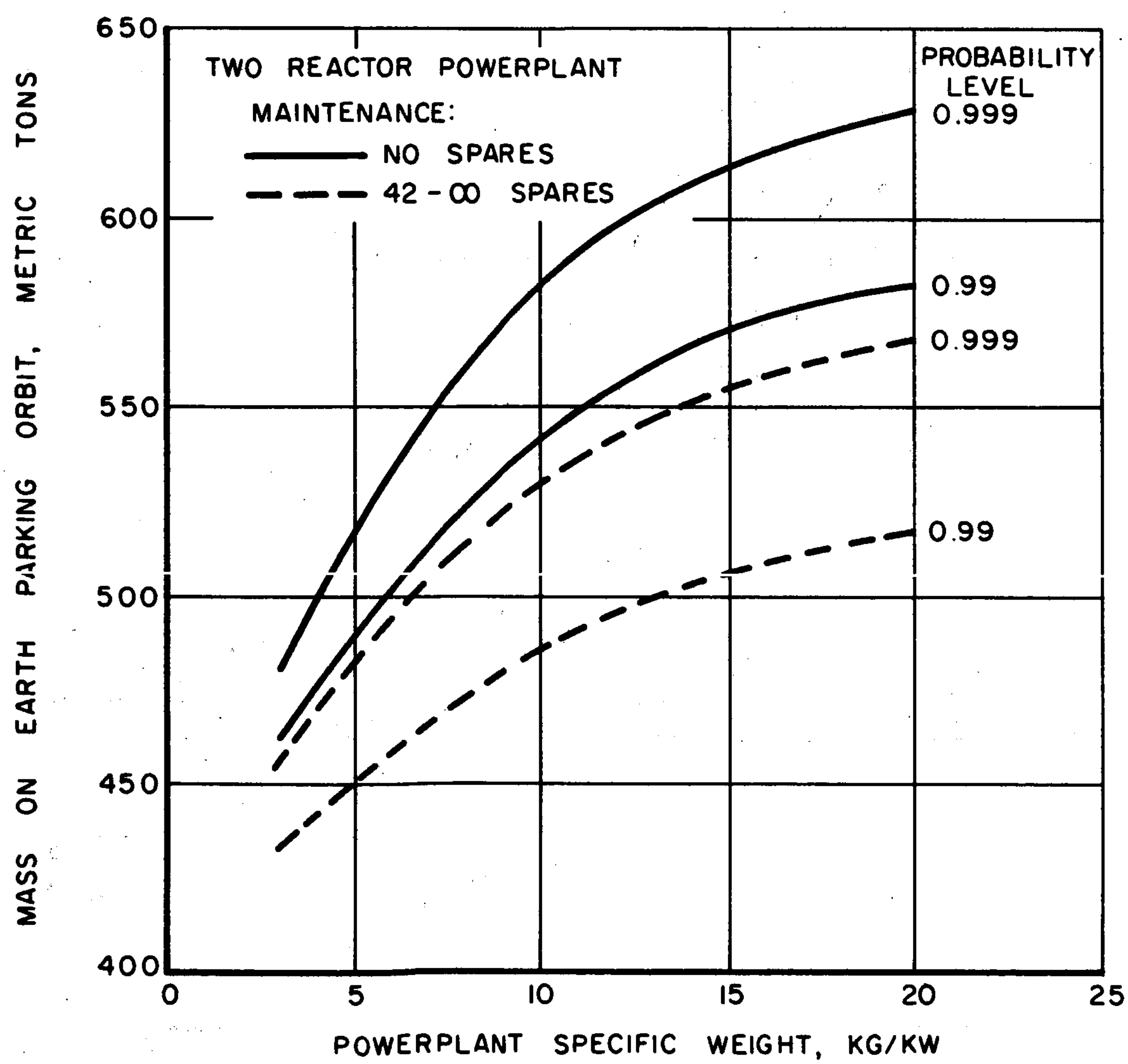
## OPTIMUM NUC + ELECT OPERATION



# EFFECT OF POWERPLANT ON VEHICLE MASS

## 630 DAY MISSION

## OPTIMUM NUC + ELECT OPERATION



# POWERPLANT REQUIREMENTS FOR 530-DAY MARS MISSION

## OPTIMUM NUCLEAR + ELECTRIC OPERATION

TWO REACTOR POWERPLANT

MAINTENANCE :

——— NO SPARES

- - - 42-∞ SPARES

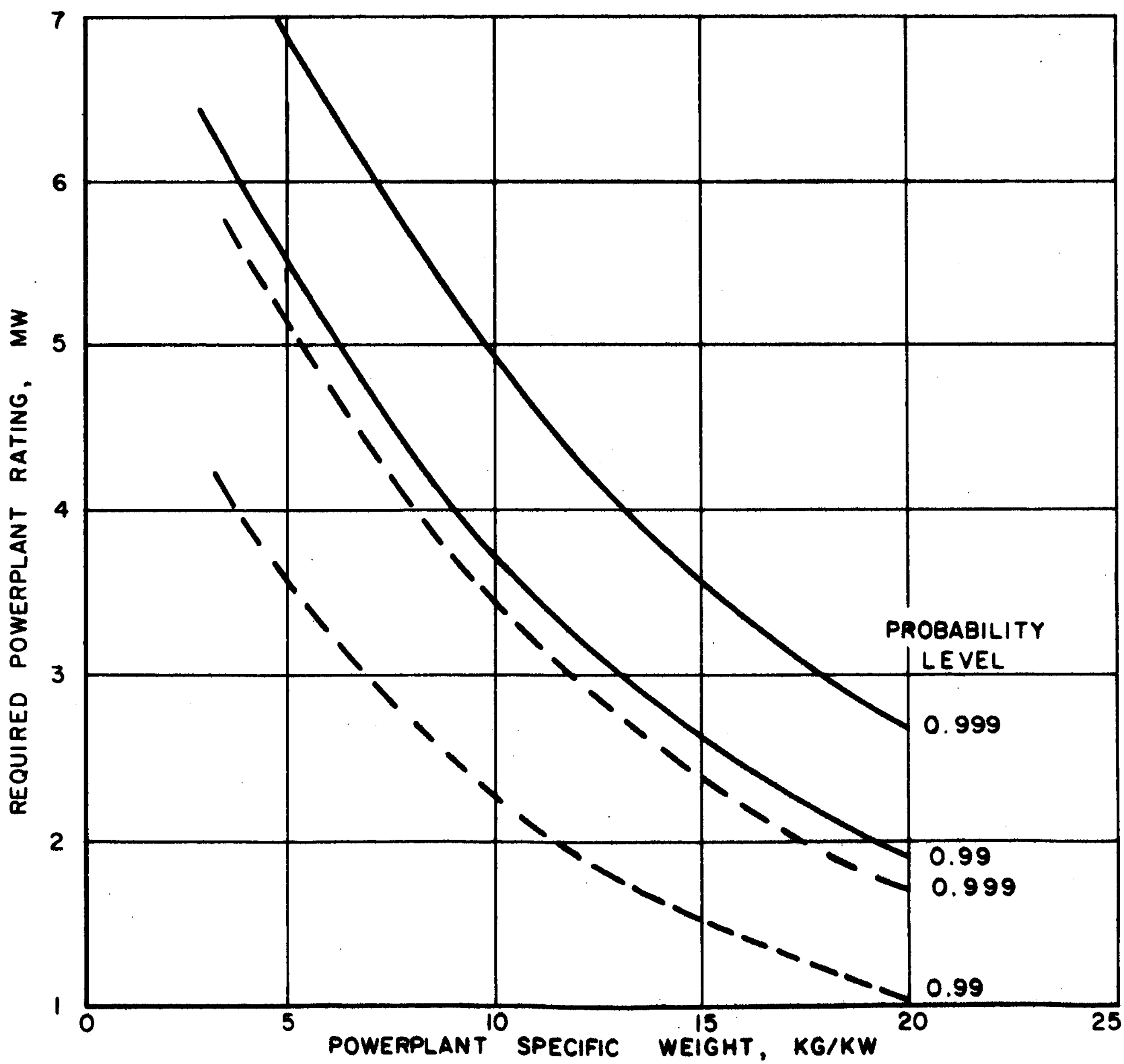


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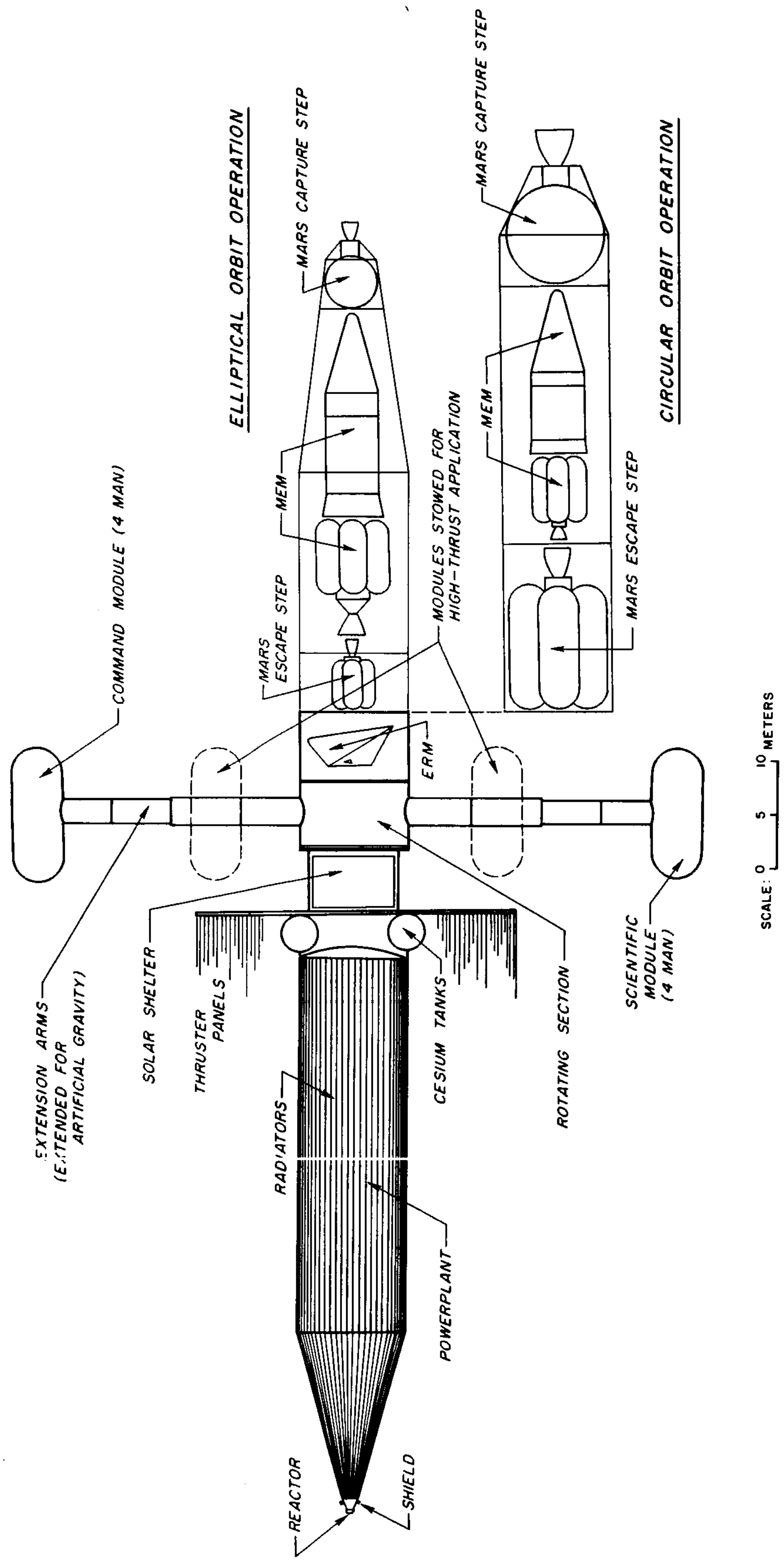
FIG. II - 6

MARS HYBRID–THRUST SPACECRAFT

CHEMICAL + ELECTRIC PROPULSION



# SECTION III
# TRAJECTORY ANALYSES

## Introduction

The search for optimal low-thrust interplanetary trajectories utilizing the methods of the calculus of variations yields a two-point boundary-value problem characterized by a set of coupled second-order, nonlinear ordinary differential equations with time as the independent variable. In the first study phase, a new approach to the solution of the two-point boundary-value problem had been tried with some success (Refs. III-1 and III-2). This method is based upon an extension of Newton's method for finding roots of nonlinear equations applied to operator equations in Banach spaces (Ref. III-3). This generalized Newton-Raphson method (developed in Refs. III-4, III-5, and III-6) offers not only a wide domain of convergence, but also quadratic convergence to the solution. Like the gradient methods, it requires that an initial approximation to the solution be supplied, but it does not require the guessing or adjustment of any numerical constants. Moreover, because of the wide domain of convergence, the initial approximation to the solution need not be a sophisticated one.

An improvement of the generalized Newton-Raphson method has been made for the present mission study through the introduction of an implicit finite-difference approach to the linear two-point boundary value problem which has been discussed in Ref. III-7 and others. The unique feature of the improvement is the combination of the generalized Newton-Raphson method with the finite-difference approach for the solution of a system of nonlinear, second-order differential equations. A detailed exposition of the finite-difference Newton-Raphson algorithm is presented in Refs. III-8 and III-9.

The algorithm was initially written to solve the problem of optimal low-thrust interplanetary trajectories in two dimensions for the particular case of constant kinetic power in the exhaust jet and completely unconstrained specific impulse. Two efforts were undertaken that modified these underlying assumptions and correspondingly altered the algorithm. First, the equations of motion were rederived (in three dimensions) to represent the case of constant specific impulse flight with optimal coast periods. This modification of the Newton-Raphson algorithm is adjoined with a second, external routine necessary to extremize the powerplant characteristics; these two routines together iteratively converge to the optimal thrust and steering schedule and the optimal propulsion parameters for the mission. Secondly, the governing differential equations were rederived for the case of completely unconstrained specific impulse and a jet exhaust power possessing a generalized dependence upon time and the position of the vehicle; the algorithm has utility in the comparison of power modes for various types of missions.

## Statement of the Problem

For power-limited propulsion systems, the mass of propellant expended is given by Eq. (III-1),

$$\frac{1}{M_T} = \frac{1}{M_0} + J = \frac{1}{M_0} + \int_0^T \frac{a^2}{2P(X,t)}\,dt \qquad \text{(III-1)}$$

where $M_T$ and $M_0$ are the final and initial masses of the vehicle, respectively, $\vec{a}(t)$ is the thrust acceleration of the vehicle over the powered flight time, T, and $P(X,t)$ is the kinetic energy in the exhaust jet relative to the vehicle; X represents the vector of state and control variables. Minimum propellant expenditure, that is, J as small as possible, is required for optimal variable thrust trajectories. The Newton-Raphson algorithm computes interplanetary transfer trajectories of a power-limited low-thrust space vehicle between two given planets with specified departure and arrival dates that are optimum in the sense that the value of

$$J = \int_0^T \frac{a^2}{2P(X,t)}\,dt$$

is a minimum. In the machine program, the algorithm numerically solves the Euler-Lagrange necessary conditions for a minimum value of J coupled with the differential equations of the motion which constrain the optimal trajectory.

## Variable-Thrust, Constant-Exhaust-Power Trajectories

In this analysis the heliocentric orbits of the departure and arrival planets are considered to be coplanar but have the correct eccentricity. This work has been reported in greater detail before (Ref. III-10) and is summarized here as a preface to the modifications of the basic algorithm that follow.

### Trajectory Profiles

The algorithm was employed to generate a set of optimum Earth-Mars round-trip trajectory data for the aphelion opposition year, 1980. Similar to plots presented in the Planetary Flight Handbook (NASA SP-35) for high thrust, Fig. III-1 shows contours of constant values of J plotted against departure and arrival Julian dates at Earth and Mars.

### Sample Round-Trip Trajectory

Figure III-2 illustrates an optimum Mars round-trip trajectory during 1980 with 630 days total trip time. The vehicle arrives at Mars at 244 4125 after a 320-day outbound flight. After a 30-day wait the inbound trajectory departs for Earth, arriving at 4435 after a 280-day flight. Also shown in the figure are vectors representing the thrust acceleration magnitude and direction.

Mixed High- and Low-Thrust Acceleration

As a result of using a high-thrust device in a low-altitude planetary orbit or of atmospheric entry at greater than parabolic speed, there arises an initial (or final) nonzero velocity of the vehicle with respect to the planet, usually called the hyperbolic excess velocity. Thus, for a given amount of high-thrust $\Delta V$, the initial (or final) heliocentric velocity of the vehicle is given by the vector addition of the planetary velocity vector and the hyperbolic excess velocity vector. The direction of the latter vector must be chosen so as to minimize the resulting value of J. The transversality condition that accomplishes this states that the direction of the hyperbolic excess velocity vector must be parallel to the resulting optimum low-thrust acceleration vector on the boundary. The machine program has been extended by substitution of the transversality conditions corresponding to impulsive changes in velocity at the boundaries for the fixed boundary conditions on velocity.

In Fig. III-3 the value of J for the low-thrust contribution to a transfer is plotted against the fraction of high thrust employed. As would be expected, the curve is monotonically decreasing from a value of 4.4 $m^2/sec^3$, for the all low-thrust case, to zero for the all high-thrust case. At each point the directions of the hyperbolic excess velocity vectors have been optimized through the transversality condition.

## Variable-Thrust, Variable-Exhaust-Power Trajectories

In this analysis, the power in the exhaust jet is chosen to be

$$P(\bar{X},t) = \frac{P_o e^{-\gamma t}}{R^n}$$

where $R^2 = x^2 + y^2 + z^2$ and $\gamma$ and n are time-independent parameters. The proportionality of exhaust power to $1/R^n$ is chosen to allow the solution latitude in taking into account the degradation of solar cell efficiency due to large thermal gradients encountered during close passage of the sun. The exponential term may represent the time decay of a radioisotope power source, or, perhaps more importantly, it may represent the reliability of the power source over the trip duration based upon a postulated power-plant component failure rate. The proper choice of n and $\gamma$ can represent several different power modes. The work is carried out in three dimensions with orbits of the departure and arrival planets supplied by an ephemeris-generating subroutine.

Figure III-4 illustrates a typical comparison of the trajectories and the attendant values of J obtained through the use of constant power, radioisotope power, and solar power. Of note here is the deviation of the solar-power trajectory from the other two trajectories caused by the vehicle seeking the most advantageous use of the Sun's energy.

## Constant-Exhaust Power, Constant-Thrust-With-Coast Trajectories

In this analysis, the thruster is assumed to operate at constant $I_{sp}$ and constant exhaust power; in general, optimal trajectories admit one or more coast periods during the flight. Here it has been assumed that there is, at most, one such period of coasting. This work is also carried out in three dimensions.

The computation of optimal trajectories is accomplished under the assumptions of Melbourne and Sauer (Ref. III-11) to the effect that:

a. The minimum value of $\int_0^T a^2\, dt$ is invariant with respect to the powerplant fraction, $\mu_W$, the ratio of powerplant mass to initial mass.

b. The average thrust acceleration, $\bar{a}$, over a trajectory with minimum $\int_0^T a^2\, dt$ also is invariant with respect to $\mu_W$.

The Newton-Raphson algorithm numerically solves for the trajectory and steering schedule yielding a minimum value of J based upon assumed values of parameters that specify the jet exhaust velocity, c, and $\mu_W$. A second routine, utilizing this value of J and the average thrust acceleration, $\bar{a}$, of this trajectory, solves for improved approximations of c and $\mu_W$ from two expressions based upon the assumptions (a) and (b). This process is repeated until it converges to values of c and $\mu_W$ that yield a maximum value of the payload fraction, $\mu_{pl}$, the ratio of payload mass to initial mass.

Table III-I illustrates a comparison of solutions generated in this manner with the tabulated results of Melbourne and Sauer for a series of Mars rendezvous missions performed by the constant-thrust-with-coast mode (Ref. III-11). In the table, $\mu_1$ is the ratio of total final mass to initial mass. In general, the two solutions show good comparison.

REFERENCES

III-1. McGill, R. and P. Kenneth: Solution of Variational Problems by Means of a Generalized Newton-Raphson Operator. AIAA J. 2, 1761-1766, 1964.

III-2. Kopp, R. E., et al: Several Trajectory Optimization Techniques. Computing Methods in Optimization Problems, Balakrishnan and Neustadt, Ed. Academic Press, New York, 1964.

III-3. Kantorovich, L. V. and G. P. Akilov: Functional Analysis in Normed Spaces. Chapter XVIII, Pergamon Press, 1964.

III-4. Hestenes, M. R.: Numerical Methods of Obtaining Solutions of Fixed End Point Problems in the Calculus of Variations. Rand Corporation Report RM-102, August 1949.

III-5. Kalaba, R.: On Nonlinear Differential Equations, The Maximum Operation, and Monotone Convergence. J. Math. Mech. 8, 519-574, July 1959.

III-6. McGill, R. and P. Kenneth: A Convergence Theorem on the Iterative Solution of Nonlinear Two-Point Boundary Value Systems. XIV International Astronautical Federation Congress, Paris, France, September 1964.

III-7. Varga, R. S.: Matrix Iterative Analysis, Chapter 6, Prentice-Hall, 1962.

III-8. Van Dine, C. P.: An Application of Newton's Method to the Finite Difference Solution of Nonlinear Boundary Value Systems. United Aircraft Research Laboratories Report UAR-D37, March 1965.

III-9. Van Dine, C. P., W. R. Fimple, and T. N. Edelbaum: Application of a Finite-Difference Newton-Raphson Algorithm to a Problem of Low-Thrust Trajectory Optimization. Vol. 17, Progress in Astronautics and Aeronautics, Methods of Astrodynamics and Celestial Mechanics. Edited by R. L. Duncombe and V. G. Szebehely, Academic Press, May 1966.

III-10. Ragsac, R. V.: Study of Low-Acceleration Space Transportation Systems. United Aircraft Research Laboratories Report D-910262-2, March 1965.

III-11. Melbourne, W. G. and C. G. Sauer, Jr.: Payload Optimization for Power-Limited Vehicles. Vol. 9, Progress in Astronautics and Aeronautics, Academic Press, 1963.

TABLE III-1

COMPARISON OF VEHICLE PARAMETERS FOR MARS RENDEZVOUS MISSION:

RESULTS OF MELBOURNE AND SAUER VS RESULTS OF THE MODIFIED NEWTON-RAPHSON ALGORITHM

| T (days) | $\alpha$ (kg/kw) | $\mu_{p1}$ | | $\mu_W$ | | $\mu_1$ | | C x $10^{-2}$ (km/sec) | | $\int_0^T a^2\,dt$ ($m^2/sec^3$) | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| | | M-S | N-R | M-S | N-R | M-S | N-R | M-S | N-R | M-S | N-R |
| 270 | 1 | 0.935 | 0.931 | 0.033 | 0.035 | 0.967 | 0.965 | 1.847 | 1.620 | 2.178 | 2.450 |
| | 5 | 0.854 | 0.844 | 0.075 | 0.081 | 0.928 | 0.924 | 0.834 | 0.730 | 2.180 | 2.450 |
| | 10 | 0.791 | 0.775 | 0.110 | 0.122 | 0.901 | 0.897 | 0.602 | 0.526 | 2.185 | 2.450 |
| | 20 | 0.695 | 0.668 | 0.171 | 0.194 | 0.866 | 0.863 | 0.443 | 0.387 | 2.199 | 2.450 |
| 240 | 1 | 0.922 | 0.920 | 0.039 | 0.040 | 0.961 | 0.960 | 1.653 | 1.480 | 3.085 | 3.240 |
| | 5 | 0.826 | 0.820 | 0.089 | 0.094 | 0.915 | 0.914 | 0.751 | 0.668 | 3.090 | 3.240 |
| | 10 | 0.750 | 0.740 | 0.133 | 0.143 | 0.834 | 0.883 | 0.546 | 0.483 | 3.103 | 3.250 |
| | 20 | 0.634 | 0.613 | 0.208 | 0.232 | 0.842 | 0.845 | 0.408 | 0.359 | 3.136 | 3.250 |
| 210 | 1 | 0.904 | 0.904 | 0.048 | 0.048 | 0.952 | 0.951 | 1.488 | 1.358 | 4.758 | 4.840 |
| | 5 | 0.784 | 0.781 | 0.111 | 0.114 | 0.895 | 0.895 | 0.682 | 0.613 | 4.772 | 4.830 |
| | 10 | 0.690 | 0.682 | 0.166 | 0.176 | 0.856 | 0.859 | 0.500 | 0.444 | 4.804 | 4.830 |
| | 20 | 0.545 | 0.525 | 0.262 | 0.290 | 0.807 | 0.815 | 0.376 | 0.331 | 4.889 | 4.830 |
| 180 | 1 | 0.876 | 0.875 | 0.063 | 0.062 | 0.939 | 0.937 | 1.398 | 1.242 | 8.082 | 8.080 |
| | 5 | 0.722 | 0.719 | 0.144 | 0.147 | 0.866 | 0.866 | 0.632 | 0.561 | 8.118 | 8.100 |
| | 10 | 0.600 | 0.591 | 0.214 | 0.228 | 0.814 | 0.819 | 0.459 | 0.407 | 8.205 | 8.120 |
| | 20 | 0.413 | 0.386 | 0.338 | 0.379 | 0.750 | 0.764 | 0.347 | 0.304 | 8.420 | 8.150 |

# MINIMUM J EARTH-MARS ROUNDTRIPS

## VARIABLE LOW-THRUST RENDEZVOUS TRAJECTORIES

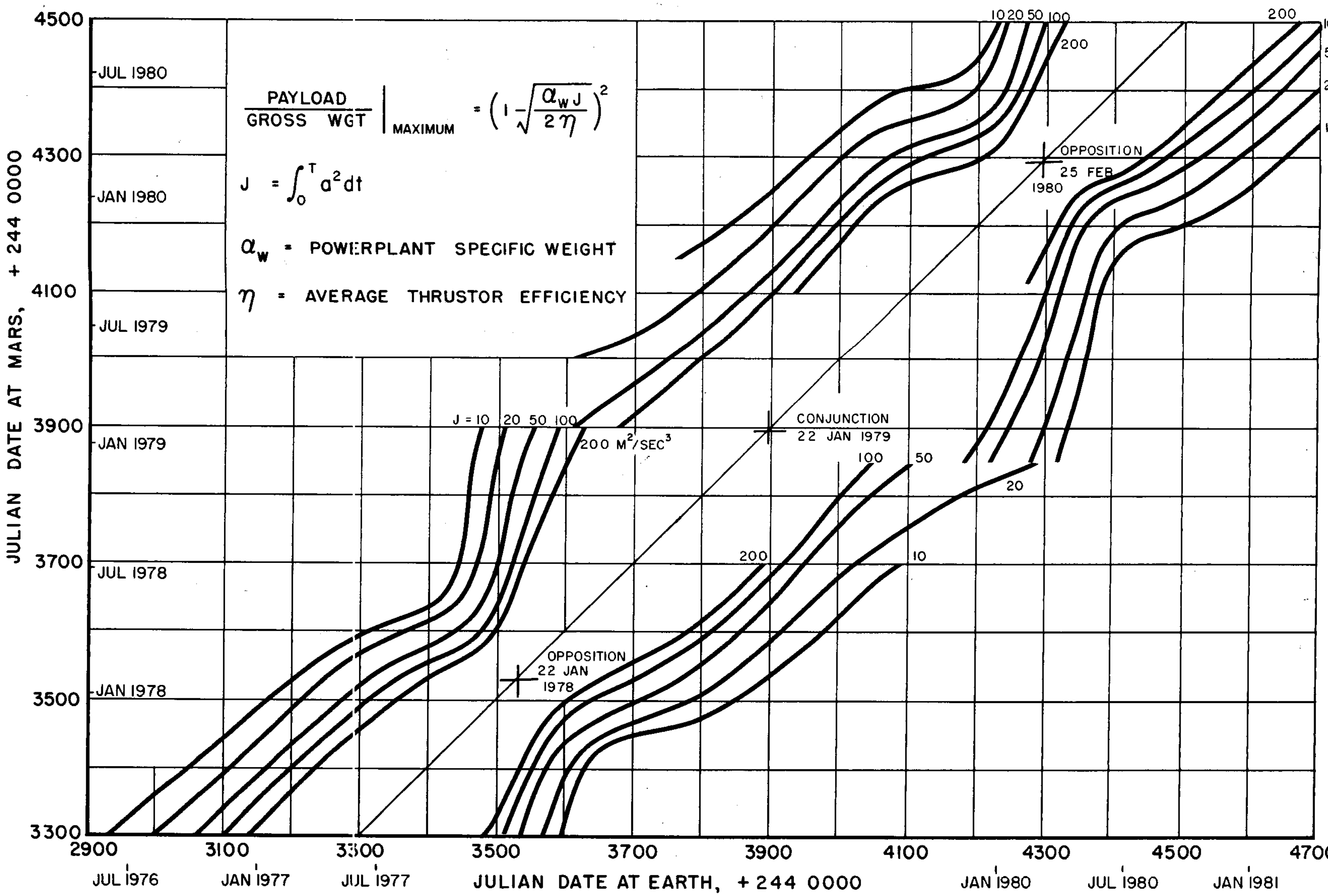
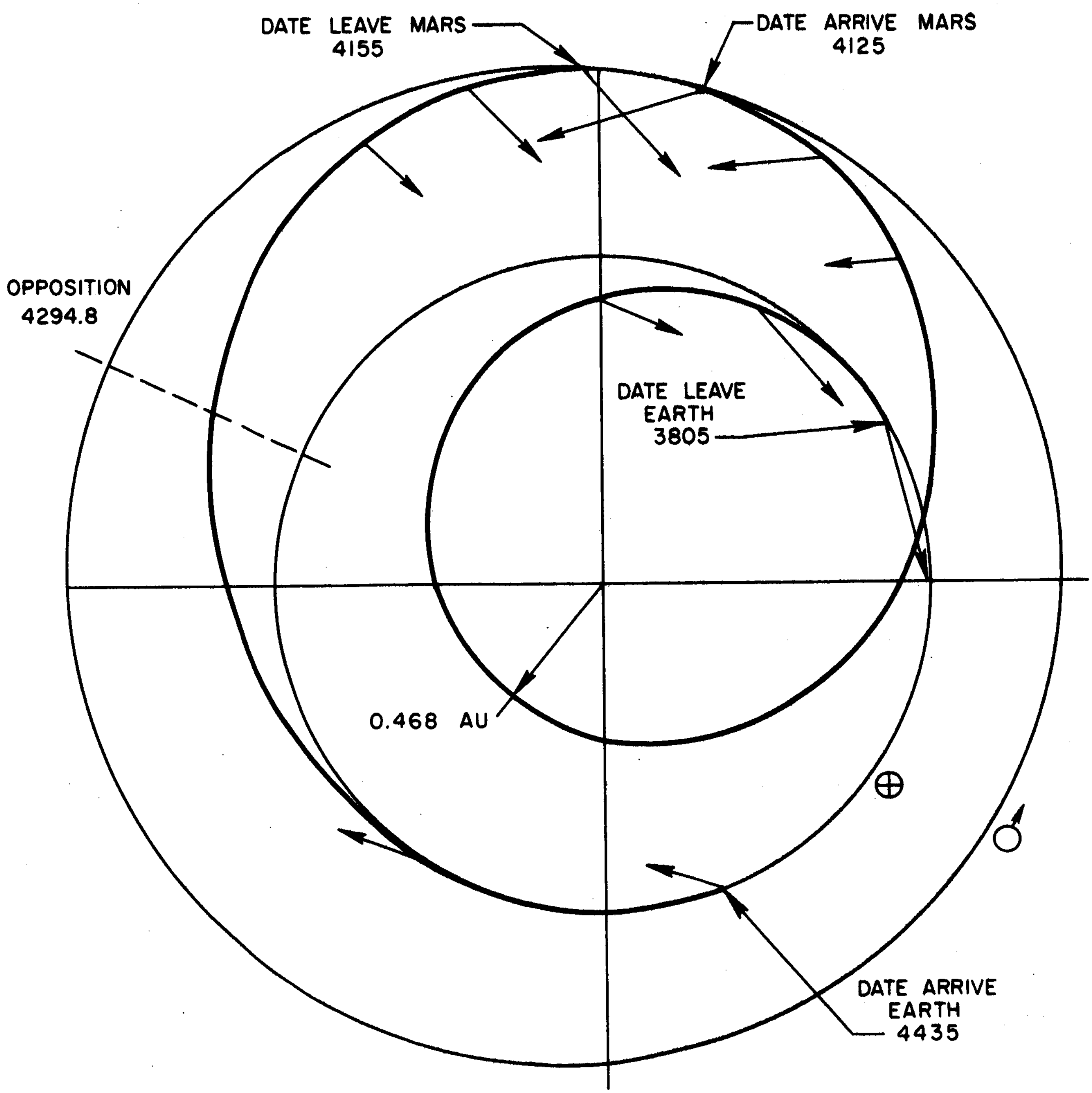



FIG. III-1

# MARS ROUND TRIP ARRIVING BEFORE OPPOSITION

TOTAL TRIP TIME = 630 DAYS

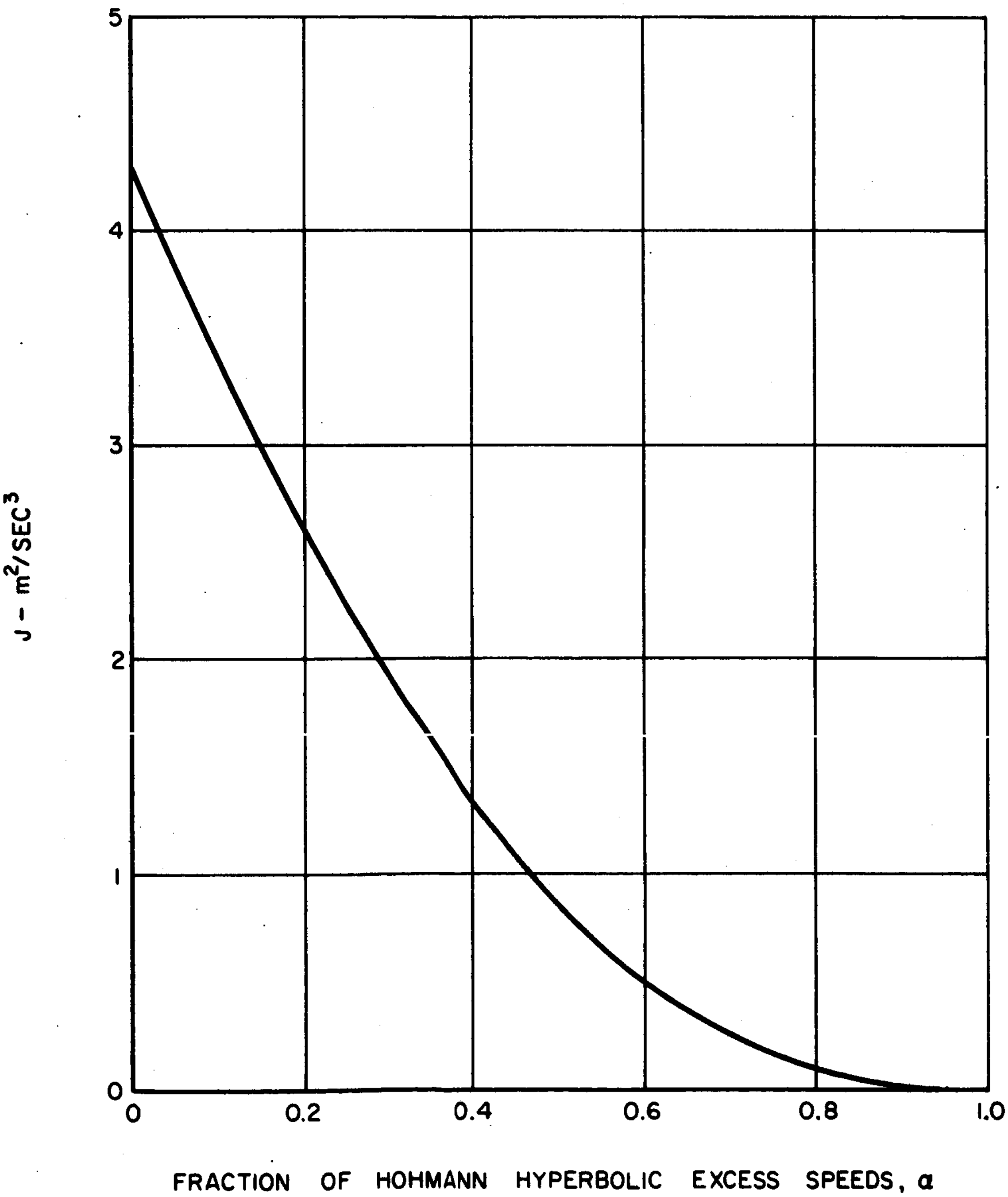


# MIXED-THRUST HOHMANN TRANSFER

INITIAL CIRCULAR ORBIT RADIUS = 1.0 A.U.

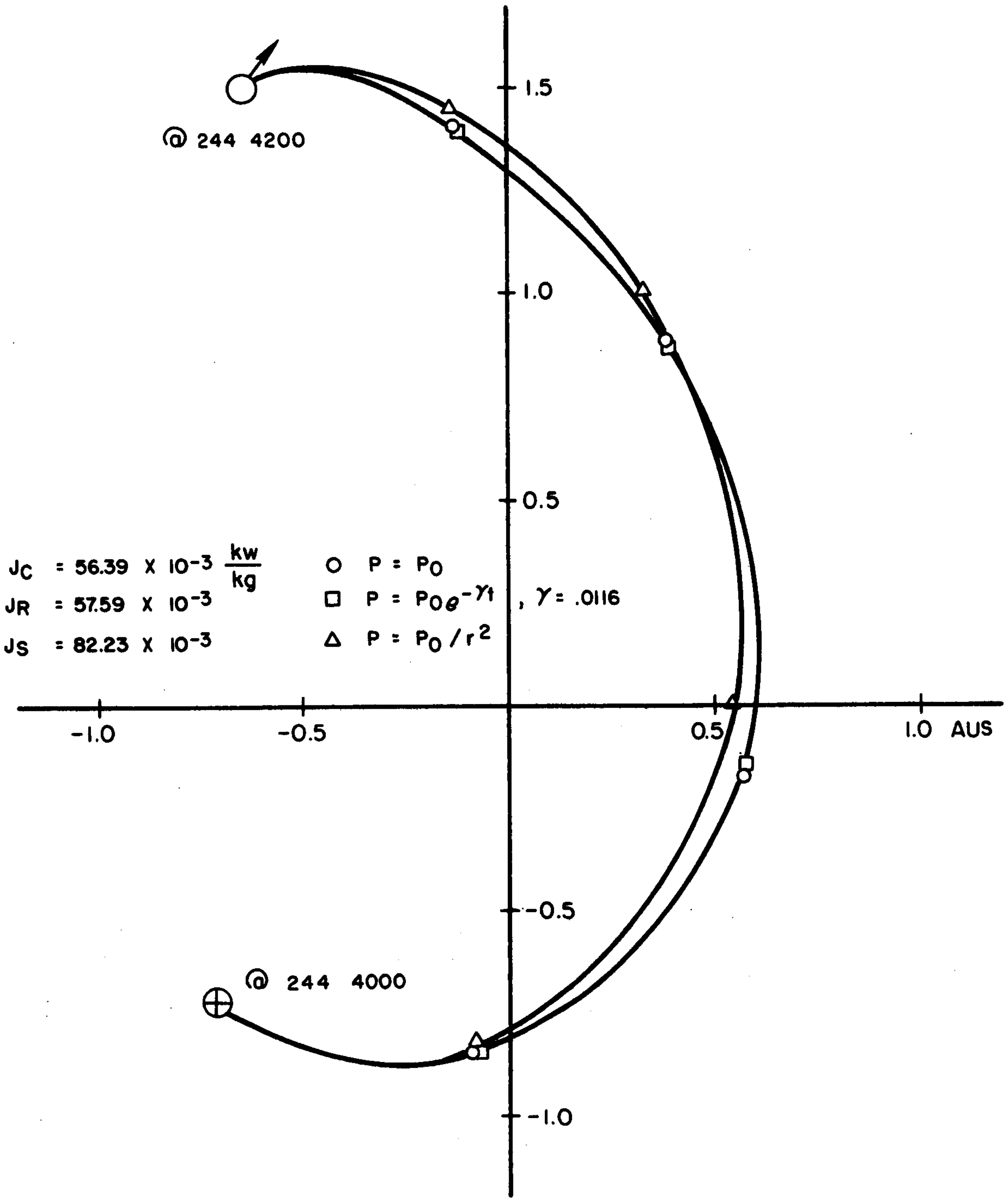
FINAL CIRCULAR ORBIT RADIUS = 1.523 A.U.

TRANSFER TIME = 258.74 DAYS



# OPTIMAL LOW-THRUST TRAJECTORIES

## MARS RENDEZVOUS



# SECTION IV
# POWER SYSTEMS STUDY

## Introduction

The objectives of the power systems study are to:

1. Determine the required characteristics of nuclear Rankine cycle space powerplants for low-acceleration manned Mars missions.

2. Determine the technological areas which can most significantly affect the powerplant characteristics through successful advanced research and technology requirements.

The work reported here is an extension of earlier work and includes:

a. Detailed powerplant design,

b. An evaluation of powerplant reliability including the determination of powerplant redundancy and the effects of being able to maintain the powerplant,

c. The determination of mission requirements (as measured by mass on Earth orbit - MEO) as a function of various powerplant characteristics,

d. An evaluation of the development program required for a nuclear Rankine cycle space powerplant.

The classical method for relating powerplant and mission performance has been by means of powerplant specific weight. The great significance of the work reported here is the identification of an additional powerplant characteristic and the development of methods for relating this characteristic to trajectory and mission requirements. This additional characteristic is the probable change of available system power output with time.

This change in available system power output with time is determined by a statistical analysis of the failure probability of the system components. The basis of this statistical analysis is the assumption that the components can be developed to achieve some stated level of technology. In this study two different levels of development were evaluated, the level of development of the aircraft gas turbine and the development corresponding to a factor of ten increase in the level of reliability. This technique allows the determination of the technological goals and the corresponding development required for the nuclear Rankine cycle system to perform the manned Mars mission.

## Powerplant Design

### Description of Reference System

The powerplant concept which was used in this study is a three-loop liquid metal system as shown schematically on Fig. IV-1.

The powerplant module was:

a. designed to fit within the payload configuration of the Saturn V using the SII stage as the orbital injection stage, and

b. arranged to provide maximum maintenance access to the powerplant components.

The resulting powerplant arrangement is shown on Fig. IV-2. The reactor is located in the nose of the payload stage in order to maximize the distance between the crew and the reactor and to minimize the diameter of the reactor shield. The reactor shield is directly behind the reactor.

The primary system (boilers, pumps, accumulator, and piping) is placed in an enclosure just behind the shield. The power-conversion systems and the electrical systems are placed at the rear of the payload envelope in a relatively cool, low radiation dose rate region in order to improve maintenance access. The power-conversion systems are placed at the rear of the vehicle in four separate compartments. A shirtsleeve maintenance environment can be provided in any one compartment with the other three power conversion systems operating. A closed breathing apparatus will be required for the maintenance personnel, however, because of the possibility of refractory metal corrosion by minute quantities of oxygen. In addition, environmental cooling of this compartment is required. Piping for the potassium vapor and condensate connects the primary system and the power-conversion systems. The vapor piping is placed in an insulated duct in order to reduce heat loss from the working fluid. Additional shielding is placed at the rear of the primary system enclosure in order to reduce the bremsstrahlung dose rate in the mission module, caused by lithium activation, to a reasonable level.

The heat rejection system has a conical-cylindrical configuration which has the same dimensions as the payload envelope. The radiator is used as structural support for the powerplant during launch and space flight. Additional support is supplied by a fairing which is attached to the radiator during launch and is ejected while the powerplant is on Earth orbit. The forward group of main heat rejection radiator segments is conical in shape. The remaining two groups of segments, as well as the auxiliary and low-temperature segments, are cylindrical in shape. The low-temperature radiator is at the rear of the powerplant while the auxiliary radiator is between the aft main radiator segments and the low-temperature segments.

The specific weight of this powerplant is 30 lb/kwe including a shield specific weight of 15 lb/kwe.

## Reliability Considerations

In order to plan the powerplant development it is necessary to have an understanding of the reliability requirements for the components and the system. An evaluation of powerplant reliability has been made which uses power availability as a measure of reliability. For any particular power system of interest an estimate of available power as a function of time can be made and this estimate can be used as the basis of a calculation of the mass-on-Earth-orbit requirements for the mission. Thus the reliability aspects of powerplant design can be evaluated on the basis of the effect of reliability on mission performance. In turn, powerplant development requirements are a measure of the technology required to achieve powerplant performance and reliability goals. In this study component failure rates are used as a measure of system development requirements.

This summary considers the following aspects of powerplant reliability:

a. The effect of component failure rate level,

b. The effect of multiple subsystems and systems,

c. The effect of being able to maintain, repair, or replace components.

The over-all method for evaluating these effects is:

a. Estimate component failure rate,

b. Estimate power availability as a function of time associated with a particular powerplant arrangement, maintenance capability, and probability,

c. Using the power availability and an associated specific weight calculate the mass required in Earth orbit to perform the mission.

### Failure Rate Study

The calculation of powerplant reliability requires the establishment of component failure rates. Insufficient data exist at this time to make a prediction of failure rates for nuclear Rankine cycle space powerplant components. Therefore, the questions to answer are: what failure rates will result in acceptable powerplant performance, and what is the effect of failure rate on powerplant performance? Approaching the question of powerplant reliability in this fashion will enable a determination to be made of the relationship between powerplant performance and the development program which may result in a given powerplant performance.

### Mass Required On Earth Orbit (MEO)

The power availability curve, and the powerplant specific weight are used in the trajectory and mission analysis to evaluate the mission effectiveness (as measured by MEO) of various alternatives associated with powerplant design and development.

### Multiple Subsystems

One method of increasing system reliability is by providing a number of components which perform the same function. In the Rankine cycle powerplant this approach results in providing multiple subsystems. Two extreme examples of multiplicity are presented by a single-thread system and the dual-reactor reference system. The single-thread system represents the ultimate reduction in the number of components for a three-loop system (except for the multiple radiator segments). Loss of any component in the system, aside from one or two radiator segments, will result in a complete loss of system output.

The corresponding power availability curves for these two systems are shown on Fig. IV-3 for the reference failure rate, $\lambda$, and without any consideration of maintenance. In addition, the power availability is shown for the dual-reactor system with improved technology ($\lambda/10$) failure rates. Two conclusions can be drawn from these results.

1. A single-thread system cannot provide sufficient power availability for the mission.

2. Provision of multiple components improves power availability significantly. However, sufficient power availability is achievable only through a combination of multiple components and the lower ($\lambda/10$) failure rates (without considering maintenance).

### Maintenance Effects*

Since the preceding results indicate the difficulty of obtaining significant power availability it is of interest to examine the benefits which might be obtained if maintenance could be performed. The evaluation presented here was limited to determining the potential benefits if it were assumed that maintenance operations are feasible.

In order to more completely determine the potential benefits from maintenance it is necessary to evaluate:

* Including maintenance, installed spares, and replacement or repair.

a. The improvement in power availability,

b. The specific weight penalty associated with maintenance.

Both of these factors have been evaluated and used to determine MEO for a variety of situations. Figure IV-4 shows the improvements in power availability made possible by maintenance for the system demonstrating improved failure rates ($\lambda/10$). This information was used to determine the MEO requirements shown on Fig. IV-5 for maintained and nonmaintained situations.

This information points out a very significant aspect of being able to perform maintenance. The MEO requirements are the same for a nonmaintained powerplant with a specific weight of 20 lb/kwe as for a maintained powerplant with a specific weight of 32 lb/kwe (including 2 lb/kwe for spares). Thus, the same mission effectiveness can be obtained by either developing a low-specific weight powerplant (with the implication of an expensive development program) or by developing the heavier system with a maintenance capability (implying a cheaper development program).

The power availability associated with a maintained, reference ($\lambda$) failure-rate system is shown on Fig. IV-6. These results show that a more extensive maintenance program is required for the $\lambda$ systems (120 rather than 42) in order to optimize the MEO requirements. A comparison of the MEO requirement for maintained $\lambda$ level and $\lambda/10$ level systems is shown on Fig. IV-7. This information shows that the MEO penalty associated with the maintained $\lambda$ level system is only 60,000 lb. This leads to the significant conclusion that the components achieving aircraft gas turbine development level are adequate if an extensive maintenance program can be developed. A complicating factor is that this level of maintenance will require a rapair operation every four days. However, even if the maintenance level is the same for the $\lambda$ and the $\lambda/10$ cases (42), the penalty for the $\lambda$ case causes a MEO difference of only 130,000 lb (less than one Saturn V launch vehicle). As will be shown later, the development cost involved in improving component failure rates by a factor of ten is much greater than the cost of a Saturn V.

To perform maintenance on liquid-metal components, the equipment must be accessible, the crew must be protected from the thermal and nuclear environment so that it is safe to work on the equipment, and the equipment probably must be cooled and drained of liquid metal. In the event of an equipment failure and resulting power reduction, about 1 to 2 hours are available for diagnosis and corrective action before the liquid-metal systems must be drained in order to avoid freeze-up in the radiators. If failure from meteoroid penetration should occur, even less time will be available to diagnose the fault and drain the failed system. Therefore a liquid-metal drain-and-fill system will be required in order to accomplish the repair of liquid-metal components. This system will have to incorporate many valves, controls and liquid-metal containment vessels. The question of the reliability of this system (including the required control system) requires examination to determine the real incentive for providing this system.

## Dependence of Powerplant Performance on Technology

Many choices must be made in the process of developing a nuclear Rankine cycle powerplant. All of these choices depend on the technology which can be achieved during the development program. The range of choice is usually narrowly restricted by a number of technological considerations and the influence of the choice on system performance (specific weight) can usually be predicted. An extensive study has been made of the influence on specific weight of many of the powerplant parameters. This study shows that most of the powerplant parameters have a relatively small influence on specific weight within a reasonable range of selection. However, three powerplant parameters stand out in terms of the effect which they have on specific weight. These are: (1) turbine inlet temperature, (2) reactor fuel burnup, and (3) radiator materials selection.

### Turbine Inlet Temperature

The selection of both reactor operating temperature and turbine inlet temperature is a very significant decision. While it is desirable to pick these temperatures as high as possible, recognition must be made of the physical limitation imposed by material properties, the development difficulties at high temperature, and the expected decrease in reliability as the temperature is increased. At the time a selection must be made a balance must be drawn between the improvements in performance which are possible at higher temperatures and the realities of hardware development. As a part of this study, an attempt has been made to shed some sort of quantitative light on this selection.

The influence on MEO of turbine inlet temperature (as reflected by powerplant specific weight and failure rate) has been evaluated. Figure IV-8 shows this effect for two different assumptions. One assumption (probably optimistic) is that the failure rate of the high-temperature components does not depend on the turbine inlet temperature. The other assumption (probably pessimistic) is that the failure rate of the high-temperature components does depend on temperature. These results indicate that, regardless of which assumption is used, the incentive for increasing turbine inlet temperature is not great. Indeed, using the optimistic assumption decreases MEO by only 10% when increasing the turbine inlet temperature from 1600 F to 2135 F for a 530-day mission. This is an insignificant amount from an over-all mission standpoint. As the mission time decreases the dependence of MEO on specific weight, and, therefore on turbine inlet temperature, increases. For instance, at 430 days the same change in turbine inlet temperature as above will change MEO by about 20%.

On the other hand, the dependence of MEO on specific weight is smaller for a maintained powerplant than a nonmaintained powerplant. All of these factors tend to lead to the conclusion that powerplant development risk can be decreased by choosing a relatively low turbine inlet (and therefore reactor outlet) temperature with only a small mission penalty.

## Reactor Fuel Burnup

Past experience with nuclear systems indicates that development of the reactor fuel is usually the most technically difficult and costly item in the system. In addition, the burnup which can be achieved has a large effect on the weight of the system. Therefore, the amount of burnup which can be achieved becomes a significant consideration in determining the development program requirements for the system under consideration. It is particularly significant that the high-performance systems under consideration here assume a combination of burnup and fuel temperature which exceeds anything which has yet been demonstrated in an operational reactor.

Figure IV-9 shows the relationship between MEO and reactor fuel burnup for two different powerplant probabilities and for two different assumptions regarding powerplant maintenance. The conclusion which can be drawn is that reactor fuel burnup has a significant effect on MEO. Over the range, evaluated burnup can change MEO by between 0.5 to 1.5 x $10^6$ pounds.

## Influence of Powerplant Technology on MEO

As indicated by some of these results, powerplant specific weight is very sensitive to the technological level, as reflected by temperature, burnup, radiator materials, etc., which can be achieved. The effect on MEO of this variation in technology is shown on Fig. IV-10 for both 430- and 530-day missions. These results are based on the optimistic assumption that the failure rate is not influenced by temperature. As the trip time increases, the advantages of improving powerplant technology become smaller. Making the technology improvements shown reduces MEO by about 30% at 430 days and by about 10% at 530 days. These results indicate that further work is required in the area of mission parameter selection before firm conclusions can be drawn regarding powerplant parameter selection. However, the results indicate that the longer trip time (530 days) results in a significant reduction in MEO.

## Development Program

In order to place decisions regarding the powerplant in proper perspective, it is necessary to make an estimate of the powerplant development cost. A first-order analysis has been made of a development program for a nuclear Rankine cycle powerplant. It should be recognized that the development program estimates for advanced systems are subject to large uncertainties because of a lack of experience with the particular system being estimated. This is particularly true for systems having development cycles lasting for 10 to 20 years. With this qualification in mind, the study reported here has been made to give the approximate magnitude of the development program cost and time, some idea of the sensitivity of cost and time to the assumptions used, and some idea of cost trends related to technological goals.

The work reported here has been based on a Rand study reported by Pinkel.* The Rand work was based on a three-loop 2000 F reactor outlet temperature Rankine cycle system similar to the baseline system used in this study. The Rand work consisted of estimating:

a. A reference development program schedule,

b. Ground and flight test requirements and costs,

c. Hardware costs for various types of subsystems,

d. Facility requirements and costs.

In addition, the Rand study evaluated the sensitivity of the program cost to various assumptions and alternatives. The study reported here used the aircraft gas turbine development experience to check various aspects of the Rand work. It should be noted that the resulting estimates substantially agree with comparative cost estimates given by Pinkel.

P&WA Aircraft Gas Turbine Development

The gas turbine experience has been based on a technology background which has been developed over the last 20 years. During this time the development effort has taken two separate but related forms.

One is the effort spent directly on specific engine system projects. The other is the effort spent on general component technology acquisition and improvement. Both of these activities contribute to the attainment of the current performance and duration levels. The attainment of these levels has been a gradual process of increasing performance (as reflected by turbine inlet temperature) and reliability (as reflected by component failure rate). The achievement of these levels did not occur through development of a single system designed to meet the current requirements but was an evolutionary process which grew from an accumulation of

* Pinkel, B.: Electrical Propulsion in Space: Mission Comparisons, Development Cost, Reliability, and Their Implications for Planning. Memorandum RM-9056-NASA, August 1964.

technological experience. It should also be noted that the experience gained in over 70 million engine hours of flight operation also contributes substantially to the achievement of the present performance and endurance levels.

The general characteristics of gas turbine development time and cost experience are shown on Fig. IV-11. This figure shows the cumulative relative cost and the time required for developing three typical gas turbine engines. Superimposed on this information are lines showing failure rate milestones during the development program.

Nuclear Rankine Cycle Powerplant Development

A number of factors peculiar to nuclear space powerplants have been taken into account in estimating the development program requirements. Again it must be noted that these factors apply to the development of any type of high-performance, high-temperature nuclear space powerplant. The significant factors are:

1. Insufficient technology now exists to directly develop the components or the system. In addition, insufficient experience exists to now judge that the system is feasible. Therefore, the first stage of the development requires the establishment and the demonstration of the required technological base prior to initiation of full-scale development. Compared to full-scale development, this initial phase of the program is characterized by a relatively low funding rate. The program is designed so that the investment required to demonstrate system feasibility is relatively low. Numerous milestones will be built into this phase of the program to measure development progress and to continually evaluate the probability of achieving the program goals. The development risk is minimized until it has been demonstrated that a reasonable assurance exists that the program will be successful.

2. As mentioned above, attainment of desired component or system reliability cannot reasonably be demonstrated. Assurance that the system is reliable enough to perform the mission will depend on the manner in which the development program is organized and operated. The following are characteristics of a development program designed to attain the desired system characteristics:

a. The program will be performed by an experienced team with a record of success in developing high performance, high temperature, high reliability powerplants,

b. Sufficient program funding,

c. Clear and constant program goals,

d. Extensive testing of components and systems to eliminate sources of failure,

e. Extensive quality assurance program for design, materials, and manufacturing,

f. Selective assembly of systems,

g. Successful completion of specified system qualification tests.

3. It has been assumed that the initial acquisition and demonstration of basic technology will begin with an 1800 F reactor outlet temperature and a 1 a/o uranium burnup system. The process of technology acquisition and demonstration will proceed in increments of 200 F or 2 a/o burnup until the technology required for the operational system has been demonstrated. At this point full-scale development will be initiated for the flight system.

The reasoning outlined above led to the consideration of a three-phase development cycle:

a. Technology Demonstration

This phase will consist of accumulating and demonstrating that the necessary technology is available to achieve the required performance. In addition, a 2000-hr test will be made to demonstrate system feasibility and to pinpoint initial sources of wearout failures.

b. System Development

This phase of the program will consist of extending the endurance of the multiple component system to that required for the flight.

c. Flight Test

This phase of the program will consist of demonstrating system operation, performance, and endurance in the space environment.

The over-all schedule for an 1800-F reactor outlet temperature, 1 a/o uranium burnup system is shown in Fig. IV-12 to include the following:

a. A 5-yr Technology Demonstration program for this system (will consist of 3 years of subsystem and component development followed by a 2-year, 2000-hr demonstration test).

b. A 6-year System Development program

c. A 4-year Flight Test program

If all of these programs are conducted in series, the total time required will be 15 years. One alternative would be to perform a portion of the programs in parallel as shown on Fig. IV-12. This would reduce the development time to about 13 years. The feasibility and degree of performing these programs in parallel depends on the degree of success achieved in the various stages of development.

It has been estimated that the Technology Demonstration phase of the development of this system will cost \$60M/yr for 5 years. In addition, the facilities required for this program are estimated to cost \$70M for a CANEL type facility and an additional \$70M for a nuclear system test facility.

Thus the total requirements for the Technology Demonstration phase are estimated to be 13 to 15 years and about \$440M.

The System Development phase includes the program required to demonstrate the performance and endurance level of the baseline system. The requirements for this phase of the program are estimated at 6 years and \$4260M. This estimate was based on methods suggested by Pinkel (op. cit.) with check-points based on aircraft gas turbine experience.

The Flight Test phase of the program is intended to demonstrate that the system can achieve the required performance and endurance in space. The Flight Test program has been set up on the basis that it is cheaper to accomplish almost any test objective on the ground. Therefore, the number of flight tests have been minimized. It is assumed that the flight tests will be performed on orbiting space stations.

The test schedule has been set up to include:

1. Three single-system tests to demonstrate operational characteristics, startup and shutdown, endurance, and maintenance capability.

2. Three 10,000-hr dual powerplant endurance tests.

The requirements for the Flight Test phase have been estimated at 4 years and \$2070M.

The total development program requirements for the 1800 F, 1 a/o burnup system therefore are estimated to be about 13 to 15 years and \$6780M. A cost breakdown by phase is shown below:

| | | |
|---|---|---|
| 1. | Technology Demonstration | \$ 440M |
| 2. | System Development | 4260 |
| 3. | Flight Test | 2070 |

A curve of the rate of expenditure for this program is shown on Fig. IV-13. This shows the low level of funding during the Technology Demonstration phase and illustrates the relatively small commitment to the concept prior to demonstration of the feasibility of the system.

An obvious question which arises is: can reductions be made in the development time and money requirements? A number of possibilities are apparent; however, further evaluation is required before it can be concluded that significant reductions are possible. Some of these possibilities will be discussed below.

1. In order to place the development cost in proper perspective an estimate should be made of the number of missions to which the Rankine cycle technology will be applicable. While the work reported here was concentrated exclusively on the manned Mars mission, the Rankine cycle technology is obviously applicable to other programs. Thus the question of how the development costs should be apportioned or, alternatively, the total range of benefits received from the development should be considered. As a simplified example, Fig. IV-14 shows how the cost per trip varies with the number of manned Mars missions which are flown. This information illustrates that if enough missions can be flown, the development program costs become relatively insignificant. Thus, an extensive space program which has a large number of Rankine cycle applications in addition to the manned Mars mission would make a large powerplant development program appear to be reasonable.

2. One possibility for reducing development costs is to perform most of the Technology Demonstration phase on lower-power level components. It is anticipated that this would reduce the cost of this phase of the program without a significant loss in assurance that the results are applicable to the larger systems. One obvious exception to this generality is the reactor control system. Since reactor size does have a significant influence on reactor control this aspect of system development must be considered at the power level of interest.

One aspect of the size effect in shown on Fig. IV-15. This information is taken from Pinkel (op. cit.). It should be noted that the range of results obtained in this study bracket the extrapolated results from Pinkel.

3. Some consideration might be given to a greater degree of paralleling of the various phases of the program. The degree to which this can be done is very uncertain. One consideration will be the degree of success actually achieved during the program. Adjustments due to this factor will not be apparent until the programs are underway. However, it must be recognized that, in general, estimates of development time and cost are usually optimistic and experience usually increases both time and cost. The baseline system development presented here has attempted to recognize this factor and to present a realistic situation. However, a more optimistic schedule has been estimated as 13 years. The "series" and "parallel" schedules are compared on Fig. IV-12. These two schedules may be viewed as presenting the uncertainty involved in estimating the development time requirements. There is no apparent reduction in development cost resulting from this paralleling process.

The selection of performance goals for any advanced system is an exercise of judgement which must take into account many factors. It is desirable to quantify as many of these factors as possible even though there is a large degree of uncertainty in this process. For this reason estimates of development program requirements which have been considered are:

a. powerplant maintenance
b. component failure rate
c. system temperature
d. reactor fuel burnup
e. radiator materials

The final selection of the costs for various alternatives requires consideration of the cost trade-offs between development costs and mission costs. Some of the factors which influence the total cost are:

a. cost of obtaining various levels of powerplant technology,

b. cost of performing the mission with powerplants of different levels of performance, in terms of:

MEO,
Launch vehicle cost,
Number of missions.

Development programs corresponding to further advances in technology have been evaluated. An assumption has been made that Technology Demonstration proceeds in 4-yr program increments. These increments consist of either 200-F increases in reactor outlet temperature or 2-a/o increases in uranium burnup. Two different assumptions have been made regarding the cost of Technology Demonstration:

a. \$60M per year

b. Annual cost doubles for each new technology increment. That is:

1800 F, 1 a/o - \$60M/yr
1800 F, 3 a/o - \$120M/yr
2000 F, 3 a/o - \$240M/yr
2200 F, 3 a/o - \$480M/yr

The results of both the temperature and the burnup influences on system development costs are plotted on Figs. IV-16 and IV-17 (normalized to a 2200 F, 3-a/o burnup system). These results indicate that:

a. The total cost of the development program becomes extremely sensitive to the assumption of the cost for technology demonstration, as the technology becomes more advanced.

b. Regardless of the cost assumption there is an incentive toward developing relatively low-level technology (1800 F to 2000 F and 1- to 3-a/o burnup).

These conclusions are further demonstrated by the information shown on Figs. IV-18 and IV-19 for the optimistic assumption of Technology Demonstration cost (assumption a).

These figures show:

a. The development cost as a function of temperature or burnup amortized over 1 mission and 10 missions

b. The vehicle cost/mission as a function of temperature. The change in vehicle cost is a reflection of the change in MEO over the range of temperature and burnup examined (Saturn V cost has been assumed as $70M/vehicle).

These results illustrate that the total cost of the mission is not strongly affected by the powerplant technology. Thus, from the standpoint of development risk, there appears to be no strong incentive for developing the higher-level technology systems.

Reliability Improvements

As indicated previously, there is no clear way in which to perform a development program to achieve an improvement in component reliability. In addition, there is no feasible method for demonstrating the level of failure rate to which long mission time components have been developed. However, the reliability analysis indicates that one way in which to achieve reasonable power availability is to develop components to achieve an order of magnitude improvement in gas turbine reliability. Thus it is of interest to examine the reasonableness of attempting to achieve this improvement.

A development program designed to achieve improved component reliability will require increased quality assurance provisions, special handling provisions, and testing. There is no existing data which directly correlate the resulting increase in development cost with the resulting reliability improvements. However, gas turbine experience suggests that the cost of an engine development program increases by a factor of three to obtain a factor of ten reduction in failure rate. In addition to the money spent directly on each program, general technology improvement funds also contribute to the reduction in failure rate. This amounts to about another factor of two increase in development program cost. Therefore, it has been assumed that improving the reliability of the Rankine cycle will increase the System Development and Flight Test cost by a factor of 3 to 6 to reduce the failure rate by a factor of ten. This will result in program costs of:

| | | |
|---|---|---|
| a. | Technology Demonstration | \$ 980M to \$ 3,170M |
| b. | System Development | 10,400M to 20,800M |
| c. | Flight Test | 4,750M to 9,500M |
| | TOTAL COST | \$16,000M to \$33,000M |

Thus, it is fairly clear that this estimate predicts a drastic increase in development cost to achieve a $\lambda/10$ failure rate level. While this is a very crude estimate, the implication is clear that, with the assumptions used, an inflight maintenance capability is undoubtedly a more attractive alternative for improving power availability.

Development of Maintenance Capability

The cost required to develop an extensive maintenance capability has been estimated to be \$680M.

It has already been noted that a nonmaintained "$\lambda$" level system provides a high probability of reasonable power level for only a short period of time (less than the mission time of interest). Thus it becomes apparent that developing a maintenance capability makes the use of "$\lambda$" level system feasible with a large saving in development costs. This situation is summarized below for a 530-day mission:

| | $\lambda$ Level | $\lambda/10$ Level |
|---|---|---|
| | 30 lb/kwe system<br>6 lb/kwe spares | 30 lb/kwe system<br>2 lb/kwe spares |
| Development Cost (average) | 7690M | 16,000M to 33,000M |
| Launch Vehicle Cost | 398M | 364M |
| Total Cost | 8000M | 16,000M to 33,000M |

Thus the trend is clear that a maintained, aircraft gas turbine failure rate level system represents a reasonable Rankine cycle system development goal.

Alternate Radiator Materials

An evaluation has been made of the effect of the choice of radiator materials on the system development cost. The choice evaluated is between a radiator with beryllium barrier and fins and a radiator with copper fins and stainless steel barrier. The cost of the development program was estimated to be \$6030M for the beryllium radiator and \$5900M for the copper-stainless steel radiator. This estimate, while confirming an intuitive judgement that the copper-stainless steel development is cheaper, does not show a significant difference between these two choices.

# 4 MW(e) NUCLEAR RANKINE CYCLE SPACE POWERPLANT

# SYSTEM SCHEMATIC

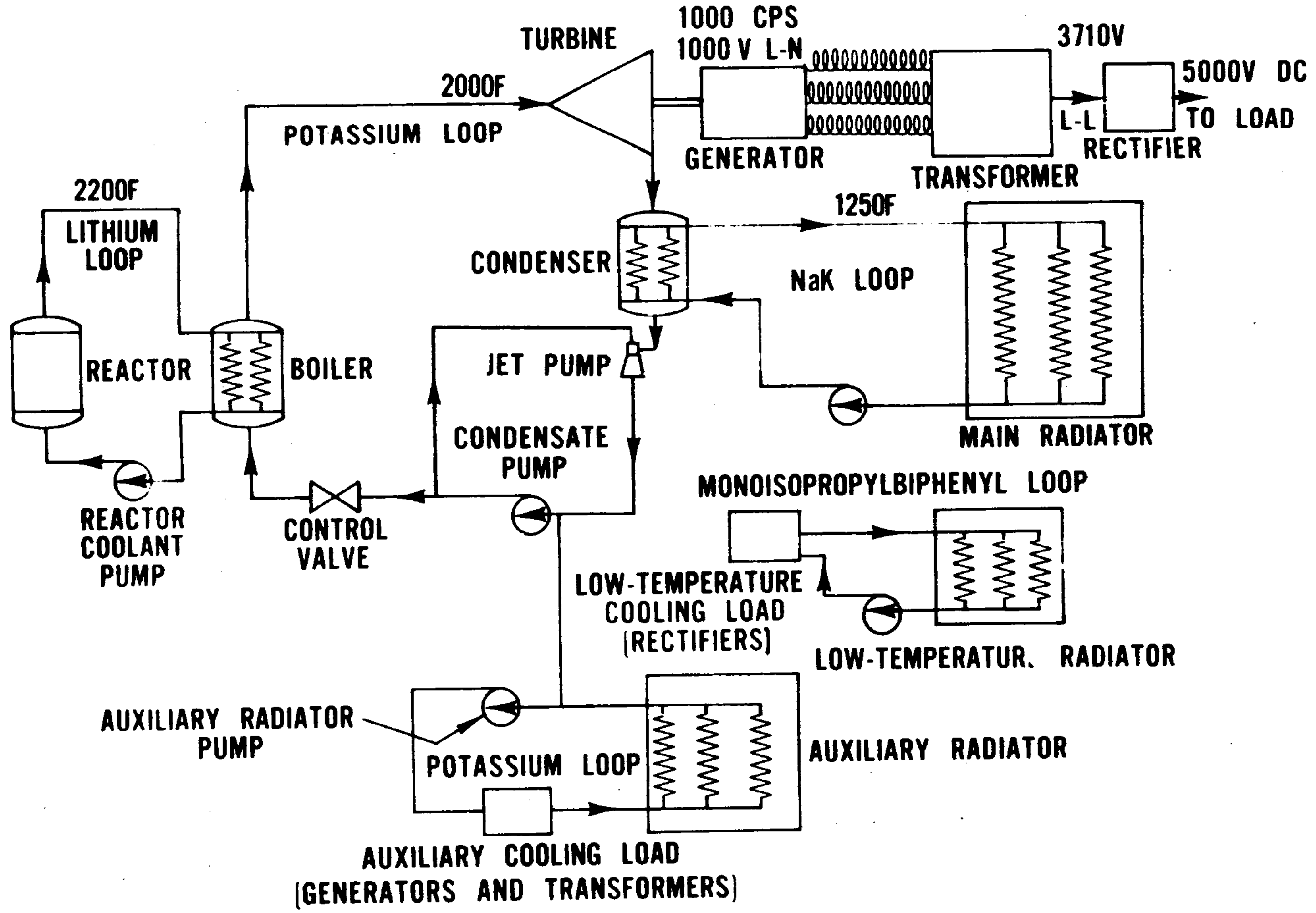
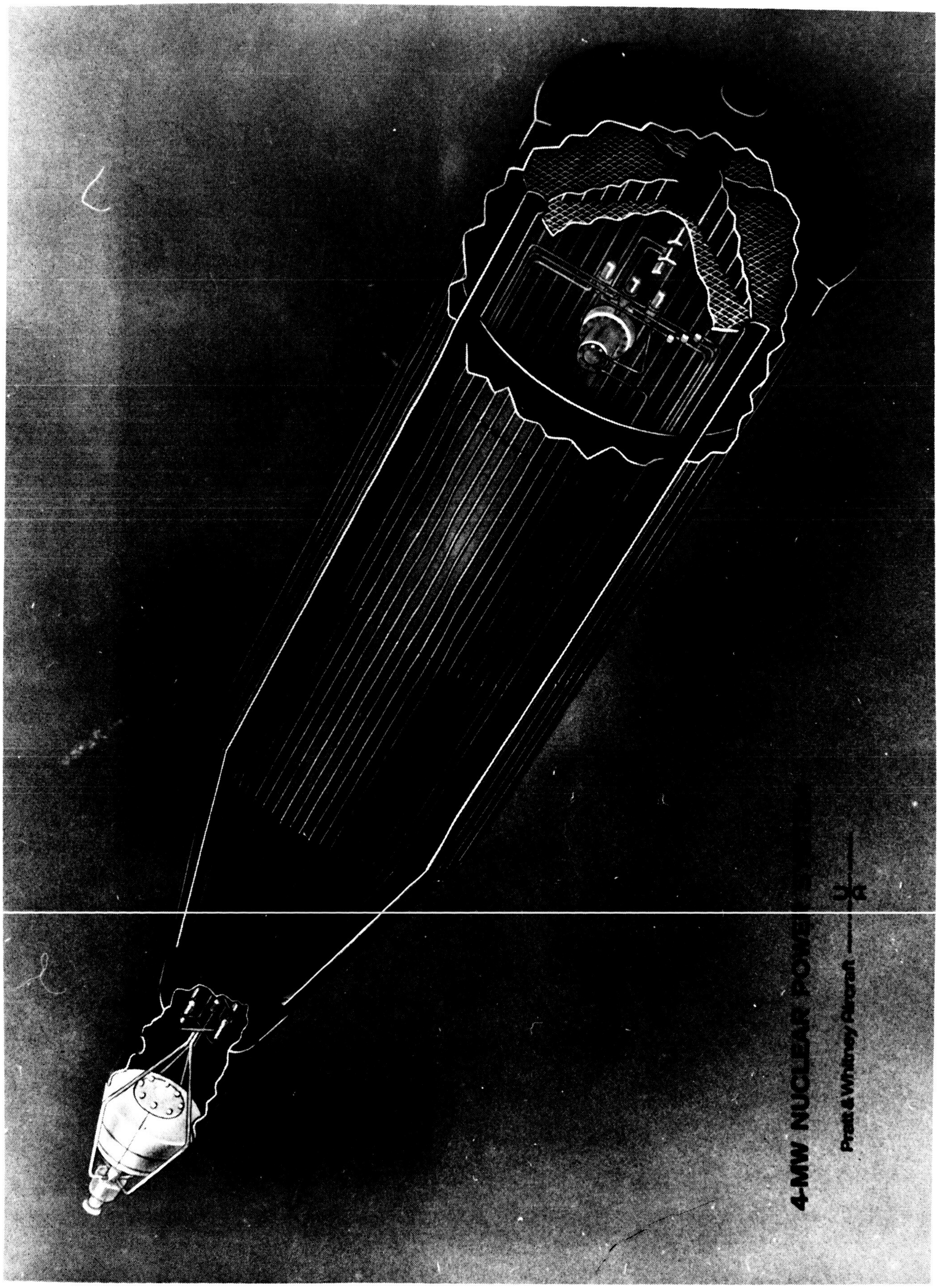
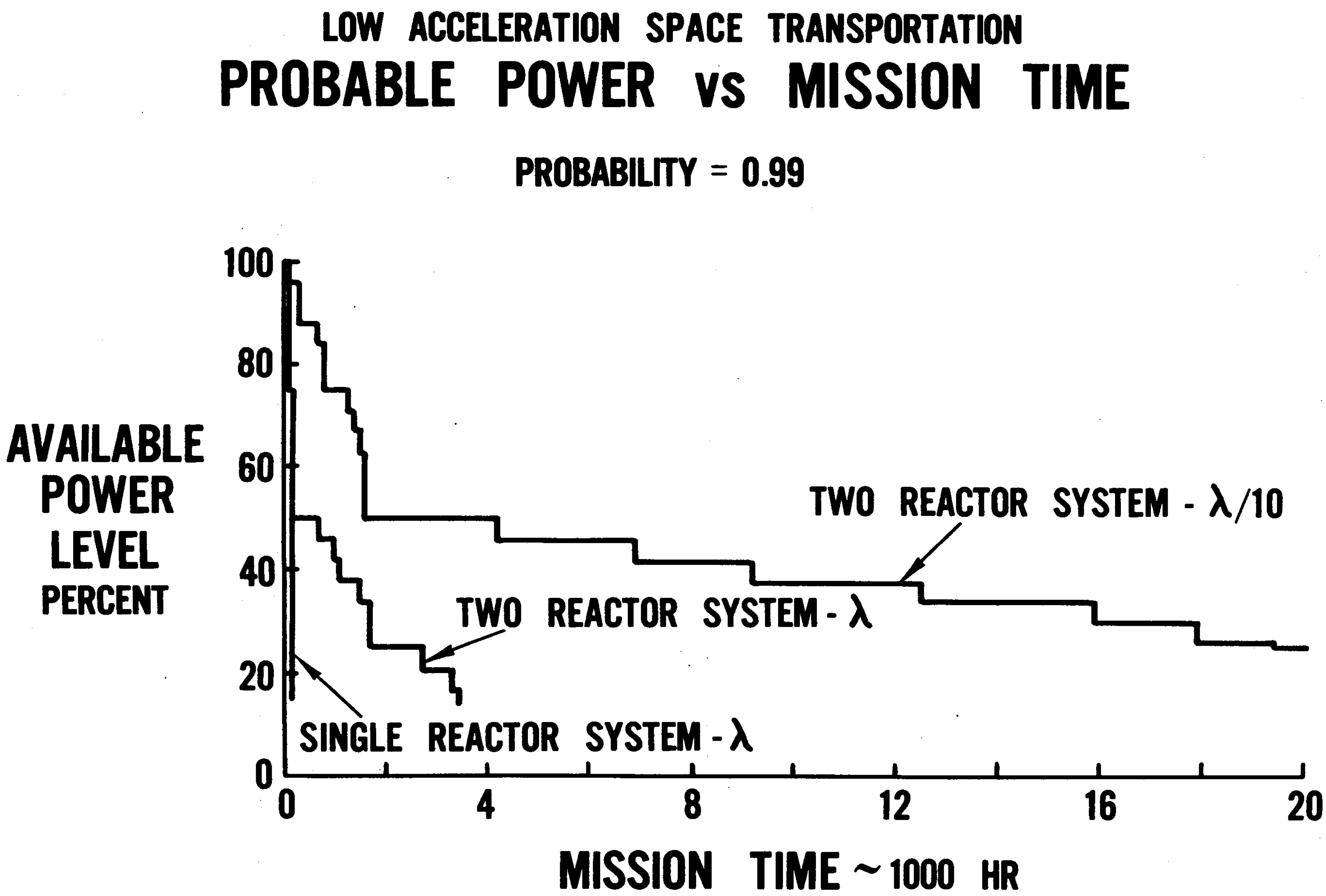


FIG. IV -2



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M-37441
661003

# LOW ACCELERATION SPACE TRANSPORTATION

## EFFECT OF MAINTENANCE ON AVAILABLE POWER

IMPROVED FAILURE RATES PROBABILITY = 0.99

ALL COMPONENTS MAINTAINABLE EXCEPT REACTOR

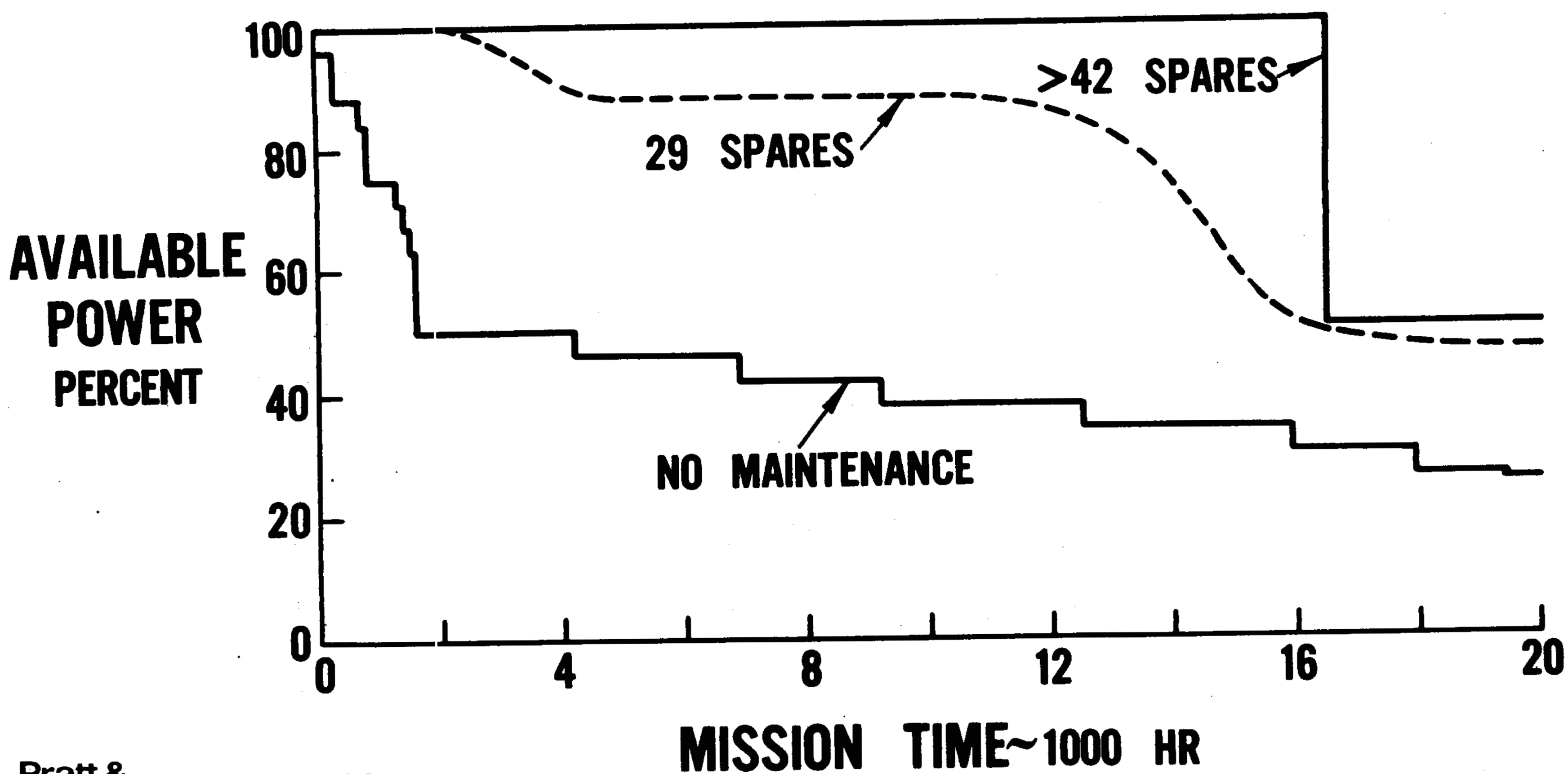
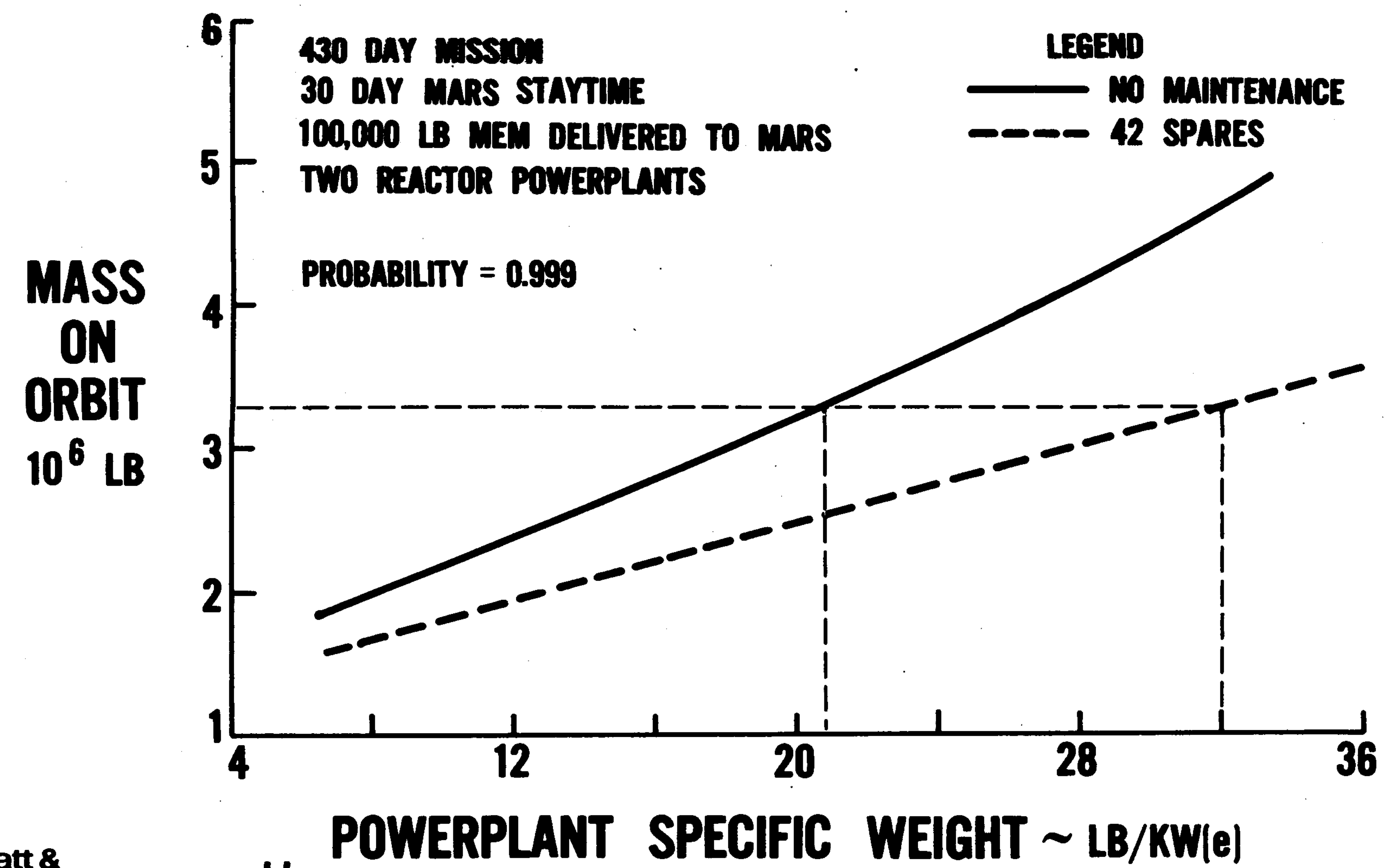


Figure IV-5

LOW ACCELERATION SPACE TRANSPORTATION

# EFFECT OF POWERPLANT ON VEHICLE MASS

OPTIMUM NUCLEAR ROCKET AND ELECTRIC PROPULSION OPERATION

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M-38729
680906

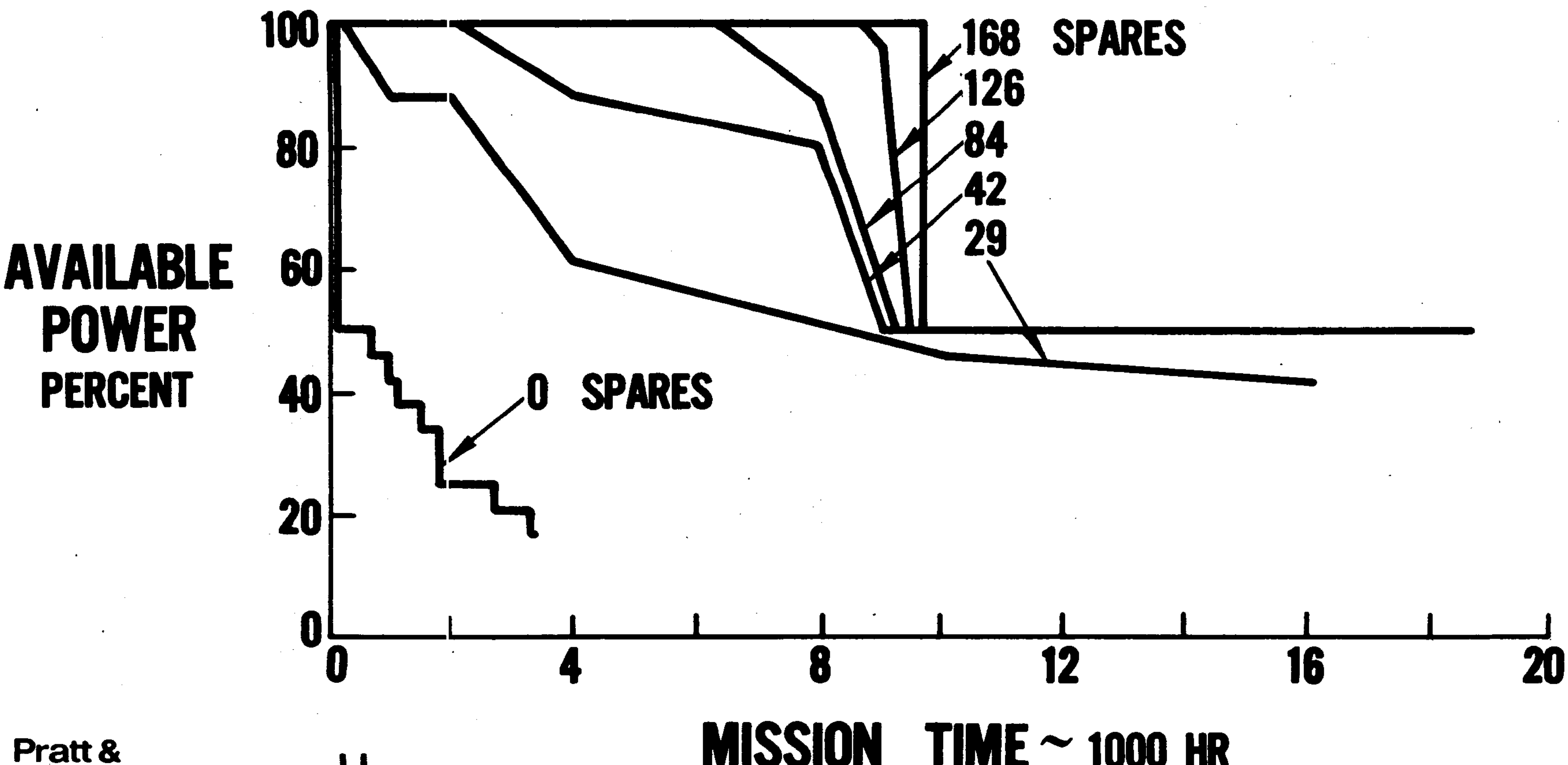
Figure IV-6

## LOW ACCELERATION SPACE TRANSPORTATION

# EFFECT OF MAINTENANCE ON AVAILABLE POWER

### AIRCRAFT GAS TURBINE FAILURE RATES PROBABILITY = 0.99

### ALL COMPONENTS MAINTAINABLE EXCEPT REACTOR

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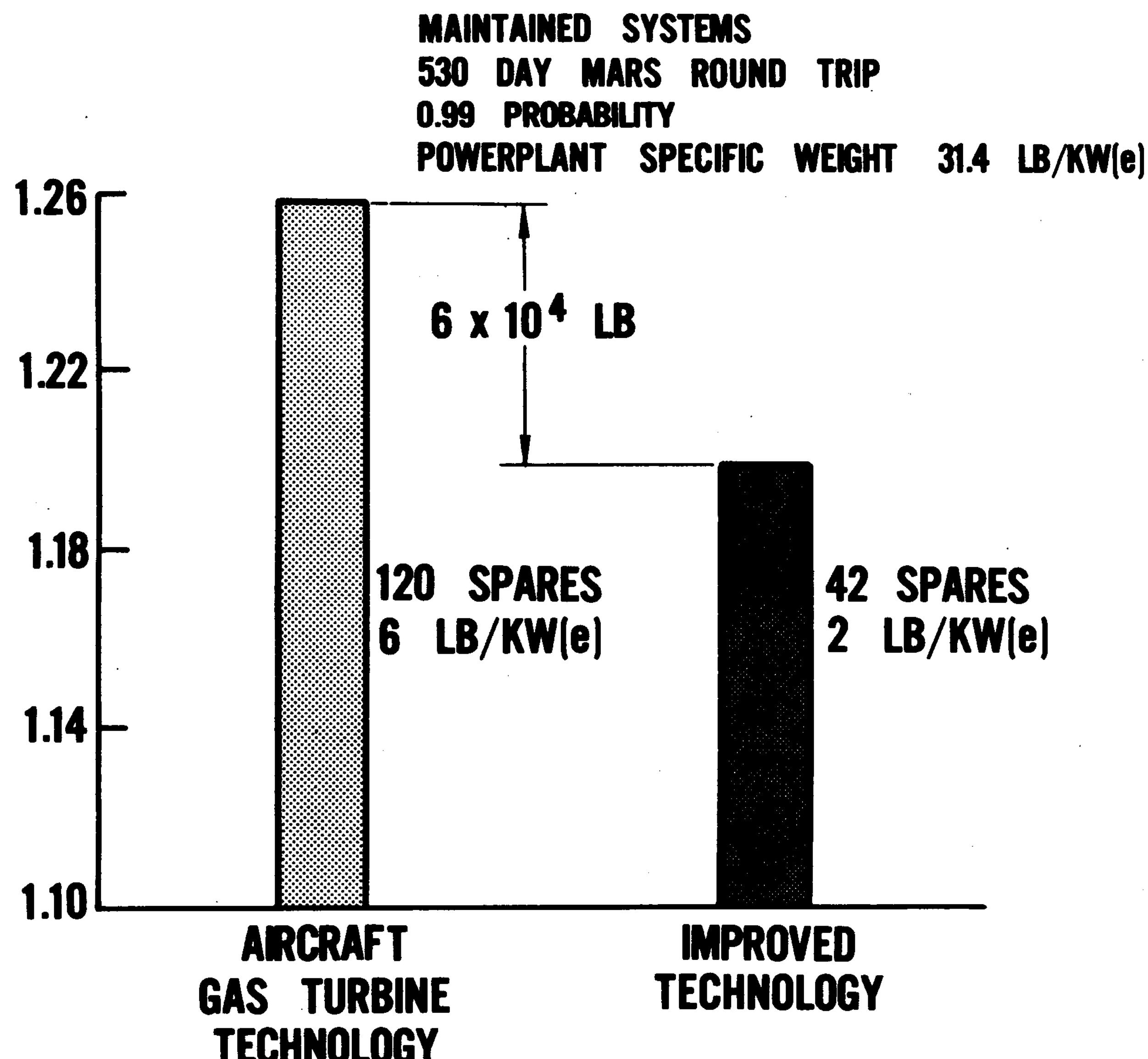
M-38903
662006

## LOW ACCELERATION SPACE TRANSPORATION
# EFFECT OF COMPONENT TECHNOLOGY ON MIEO

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M-38731
661006

Figure IV-7

# EFFECT OF TURBINE INLET TEMPERATURE ON VEHICLE REQUIRMENTS

## 530 - DAY MARS ROUNDTRIP

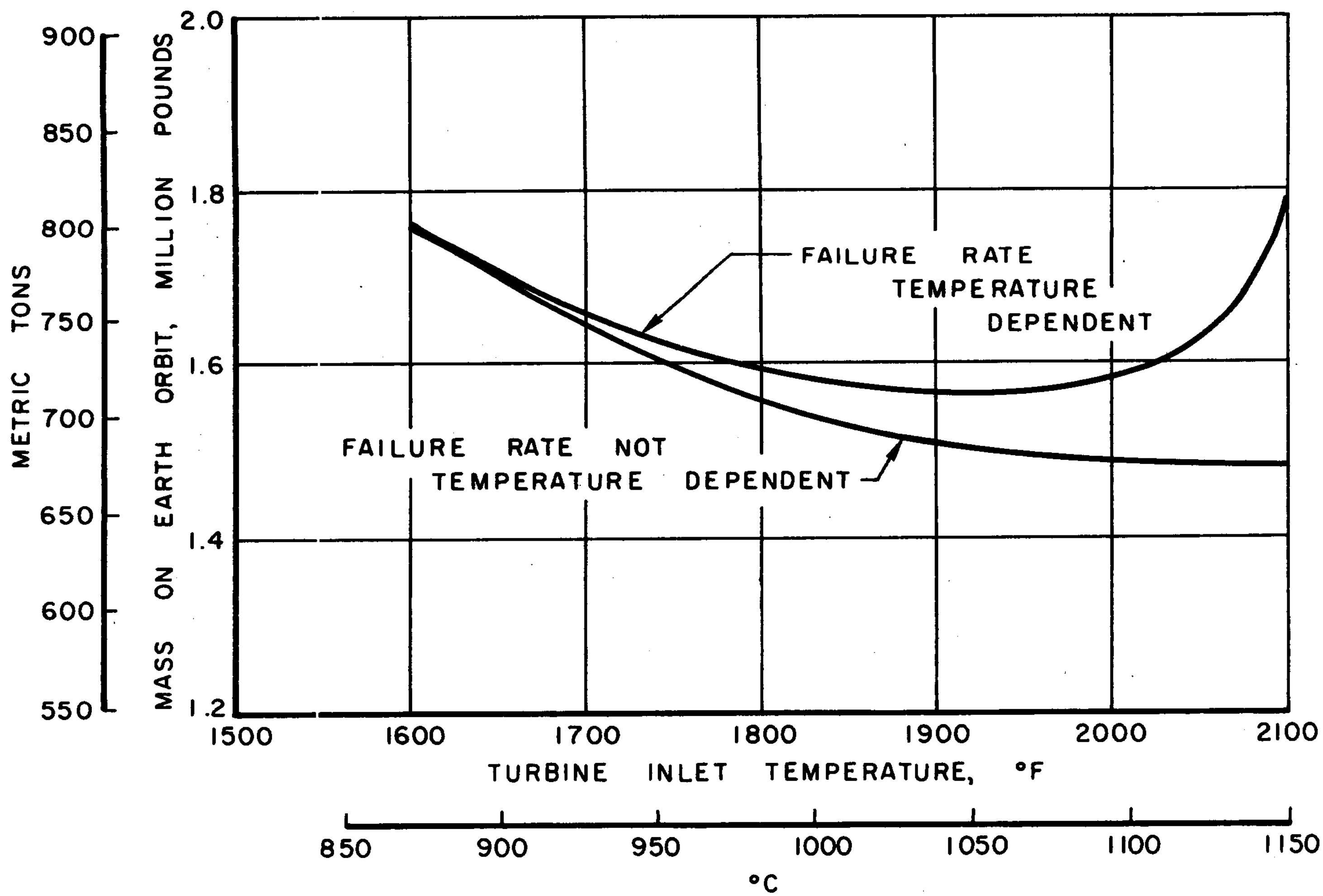
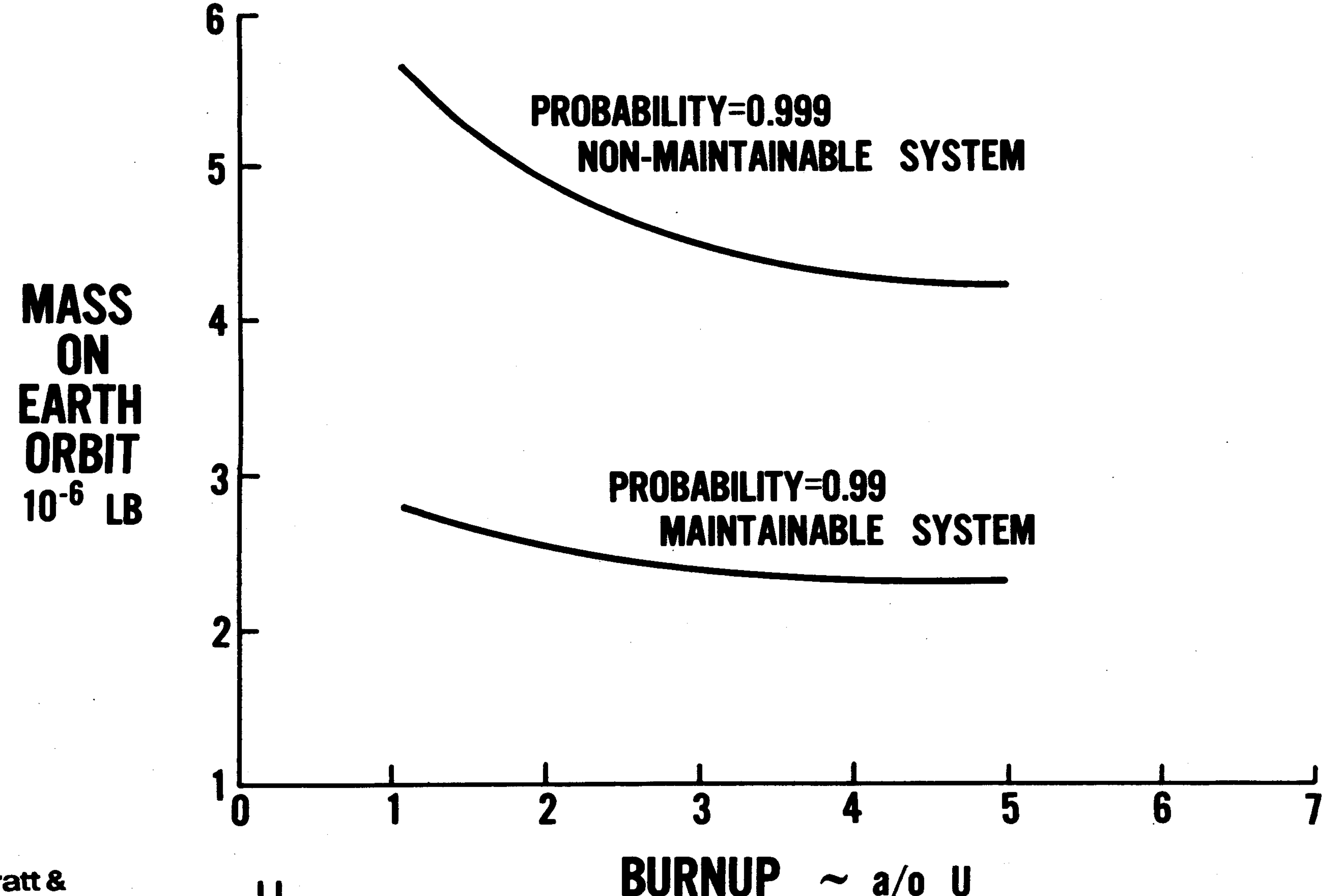



FIG IV - 8

Figure IV-9

LOW ACCELERATION SPACE TRANSPORTATION

# EFFECT OF BURNUP ON MIEO

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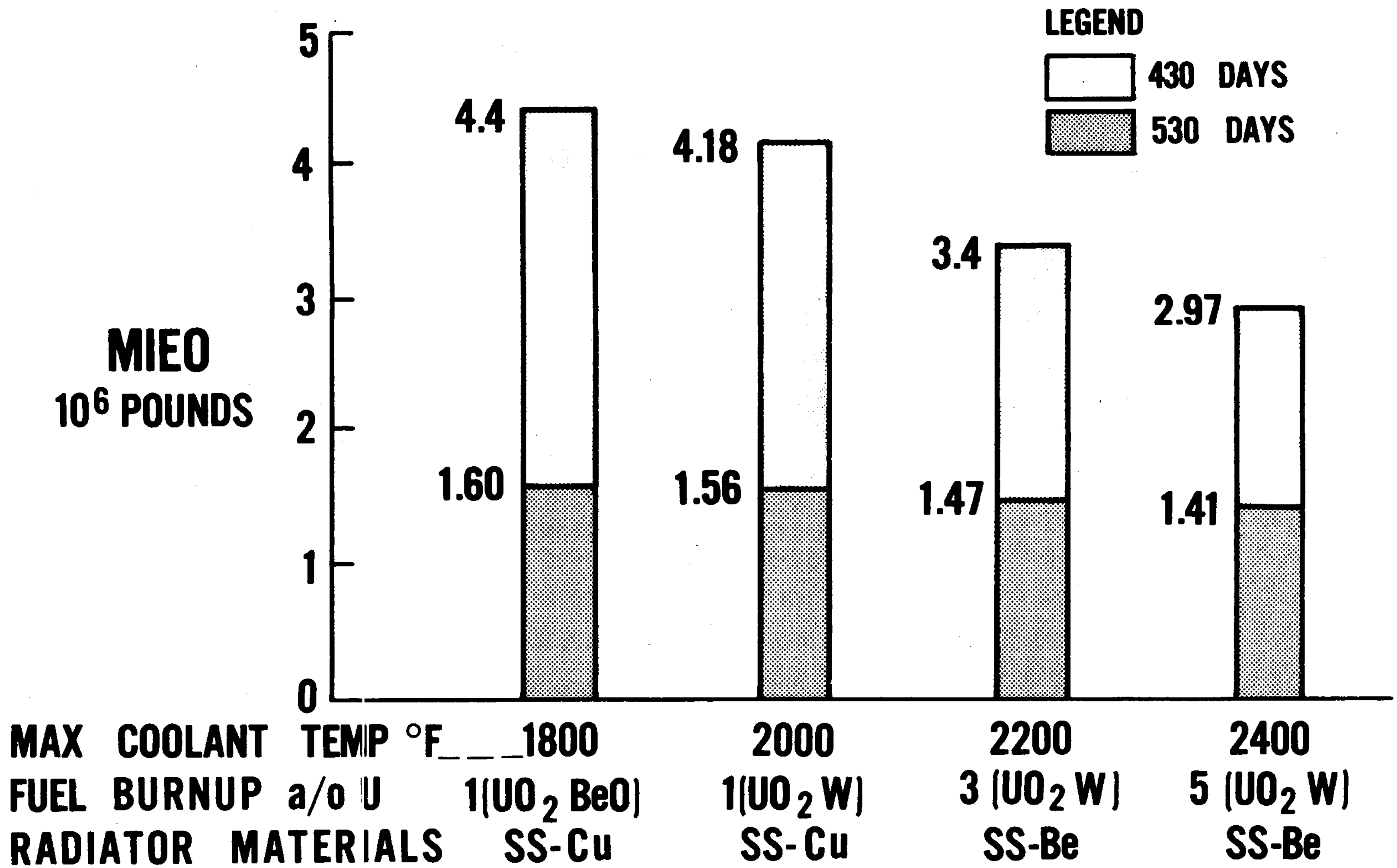
M-37513
661403

Figure IV-10

LOW ACCELERATION SPACE TRANSPORTATION

VARIATION IN MIEO WITH MAJOR POWERPLANT PARAMETERS

4 MW (e), 0.99 PROBABILITY, IMPROVED FAILURE RATES

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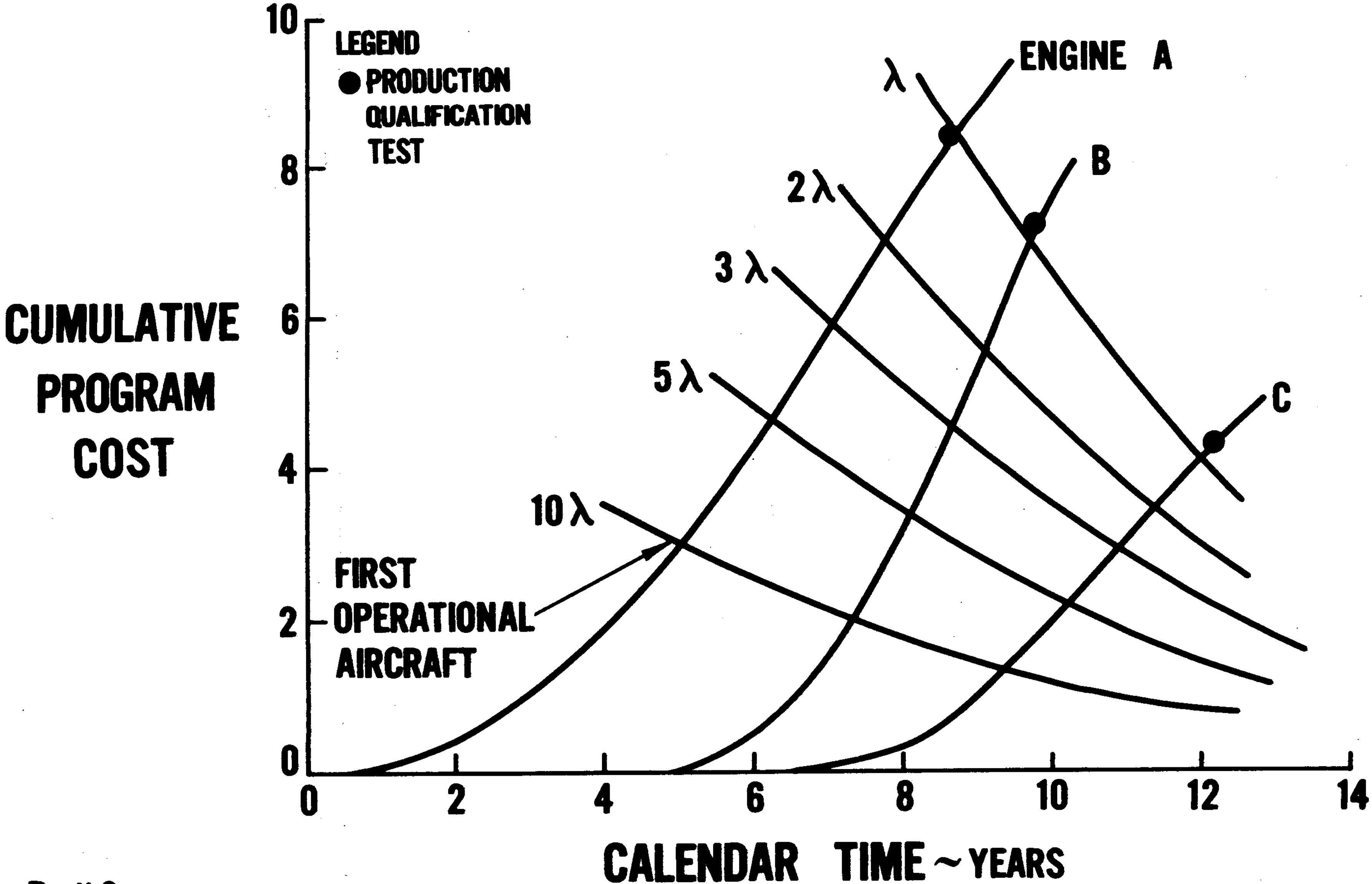
M-38936
66240E

# GAS TURBINE DEVELOPMENT PROGRAM CHARACTERISTICS

## BASED ON P&WA EXPERIENCE

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M-38902
662006

Figure IV-11

## LOW ACCELERATION SPACE TRANSPORTATION
# DEVELOPMENT PROGRAM SCHEDULE
### 1800°F 1 a/o BURNUP SYSTEM

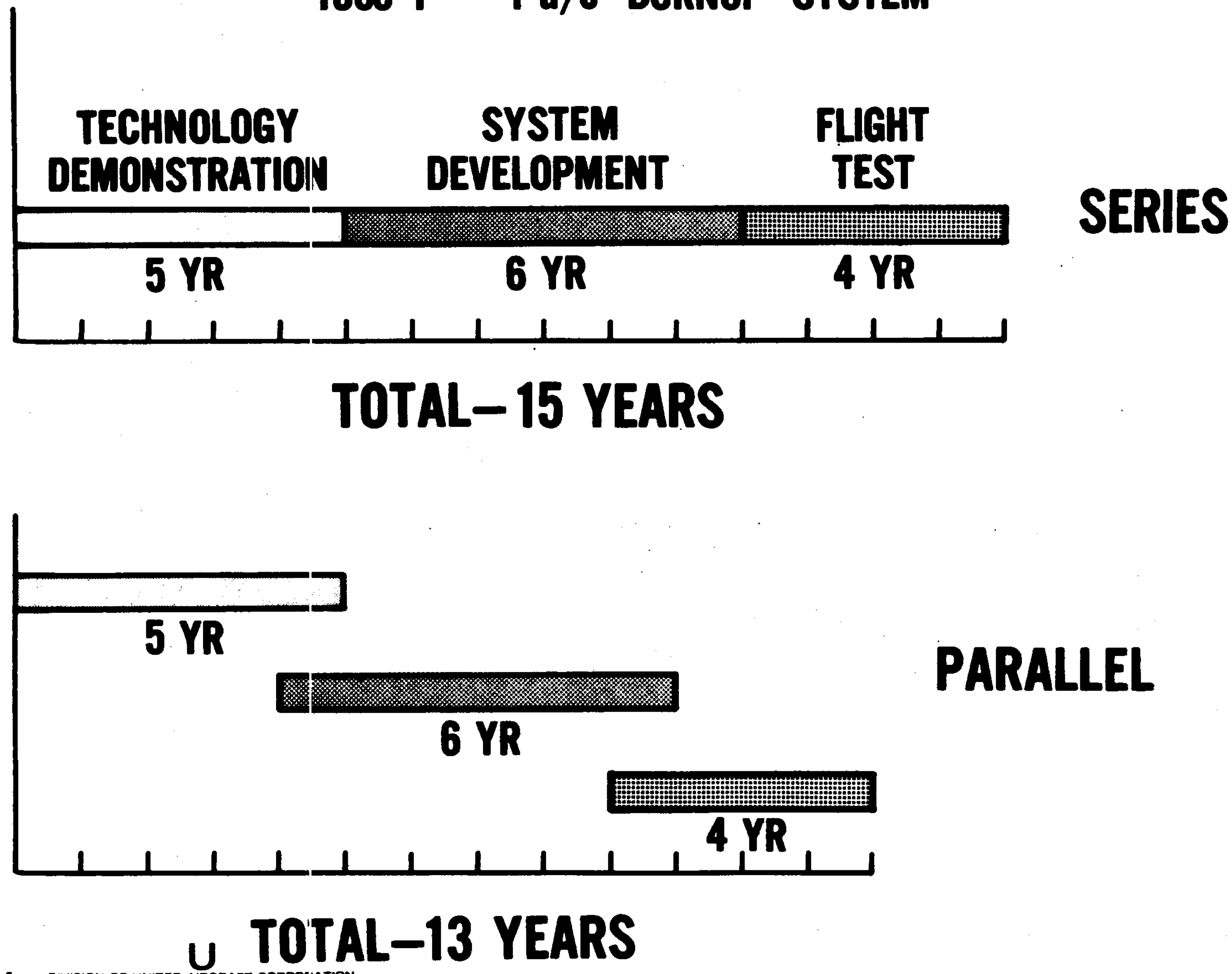




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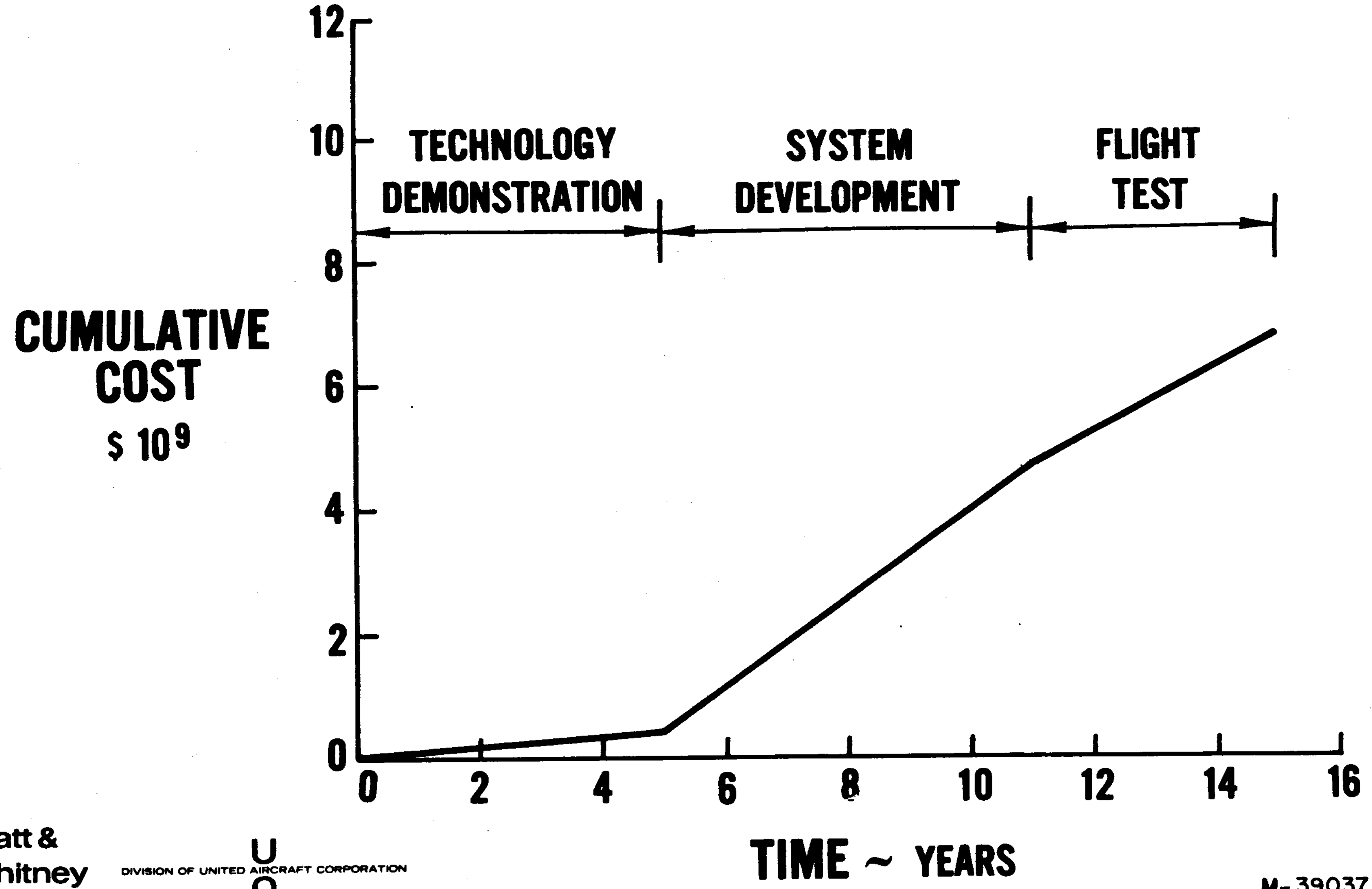
Figure IV-12

Figure IV-13

LOW ACCELERATION SPACE TRANSPORTATION

# DEVELOPMENT PROGRAM COST

1800°F 1 a/o BURNUP

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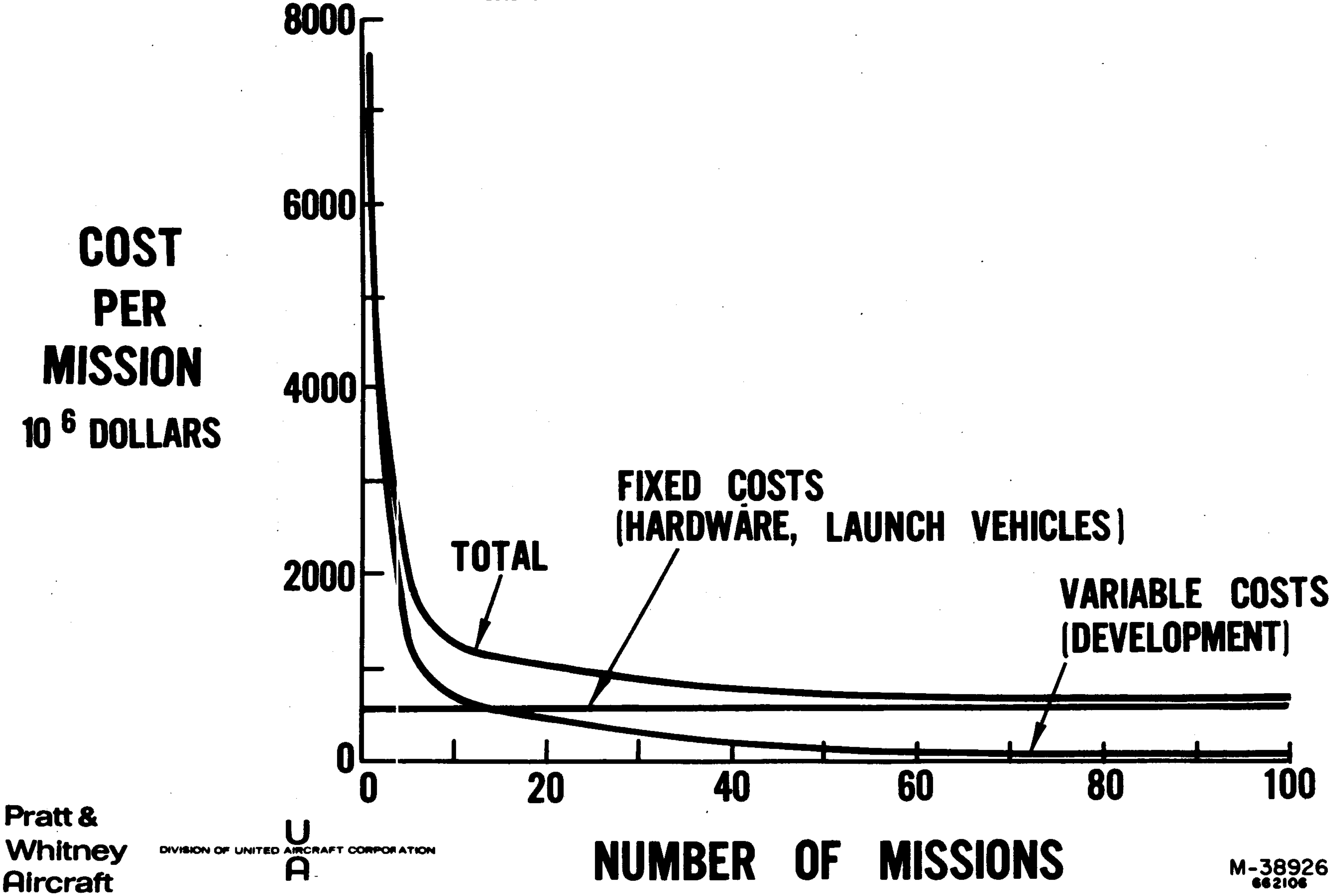
660107

LOW ACCELERATION SPACE TRANSPORTATION

# TRIP COSTS AS A FUNCTION OF NUMBER OF MISSIONS

MANNED MARS MISSION

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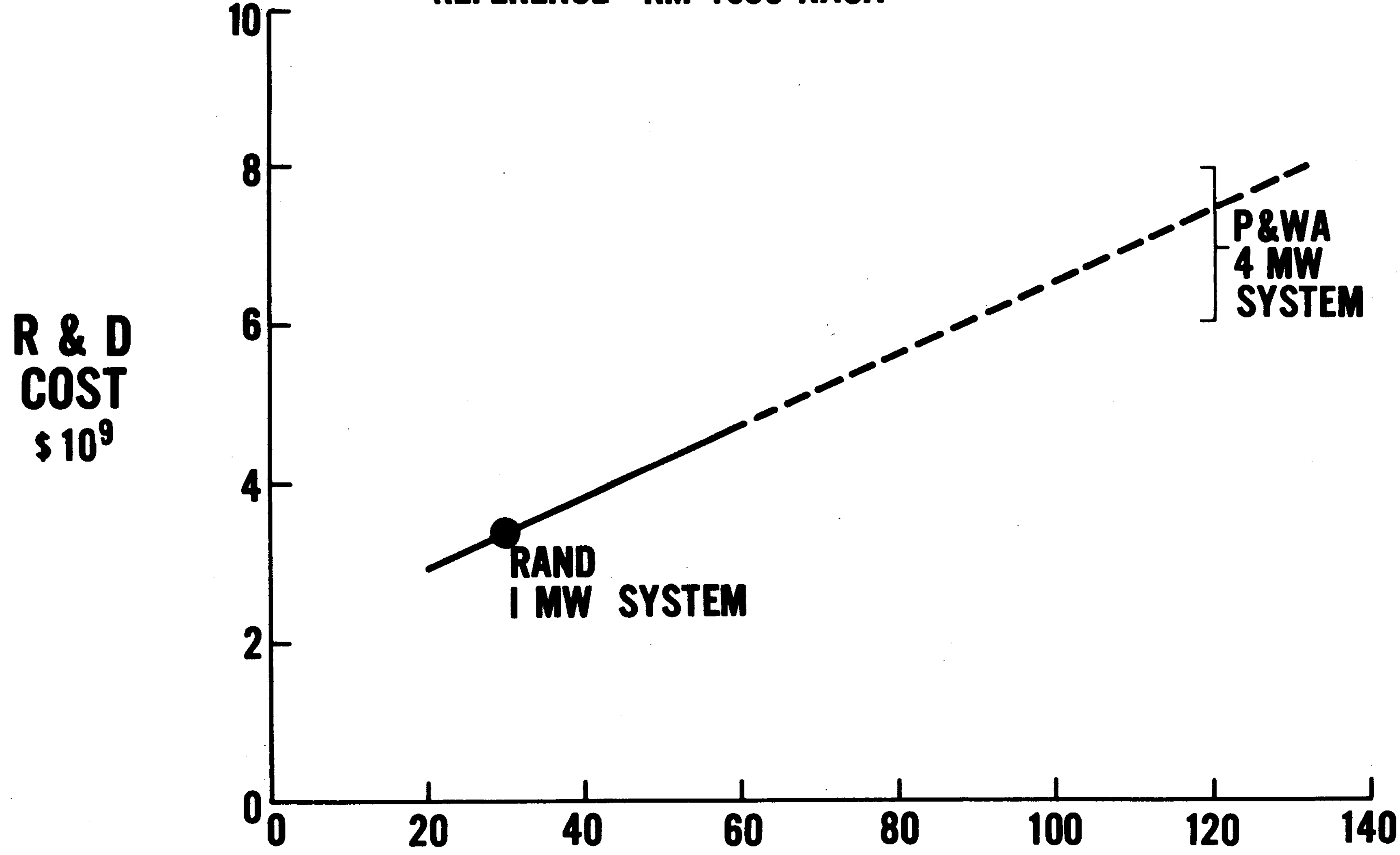
M-38926
662106

Figure IV-14

Figure IV-15

# INFLUENCE OF SYSTEM MASS ON R & D COST FOR SYSTEMS OF SIMILAR STATE OF TECHNOLOGY

REFERENCE RM-4056-NASA

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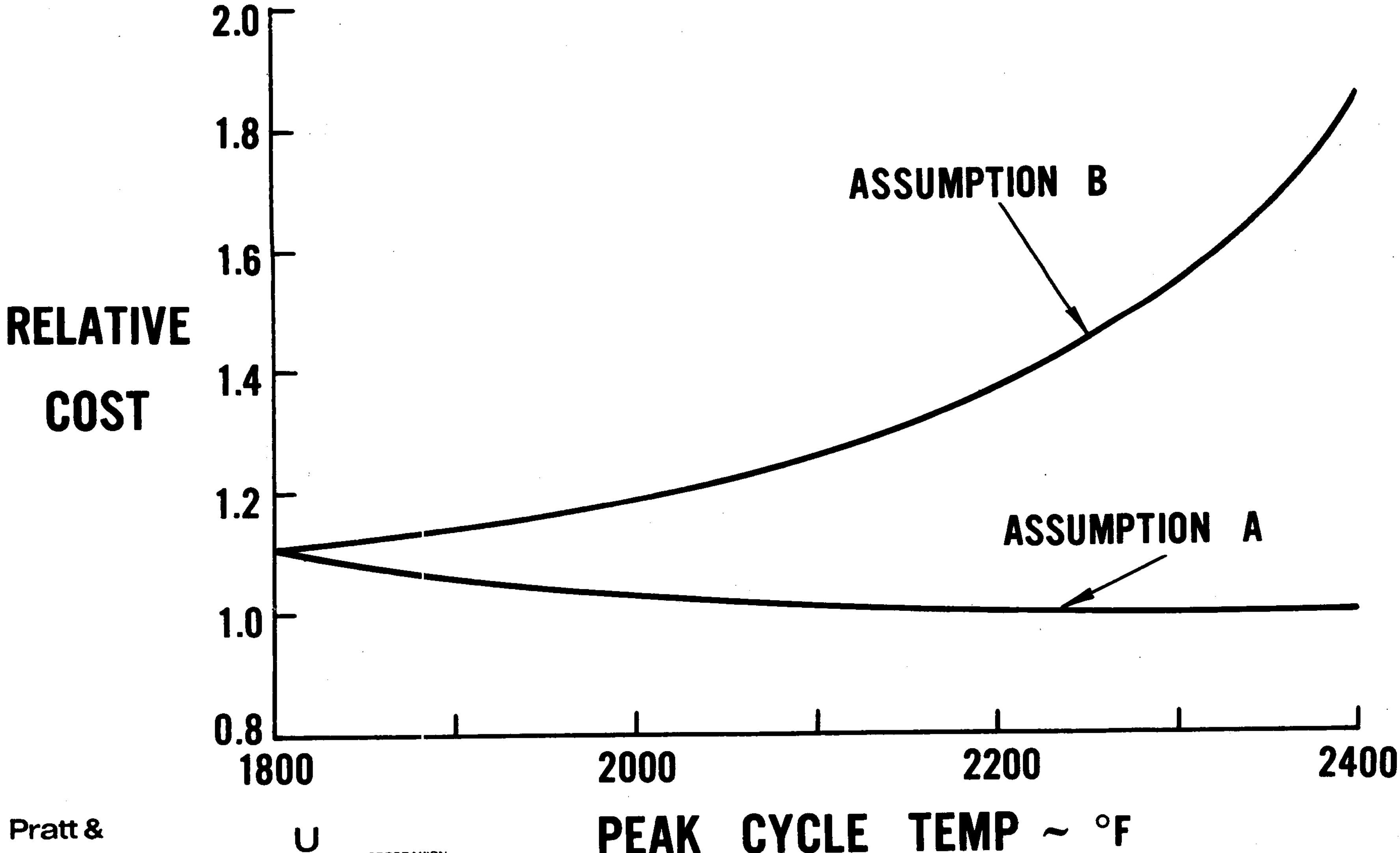
R660108

LOW ACCELERATION SPACE TRANSPORTATION
DEVELOPMENT COSTS
AS A FUNCTION OF PEAK CYCLE TEMPERATURE

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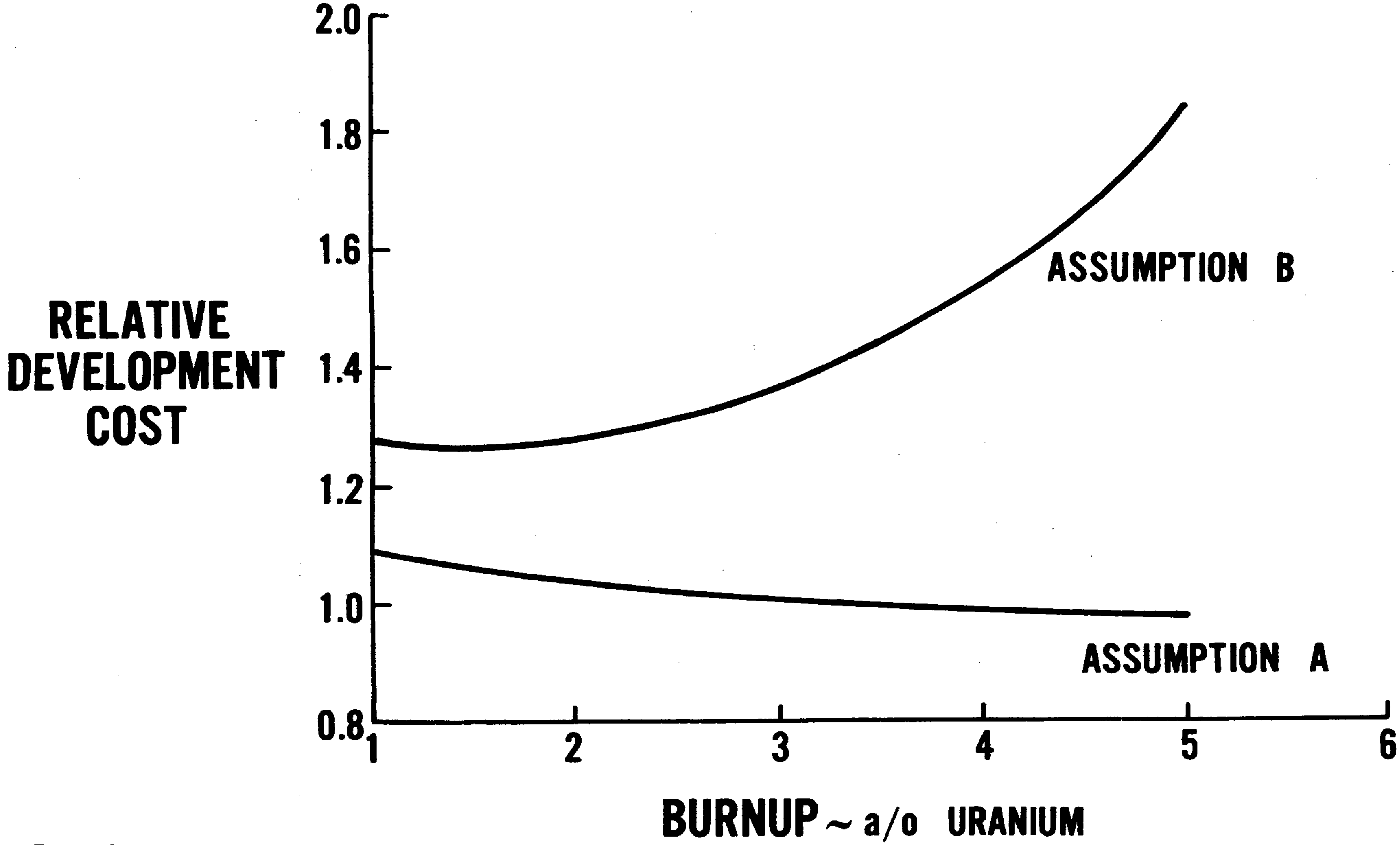
660107

Figure IV-16

LOW ACCELERATION SPACE TRANSPORTATION

# INFLUENCE OF BURNUP ON DEVELOPMENT COSTS

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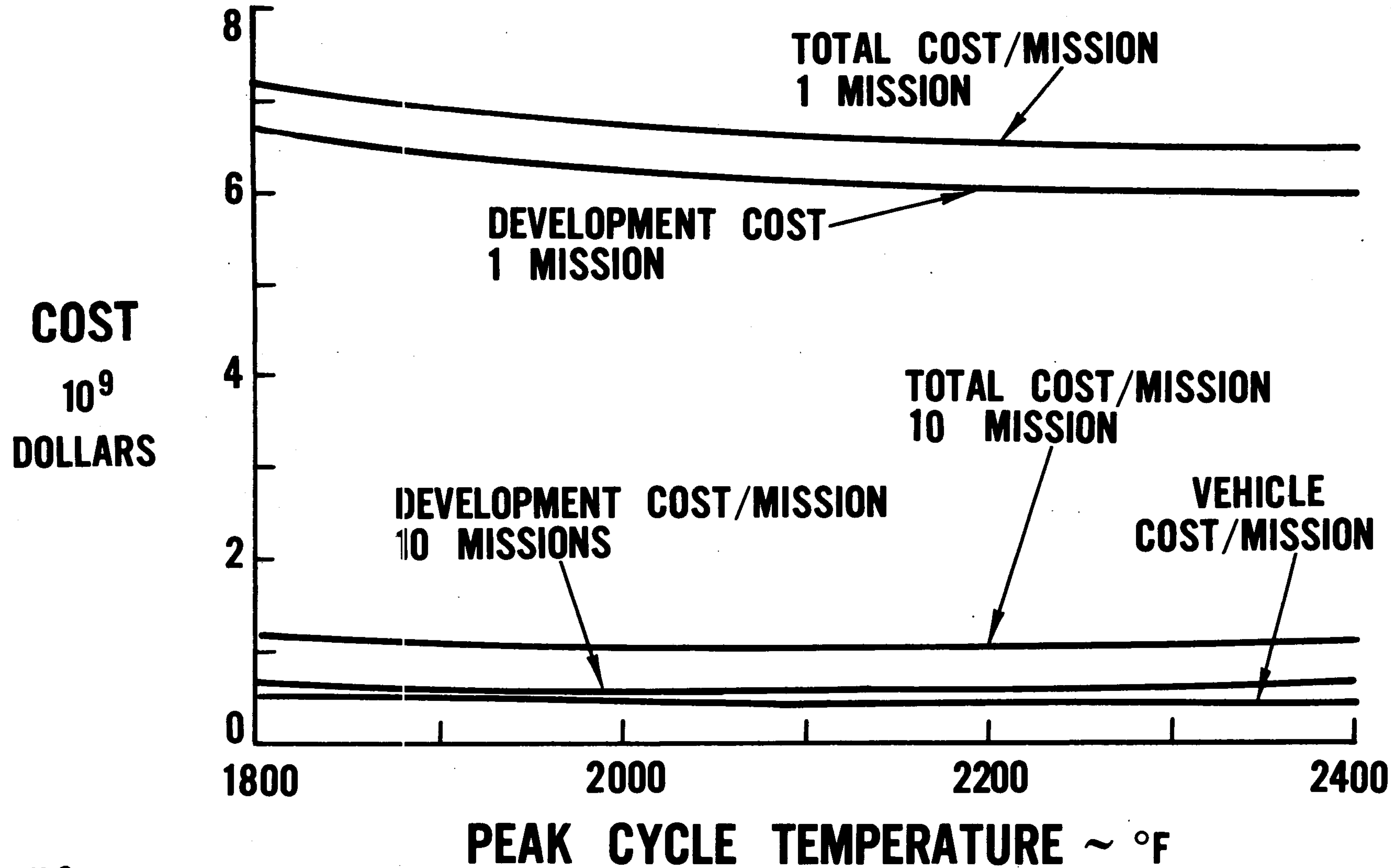
M-39008
663006

Figure IV-17

LOW ACCELERATION SPACE TRANSPORTATION

# EFFECT OF PEAK CYCLE TEMPERATURE ON MISSION COST

530 DAYS ASSUMPTION a 0.99 PROBABILITY

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663006

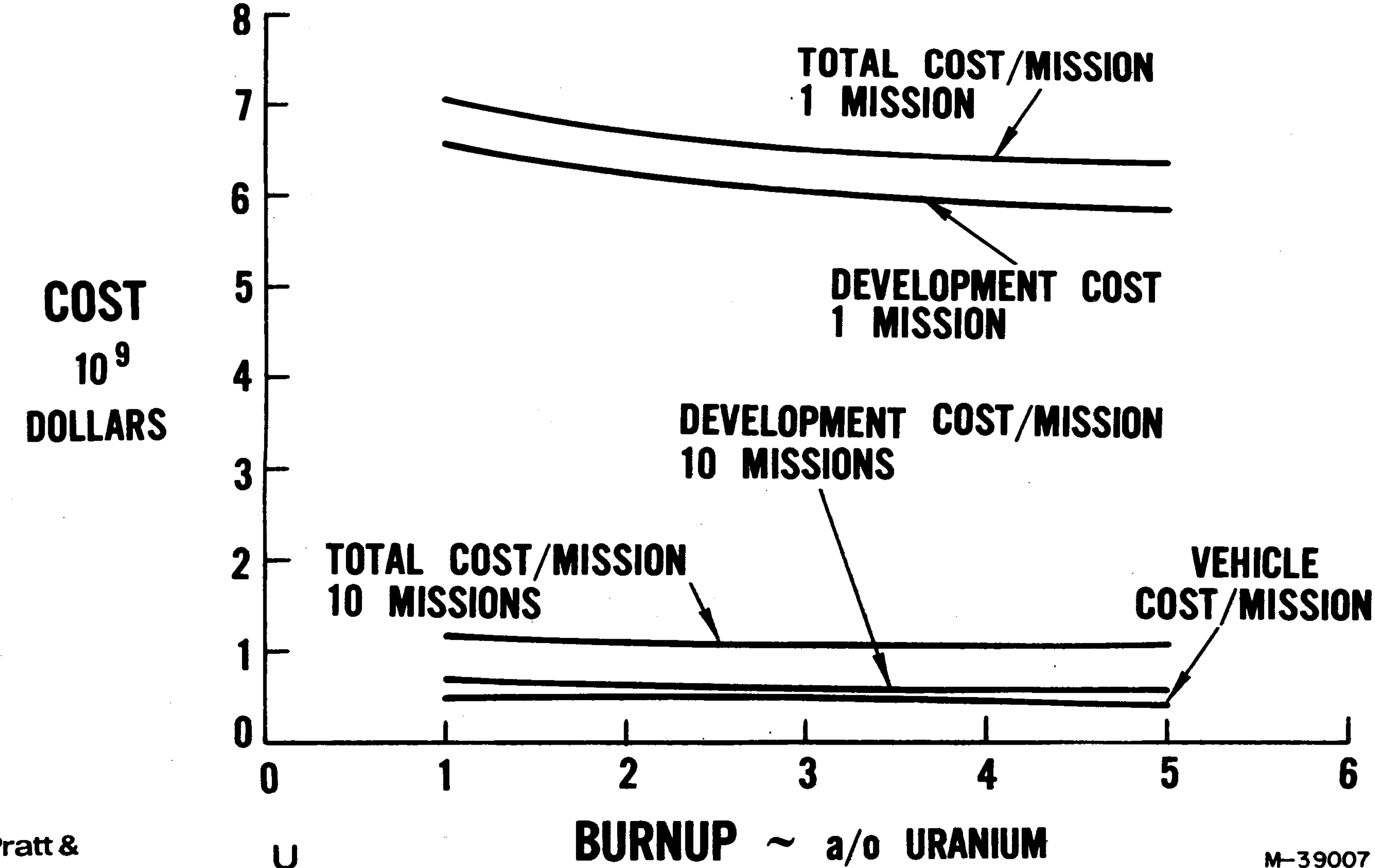
Figure IV-18

Figure IV-19

LOW ACCELERATION SPACE TRANSPORTATION

# EFFECT OF BURNUP ON MISSION COST

530 DAYS 0.99 PROBABILITY

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662906

# SECTION V
# ACCOMPLISHMENTS AND CONCLUSIONS

The major tasks which have been accomplished as a result of the study effort and the significant conclusions derived therefrom are summarized in this section. The implications to technological research activities wherein further efforts are desirable are listed in Section VI, Research and Technology Implications. Also listed in Section VI are study areas recommended for the future analysis of manned interplanetary hybrid-thrust missions.

## General Conclusions

1. A useful manned Mars mission employing a hybrid-thrust vehicle system can be performed in the 1980 time period. However, prompt initiation of the development of the nuclear electric powerplant and propulsion system is required due to the extended time required to develop this system. The "usefulness" of the mission is judged by the total vehicle mass required on Earth parking orbit (typically 600 to 800 metric tons) in relation to the payload delivered (MEM, 45 metric tons).

2. For extended trip times (about 500 to 600 days) the influence of powerplant specific weight on vehicle mass is reduced under hybrid-thrust operation so that high specific weights (15 to 20 kg/kw) are tolerable. This level of powerplant specific weight may be achieved by nuclear Rankine cycles with maximum cycle coolant temperatures in the range of 1800 to 2000 F and reactor fuel burnup in the range of 1 to 3 a/o uranium. Satisfactory powerplant reliability can be achieved if this type of system can achieve a state of development equivalent to that experienced by aircraft gas turbines and if an extensive inflight maintenance capability can be developed. It is estimated that the development program required for this type of powerplant will require about 13 to 17 years and 6 to 8 billion dollars. It is anticipated that any of the high-temperature nuclear powerplant systems currently proposed will require development programs of a similar magnitude.

3. In order to significantly decrease vehicle mass requirements, the powerplant specific weight must be less than 5 kg/kwe. However, it is not possible to identify a power system design which would result in a specific weight less than 12 kg/kw. Thus, there is little incentive for developing powerplants to achieve above-2000 F reactor outlet temperatures or 3 a/o reactor fuel burnup.

4. Under hybrid-thrust operations, the power requirements for the manned Mars mission are about 4 Mwe or less depending on the powerplant characteristics selected and the trip time. Reliability considerations

indicate that the favored powerplant for this mission should employ two reactors. From preliminary conceptual design analysis, the vehicle system integration problems are alleviated considerably provided the favored powerplant can be contained in one power module.

5. The available power, and consequently those characteristics which affect power, strongly influences the mass of the entire vehicle system. Major power system characteristics which determine the power availability are maintenance level, component failure rates, and probability level.

6. The mass requirements for extended-duration trips and high specific weights are essentially compatible with the Saturn V orbital payload capability. The actual number of Saturn V launches required to fulfill a given mission depends not only upon proper selection of the mass minimization techniques, but also on the level of sophistication in the design and packaging of the space transportation system for orbital assembly.

7. The hybrid-thrust system using a solid-core nuclear propulsion system in combination with electric propulsion is highly competitive with the vehicle systems employing highly advanced all high-thrust nuclear systems such as liquid-core and gaseous-core rockets. The major advantage of the advanced high-thrust nuclear system is the approximate 100 days less trip time required for the manned Mars mission.

8. Employing an Earth return rendezvous mode by using an Earth surface launched system to retrieve the crew at parabolic conditions results in a large mass penalty compared to an ablative atmospheric Earth capture system. The penalty in mass does not include the mass of the resulting Earth-based rendezvous system.

9. For optimum hybrid-thrust operation, the fractional hyperbolic excess speeds should be between 0.5 and 0.8 for Earth and Mars departure and 0.3 and 0.6 for Mars arrival, where the values represent the fraction of the impulsive (all high-thrust) transfer. For Earth arrival the values are about 1.0 depending upon the mass growth of the ablative entry system with atmospheric entry speed.

10. The over-all mission energy requirements are dependent on the orbital operations at the planet. The eccentricity of the parking orbit should be defined as a function of the retro $\Delta V$ accuracy, the number of monitoring passes the spacecraft makes during surface exploration, and the required total stopover time. The energy requirements for the orbital operations are a strong function of the percent of the hyperbolic excess speed which is applied to the high-thrust system.

11. The spacecraft design is strongly influenced by the type of orbit established about Mars and the type of propulsion used for the

retro-braking. Circular orbits tend to minimize the size of the MEM and maximize the Mars capture and departure propulsion requirements. As the eccentricity of the parking orbit is increased, the MEM weight increases while the capture and departure stage decrease in size. For highly elliptic parking orbits about Mars, the capture and departure energy requirements are so small that an $O_2/H_2$ chemical system can be used for planetary orbital operations and provide spacecraft weights which are lower than if solid-core nuclear propulsion were used. Also, the radiation problem is alleviated with a chemical system.

12. The favored trips which tend to reduce the mass of the vehicle arrive at Mars after the opposition date. Depending on the mission duration, the time after opposition ranges from 80 to 120 days.

## Major Study Accomplishments

The following list summarizes the major accomplishments of the study. The relevant section of the Technical Report (Vol. II) in which further detailed information may be obtained is indicated in parenthesis.

1. The finite-difference Newton-Raphson algorithm was successfully applied to solving the calculus of variations problem of variable low-thrust trajectory optimization. This approach resulted in a fundamental advance in the numerical solution of nonlinear two-point boundary-value problems and has yielded optimum variable-thrust trajectories by an order of magnitude faster than the most recent competitive methods. (Section V and Appendix A).

2. The basic algorithm and associated computer program have been extended to include hyperbolic excess speeds of planetary departure and arrival (mixed high- and low-thrust systems), variable power in the exhaust jet (Section V), planetary flybys and solar probes (Appendix B). Variable jet power includes variations in powerplant output due to sources such as radioisotope systems and solar cells, and component failures within the power system.

3. A numerical technique was developed for maximizing the payload of a vehicle operating under constant thrust, constant power with a single coast (Appendix B). A computer program was written which implements the optimization technique (Section V).

4. For variable-thrust operation, comparisons of optimum Earth-to-Mars trajectories were made for constant-power, radioisotope-power, and solar-power modes. Using rendezvous boundary conditions, contours of constant J were computed and organized for the Mars oppositions of 1978 and 1980 (Section V).

5. A technique was developed which maximizes (under ideal conditions) the payload-to-gross weight ratio of a combined high- and low-thrust vehicle system. This method and the implementing computer program is sufficiently flexible to include different propulsion parameters for the high-thrust and low-thrust systems, and also an atmospheric (rather than high-thrust) capture system (Section IV).

6. For hybrid-thrust systems employing rendezvous conditions at the boundaries of the low-thrust trajectories, the minimum-mass trips were found to be those arriving at Mars after the opposition date. A simple graphical approach is used in determining the optimum distribution of outbound and inbound leg times for a given mission duration and for establishing the arrival date which minimizes vehicle mass (Section IV).

7. The relative importance of powerplant specific weight, component technology level, onboard maintenance, and reliability level on the hybrid-thrust vehicle mass required on Earth parking orbit was determined for a range of the foregoing parameters and total trip times. The influence of the different powerplant characteristics was evaluated for their effectiveness in reducing the vehicle mass (Section III) and in establishing desired powerplant operating characteristics (Section VI).

8. The influence of the Mars parking orbit operational mode on the mass of the MEM and the parent spacecraft was established for highly elliptical (near-parabolic) and circular (926-km) parking orbits. The trade-off of parking orbit mode between the MEM and the spacecraft was identified in terms of system mass and favored spacecraft departure and arrival propulsion systems ($O_2/H_2$ or solid-core nuclear) (Sections III and IV).

9. A simple hybrid-thrust spacecraft conceptual design was established. Variations of the basic design layout were determined for elliptical or circular orbit operations and nuclear or chemical spacecraft propulsion. Problem areas relevant to the over-all optimization of the space system and the design integration of such a system were tentatively identified (Section III).

10. A series of flight profile and system considerations were delineated which were determined to be important approaches for minimizing the vehicle mass required on Earth parking orbit (Section III).

11. Power system output as a function of operating time (power profiles) was established for different subsystem and component redundancies, probability levels, and component failure rate levels. The influence and importance of maintenance level and powerplant technology on the power profile was determined for a two-reactor power system (Section VI).

12. Critical system design considerations and technology areas were identified which strongly influence powerplant specific weight (Section VI).

13. The time and cost for developing a 4 Mw maintained powerplant was evaluated (Section VI).

## Mass Minimization Considerations

1. For a given total trip time, the best combination of outbound and inbound leg times should be analyzed along with the arrival date.

2. The benefits in reduced mass requirements derived from operating under combined high and low thrust indicate that the hybrid-thrust mode should be employed in the more difficult (higher-energy) missions.

3. Ablative entry via an "advanced" type Apollo entry system yields less mass than arrival at Earth under parabolic conditions with retrieval by an Earth-based rendezvous vehicle.

4. Inflight maintenance strongly reduces the vehicle mass required, but it is an anticipated activity based on probability analyses.

5. Reduced probability levels tend to reduce mass requirements but also tend to reduce the probability of safe return of the crew.

6. The mission analysis should include the operational mode at the planetary parking orbit for its effect on the mass of the MEM and parent spacecraft.

7. In some instances the velocity requirements for effecting the desired planetary parking orbit are of such magnitude that no one propulsion system ($O_2/H_2$ or solid-core nuclear) can be considered a priori to have a distinct mass advantage; both systems should be investigated.

# SECTION VI
# RESEARCH AND TECHNOLOGY IMPLICATIONS

This section summarizes some of the important technical problems which must be solved in the development of a nuclear electric powerplant. Advanced research and technology programs are recommended which will contribute to the solution of these key problems.

Also presented herein are recommendations for future studies of a general nature which influence the employment of mixed high- and low-thrust systems in manned planetary missions.

## Major Power System Technical Problems

### Powerplant Startup

An important problem affecting the complete system is startup in a space environment. Included in the startup considerations are shutdown time prior to starting or restarting and the associated possibility of freezing various liquid metals, the initiation of stable boiling without significant liquid carryover, control of condensation and flooding in the turbine, adequate bearing fluid supply, stable condensing at low flow rates and startup thermal stresses.

The need for reliable valves in liquid-metal systems is generally recognized. However, this is an area in which relatively little technical effort has been expended. Control valves, shutoff valves, and check valves capable of extended operation in contact with high-temperature alkali metals will be required.

Another component important to the startup and operation of the Rankine-cycle space powerplant is an accumulator for service with high-temperature liquid metals in a space environment. The accumulator stores liquid metal for system makeup during starting and transient operation. It also must supply this liquid metal at nearly constant pressure to maintain the condensing temperature. An additional requirement for a maintainable liquid-metal system is a drain-and-fill system for the storage of liquid metals during some of the maintenance operations. One possibility is to combine the drain-and-fill and the accumulator functions into a single system.

### Reactor

The development of a lithium-cooled reactor capable of operation at high temperature is required. The magnitude of temperature and power reactivity coefficients remains to be demonstrated before a complete understanding of the reactor control requirements is established. Obviously, the operation and test of a power reactor of this type is required.

The development of a long-life high-temperature fuel presents an important development problem to be solved. The desirable properties for the fuel are low fission-gas release, high allowable burnup, and resistance to cracking and swelling.

## Shield

When LiH is incorporated into a shield design, its containment produces a major problem. If the LiH in the shield is allowed to reach temperatures around 800 F, it will dissociate to form gaseous hydrogen and liquid lithium. An overpressure of hydrogen can slow this dissociation, but hydrogen gas diffuses through most materials such as stainless steel, and will be lost during operation for extended periods at high temperatures. Also, the ability of the shield to stop neutrons will be decreased, and liquid lithium will be liberated with its attendant corrosion problem. If the LiH containment problem dictates the canning of LiH in small sections, the heat removal capability of the shield could be decreased, causing temperature problems. Heat generation rates in the shield are significant and a detailed examination of the shield design problem is warranted.

## Boiler

Boiling instabilities in a space environment represent a major problem to be solved.

## Turbine

### Materials

The peak temperature in the power-conversion system is determined by the capability of the turbine. Data are required which define the long-term strength and corrosion resistance of high strength-to-weight ratio materials capable of operating for a long period in a potassium vapor environment at temperatures close to 2000 F. Experience must be gained with casting, forging, machining, and welding of such materials in the forms required for the construction of a turbine. Materials must be developed which can guarantee the structural integrity of the turbine rotor. The effects of liquid potassium on blade life and turbine performance must be determined and methods of interstage moisture removal developed.

### Bearings

The hydrostatic bearings employed in the turbine require extensive development. Particular problems will be the startup of the turbine in a dry condition and the evaluation of liquid-metal flow rates required during startup and operation. The stability of radial bearings and the stiffness of these bearings needs to be investigated for the rotational speeds anticipated in this application. The possibility of bearing damage due to solid impurities in the liquid metal is relatively unknown and needs to be investigated in order to provide proper clearances and flow rates. The possibility of requiring filters to remove these particles should be investigated.

Seals

The use of dynamic seals in the turbine appears to be a feasible approach to provide for prevention of leakage of potassium liquid into the vapor section of the turbine. Extensive experience with this type of seal in liquid-metal pumps has been obtained at the P&WA CANEL facility. However, development work is required to produce a reliable seal with adequate cooling and startup characteristics. The problem of interface instabilities caused by changes in turbine rpm must also be investigated. This problem will be particularly important when the turbine is shut down for maintenance.

Condenser

The major problems for the condenser are the heat transfer and two-phase pressure drop correlations and condensing stability in a space environment. Further data on condensing heat transfer of liquid metal vapors are required.

Radiators

The main and high-temperature auxiliary radiators are constructed of stainless-steel tubes which contain the liquid metal. The tubes are covered with a beryllium barrier and have beryllium fins. A good metallurgical bond is required between the tube and barrier in order to assure good thermal contact. The bond between these materials must be strong enough to withstand the large thermal stresses imposed during fabrication and system operation. Materials are required with similar thermal expansion coefficients, which are chemically compatible and which are functionally suitable as tube and barrier.

The radiator surfaces are designed with a high-emissivity coating and such coatings have demonstrated high performance and extensive life in a high-vacuum environment. However, an examination of high emissivity coatings on beryllium is required to demonstrate the compatibility of the coating with beryllium.

Electrical

Electromagnetic materials which would permit operation at higher temperatures and higher stress levels could improve efficiency and reduce auxiliary cooling system weight. Switchgear which can be developed to operate in a high-temperature vacuum environment without welding or extensive contact resistance would be attractive in this application. Also, efficient electrical power-conditioning equipment which can operate above the present limiting temperature for semiconductors would allow system weight reductions. A major development item is the nonmagnetic and nonconducting bore seal required to separate the alternator rotor and stator. The seal will probably be made of a ceramic material.

System Maintenance

The results of this study indicate that an inflight liquid-metal system repair capability is required. Feasible methods must be developed for providing such a capability without excessive shield and equipment weight or powerplant shutdown penalties. Tools and equipment are required for cutting and welding of liquid-metal systems. Systems are required for detecting component failures, isolating components, and draining and filling liquid-metal systems.

## Recommended Power Systems Research and Technology Programs

Startup

The feasibility of space startup of the powerplant in various flight modes must be demonstrated. Turbine startup, including bearing supply, flooding and draining, and thermal transients are particularly important.

Reactor

Development of a high-temperature liquid-metal-cooled reactor is a requisite for this system. The key item in the reactor development is that of a reactor fuel which will tolerate the high-temperature long-lifetime environment, and which can tolerate the effects of radiation damage without failure. An additional item of importance is the development of a reliable reactor control concept with a sufficient reactivity effect for all mission modes. Control system development includes the development of a reactor vessel material capable of operating at high temperature with nuclear properties which do not adversely affect reactor control.

Radiation Shielding

The radiation shielding requirements are the single most significant consideration in determining powerplant weight. Extensive study is required in the determination of shield criteria. Development programs should cover the areas of shield materials and fabrication. This requirement is common to all types of nuclear power systems.

## Boiler and Condenser

The question of boiling and condensing stability is not yet well understood and further investigation of these phenomena is warranted.

## Turbine

Key items in turbine development include a high-temperature material with a high strength-to-weight ratio and good resistance to erosion by moisture in the vapor stream, the development of reliable turbine bearings and seals, and a method of extracting moisture between stages.

## Radiator

The fabrication procedure for the beryllium and stainless steel radiator requires development. The key element is a stainless steel tube with a finned beryllium barrier. A metallurgical bond is required between the stainless steel tube and the beryllium barrier in order to provide good heat transfer from the radiator.

## Valves

All power systems share a requirement for reliable valves. Development of valves for high-temperature liquid-metal systems will require an extensive program of materials development, design and fabrication techniques, and reliability demonstration.

## Electrical

The development of alternators and motors in the size required and suitable for operation in the required environment needs to be continued.

## Reliability and Maintenance

The requirements for powerplant reliability and maintenance are prime considerations for manned missions. Study programs are required to understand the interaction between reliability and maintenance. Development programs are required to demonstrate feasible and reliable maintenance techniques and equipment. A related development item is a reliable liquid-metal fill-and-drain system.

## Recommendations for General Future Studies

The following list presents the recommendations for further analysis based on the results and conclusions of this study.

1. Manned missions to planets other than Mars should be analyzed to clarify the mission spectrum which favors the use of combined high- and low-thrust space transportation systems.

2. Unmanned, automatic, planetary orbital and surface probe missions employing hybrid-thrust vehicles should be studied and compared against the system requirements of all high-thrust vehicles in order to establish further the role of mixed-thrust systems in planetary exploration.

3. Different mission modes employing constant- rather than variable-thrust trajectories should be analyzed to further check the validity of the conclusions derived herein.

4. Hybrid-thrust planetary missions employing swingby profiles should be investigated to determine possible mass savings and to uncover other types of propulsion-profile mixes which may prove advantageous mass-wise.

5. Solar cell and radioisotope power sources employing constant-thrust operating modes should be studied and compared with reactor power sources to establish the favored powerplant for various interplanetary missions and flight profiles.

6. Further analysis should be applied to identifying the influence of probable decreasing powerplant output with time (regardless of power source) on the vehicle mass requirements and to establishing the trade-off of power system reliability with specific weight.

7. For the same missions utilized in studying the Rankine cycle power system, powerplant comparisons should be made which include nuclear Brayton cycle and nuclear thermionic systems.

8. Investigations should be initiated into the effect that planetary parking orbit operations have on the over-all hybrid-thrust optimization.

9. An in-depth design study should be made of mixed-thrust vehicle systems in order to uncover operational and integration problems and to establish engineering feasibility of such spacecraft designs (especially in regard to the use of nuclear propulsion for planetary capture and departure and to the packaging requirements for orbital assembly).

10. Additional effort should be expended in analyzing the constant-thrust operating mode to determine the classes of trajectories which possess two rather than one coasting arc.

11. The basic Newton-Raphson algorithm should be applied to the constant-thrust-with-coast trajectory problem which has the payload optimization aspect as an integral part of the computational procedure.

12. The influence of abort at various phases of the manned planetary mission should be studied for its effect on the mission-flight profile selection and, consequently, on the vehicle system mass and design.

13. Analysis should be initiated into the problem of computing the low-thrust trajectory requirements under different abort modes and related remedial strategies.

14. The influence on the design of the heat rejection system when the powerplant is operating close to the sun (less than 0.5 AU) should be investigated. The consequent effect on flight profile selection and vehicle system mass should also be analyzed.