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by

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RESULTS OF SATELLITE RAISING AND ORBIT TRANSFER MISSION STUDIES FOR LOW POWER ELECTROTHERMAL ARC-JET PROPULSION SYSTEMS*

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ABSTRACT

Low power electrothermal arc-jet engines in the one to three kilowatt range are currently under development. Solar panel-battery power supplies are also under development at power levels up to several kilowatts, and can be made available at an early date for space vehicle applications. These engine-power supply combinations will have a great many potentially valuable space vehicle applications. Several applications have been investigated analytically, including satellite raising and orbit transfer, drag makeup, attitude control, station-keeping and trajectory control. This paper considers only the satellite raising (or orbit transfer) applications of such engines. Advantages and disadvantages of using solar panel-electrothermal propulsion for satellite raising are briefly discussed, and parametric mission study methods and results are reported for a specific mission of raising a satellite from an inclined parking orbit (AMR launch) to a synchronous equatorial orbit. Effects of independently varying the following parameters were investigated: propellant type (hydrogen and ammonia), engine power level, power supply specific weight, thruster specific impulse, and payload. Ascent time and parking orbit altitude were the dependent variables. Arc-jet engine design and development goals, in terms of propellant type, power level, and specific impulse, are optimized for the specified mission. Results indicate an optimum power level between 1.5 and 3 KW, based on projected power supply specific weights, and further indicate potential advantages of ammonia over hydrogen as the propellant, provided required engine life can be obtained. Principal potential advantage of ammonia over hydrogen is the much shorter ascent time required, while principal disadvantage involves the more severe thruster development problems. Criteria are suggested for selecting design and development goals for engine specific impulse.

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INTRODUCTION

The use of an electrically-propelled third stage on a two-stage chemical rocket booster can be very advantageous for certain classes of missions and payloads. Electrically-propelled upper stages have at least three essentially unique characteristics which make them significantly different from chemical upper stages, and one or more of these characteristics may be advantageous for any given mission. These characteristics include:

1. Higher specific impulses are provided than for any other space propulsion systems under development.
2. Relatively high levels of electrical power are required for propulsion which will be available for payload use without additional weight penalty, either intermittently in transit, or continuously after arrival in final orbit.
3. Propulsion will be available either continuously or intermittently, as needed, for durations of many months at very low thrust levels.

Potential applications of electrical propulsion are numerous. Some fall into the category of primary space propulsion for orbit transfer, while others are in the category of auxiliary space propulsion for such functions as attitude control, station-keeping, mid-course trajectory control, drag makeup, and others. In this paper the discussion will be limited entirely to (1) certain applications in the primary propulsion category, (2) the use of advanced types of solar panel-battery power supplies in the 1-10 KW power range, and (3) the use of electrothermal arc-jet engines in the same power range.

There are two major potential uses for this kind of primary propulsion capability. One is to allow the placing of a heavier payload, with a larger power supply, into a final high orbit, compared with the payload and power supply which can be placed in the same final orbit using the same booster and a chemical upper stage. The second potential use for the primary propulsion capability involves the fact that a vehicle propelled by an arc-jet propulsion system ascends from initial to final orbit slowly in a tight spiral, during which scientific data on the near-earth space environment can be gathered and relayed in far greater detail than is feasible with chemically boosted space vehicles. In this case it is the ascent itself which is of interest. The reasonable ascent time required by the electrothermal propulsion system probably represents no penalty, but is an advantage. The data to be collected in such missions might include the detailed mapping of the various natural and artificial radiation belts over a period of time, detailed mapping of the magnetic field of the earth, and the collection of micrometeoroid impact data over a period of time at a given altitude.

Parametric mission studies have been carried out by Plasmadyne for these types of missions under a NASA applications study contract which is intended to provide guidance in the formulation of design and development objectives for low power electrothermal arc-jet engines. For any specified type of mission within the capabilities of electrothermal propulsion, these studies allow the determination of "optimum" power level, specific impulse, and initial parking orbit altitude, for either hydrogen or ammonia as a propellant. "Optimum" is defined as the engine operating point resulting in minimum ascent time consistent with maximum probability of mission completion. Results can be used to guide future engine, power supply, and propellant storage developments.

Some of the methods and results of such a mission study are presented in this paper for one of the specific missions investigated, namely, the boosting of a moderate-size satellite into a synchronous equatorial orbit from an AMR launch into an inclined parking orbit.

DISCUSSION

It is desired to analyze and optimize missions, such as the orbit transfer mission of raising a satellite from a low inclined orbit to a synchronous equatorial orbit, using input data which are as accurate as possible, with a minimum of simplifying assumptions and approximations. However, input data in several categories are not completely and accurately known, and therefore simplifying assumptions and estimations become necessary if any useful results are to be achieved from mission analyses. Two such input approximations are shown in Figures 1 and 2.

Figure 1 shows estimated performance characteristics for low power hydrogen and ammonia arc-jet thrusters. In general, the efficiency of a thruster (ratio of directed kinetic energy (thrust) to input electrical energy) is a function of propellant type, design configuration, specific impulse, power level, and endurance time on thruster. If an optimized design configuration is assumed for each propellant, that factor can be eliminated as a variable. Knowledge of the variation of thruster performance with power level and endurance time is not adequate at the present time for accurate quantitative representation, and therefore it was necessary to neglect such variations in the analysis. Actually, efficiency will tend to increase slightly with increasing power level, and decrease somewhat with increasing endurance time, but probably not very greatly for the mission times under consideration if the thruster design is to be a successful one. Elimination of the variation of efficiency with power level and endurance time in the analysis leaves propellant type and operating specific impulse as the primary variables. The estimated effects of these variables are shown in Figure 1 for successful designs. A power adapter efficiency of 95% was estimated for the study. The overall engine efficiency is then 95% of the values shown in Figure 1.

The assumed capabilities of an intermediate size booster are shown in Figure 2 for a circular orbit at an inclination of 28.5 degrees, the minimum inclination possible for a launch at the Atlantic Missile Range (Cape Canaveral). The capabilities of a typical two-stage launch vehicle, in terms of parking orbit payload weight vs. altitude, were estimated from information given in Reference 1 for the Atlas-Agena B, and are therefore indicative of current or near future capability for boosting into circular parking orbits.

The characteristic velocity (time integral of applied acceleration) required to cause the vehicle to spiral out to the synchronous altitude with low thrust can then be approximated by (Ref. 2):

$$V_{ch} = \sqrt{\frac{\mu}{r_1}} - \sqrt{\frac{\mu}{r_2}} \tag{1}$$

where r_1 and r_2 are the initial and final radial distances respectively and μ is the universal gravitational constant. If the orbital inclination is to be changed, as is the case here, the characteristic velocity increases by $1/\cos\varphi$, where φ is the constant out-of-plane thrust deflection angle required to produce the desired 28.5 degree inclination change between r_1 and r_2 (Ref. 2). The characteristic velocity requirement for the low thrust portion of the mission is shown in Figure 3. Employment of an out-of-plane thrust deflection angle which increases with altitude would result in a slightly more economical trajectory at the possible cost of some additional control complication.

The vehicle characteristics which have been assumed for the study include structural weights, propellant storage and feed system weights, and the fixed weights.

The structural weight of the satellite was assumed to be a constant 5% of its initial weight in the parking orbit, which is consistent with results published by other vehicle and mission investigators (Ref. 3 and 4). Studies of propellant storage and feed systems which have been made indicate that ammonia should be stored in a relatively simple tank system as a dense liquid at approximately the equilibrium temperature of the vehicle. Therefore an ammonia propellant tank can be comparatively small and light, with the weight a constant fraction of the propellant weight, independent of mission time. The ratio of propellant storage and feed system dry weight to total propellant weight, a', has been taken as 0.20 for the ammonia systems.

For the hydrogen systems the situation is not quite so simple. Hydrogen must be stored as a cryogenic fluid, either subcritically or supercritically, at very low temperatures. The density of the cryogenic fluid is low, requiring sizable tanks. There are at least two realistic methods for storing hydrogen for steady use over a long period of time. One involves the extensive use of multi-layer reflective insulation, vapor-cooled shields, and special supports. The system weight ratio, a', is a function of both the absolute quantity stored and the mission duration. The other method involves the use of an on-board mechanical refrigeration system, for which the system weight ratio, a', is primarily a function of the absolute quantity only, and is independent of mission duration. Substantial progress has been made in both types of storage systems, and recent studies indicate that both would be competitive weight-wise for the missions under study. For this analysis the mechanical refrigeration type of system was selected because of its independence of mission duration. The results of a weight study were approximated by the following empirical expression:

$$a' = 1.18 + 0.374 \exp\left(\frac{-w_p}{346}\right) \tag{2}$$

where w_p is total propellant weight, lbs.

For a typical mission the estimated ratio of storage and feed system dry weight to total hydrogen propellant weight, a', is approximately 1.25 or 1.30. The most recent work in the design of insulated (non-refrigerated) storage systems indicates that a significantly lower ratio is probably achievable for this kind of mission, but this information was not available at the time the mission calculations were made.

The vehicle fixed weights, which include guidance and control equipment, electronics, and the electrothermal engine group, was estimated to be 150 lbs.

The rocket equation, and the basic relationships between electrical power, thrust, and propellant flow rate are as follows:

$$V_{ch} = I_{sp}\, g_o \ln \frac{1}{1 - \frac{w_p}{w_o}} \tag{3}$$

$$F = \frac{45.9\, P\eta}{I_{sp}} \tag{4}$$

$$\dot{w}_p = \frac{F}{I_{sp}} \tag{5}$$

$$t = \frac{w_p}{\dot{w}_p} = \frac{w_o - w_f}{\dot{w}_p} \tag{6}$$

where:

V_{ch} = mission characteristic velocity, ft/sec.

I_{sp} = specific impulse, sec.

w_p = total propellant weight, lbs.

$\dot{w}_p$ = propellant flow rate, lbs/sec.

w_o = initial vehicle weight, lbs.

w_f = final vehicle weight, lbs.

F = engine thrust, lbs.

P = engine input power, KW

η = engine efficiency

t = thrusting time, secs.

These four equations, together with the input data described above and the relationships shown in Figures 1 through 3, were employed in mission calculations. Part of the results of these calculations are shown plotted in Figure 4. Figure 4 is typical of the plots which have been made for different power levels, with hydrogen or ammonia as a propellant, and with initially circular and elliptical orbits. Only circular parking orbit studies will be presented in this paper. Note that it is unnecessary to present a set of curves for each power supply specific weight since different power supply specific weights can be estimated merely by reading from different payload lines. The specified payload is to be exclusive of the power supply weight. From Figure 4 it is seen that there is both an optimum I_{sp} for minimum ascent time and a minimum required I_{sp} for a given payload. These conditions occur at different altitudes for different payloads, with different times required for ascent to orbit. Thus a study to optimize a mission of the type assumed should consider the complete spectrum of booster capabilities instead of being limited to one particular parking orbit altitude.

Further analysis shows that if the minimum time conditions for different power levels are plotted for one payload, the optimum power level and specific impulse can be identified. The results of such a plot are shown in Figure 5 for 100 pounds payload. Note that these curves are not envelopes of the payload curves shown in Figure 4, but only show the minimum time of transfer for this particular payload at the indicated power levels and power supply specific weights. The required power level to minimize time can be obtained from Figure 5.

Performance charts similar to the one shown in Figure 5 have been developed for other payloads. These data have then been used to construct a plot of minimum ascent time as a function of payload, power supply specific weight, and propellant type, as shown in Figure 6. Each point along the curves of Figure 6 corresponds to an optimum combination of power level, specific impulse, and circular parking orbit altitude which permits the minimum ascent time for the particular values of payload, power supply specific weight, and propellant of interest. Note that the optimum combination of power level, specific impulse, and parking orbit altitude varies continuously along each of the curves of Figure 6, and that the values for any specific point can be determined by referring back to preceding plots.

The same set of curves required to produce Figure 6 can be used to determine the optimum I_{sp} required to perform the mission. These results are shown in Figure 7. In the low payload weight ranges the optimum I_{sp} is independent of the power supply specific weight, although the minimum ascent time is strongly affected by power supply specific weight. The power supply weights are not included as a part of the payload, and therefore care should be taken in the use of these mission study results since at least some power must be supplied to the payload in almost every conceivable mission, and the weight of that portion of the power supply should be considered as additional useful payload.

From Figure 5 it can be seen that there is an optimum power level associated with each power supply specific weight. Since detailed calculations have shown that the optimum power level is relatively independent of the payload and ascent time, the optimum power level can be plotted solely as a function of power supply specific weight and propellant type, as shown in Figure 8.

The engine design and mission optimization procedures described above, and the results presented, have necessarily been based on certain simplifying assumptions and input data approximations, as described earlier in the paper. These assumptions and approximations have been necessary because of the absence of sufficiently complete information in certain categories. One of the simplifying assumptions of this study is the independence of efficiency with power level. This is not really the case, but not enough development testing has been reported to include any such results in this study at this time. Another critical unknown is the actual life of the arc-jet engines. It is known that the life is a function of I_{sp}, and to some extent power, but no life test results of the lifetime assumed have been reported. The mission studies methods and graphs presented in this paper offer opportunity for including such information when it is available in order to determine the final optimum operating conditions. For example, Figure 9 shows what a typical engine life curve (90% probability) might look like as a function of specific impulse. Also shown on the same chart is a typical plot of ascent time as a function of specific impulse for a payload and mission of interest. This plot suggests that minimizing the ascent time may not be the optimum criterion for selecting operating specific impulse because the probability of successfully completing the mission may be greater at a somewhat lower value of specific impulse. This line of reasoning applies primarily to the case where the number of engines which can be used for the mission must be limited to one or a specific small number of engines employed sequentially. If there is no rigid limit (other than sound system design) on the number of engines which can be employed sequentially, then perhaps the engines should be operated at their optimum specific impulse and their number increased slightly to compensate for the shorter individual lifetimes.

CONCLUSIONS

A method has been developed and presented for the graphical optimization of earth-orbit satellite raising missions for satellites employing low power arc-jet propulsion and solar panel-battery power supplies. The method has been presented in some detail for the specific case of raising a satellite to a synchronous equatorial orbit. Using a conventional launch vehicle to boost the satellite into an initial non-equatorial orbit, the arc-jet propels the vehicle in a continuously turning spiral to the final radius in the equatorial plane. Parameters varied in the study include thruster specific impulse, efficiency, power level, and propellant type (hydrogen and ammonia), booster parking orbit altitude vs. injected weight, final payload weight, and power supply specific weight. The output of the study is information on the optimum specific impulse, power level, and parking orbit altitude for any specified payload, booster characteristic, power supply specific weight, and propellant.

For example, for the Atlas-Agena B class of boosters, and for a power supply specific weight of 150 lbs/KW, the optimum power level is approximately 2.3 KW -- independent of propellant type at this specific weight. For a payload of 100 lbs. (or 445 lbs. including the 2.3 KW power supply), an ammonia arc-jet should operate at a specific impulse of approximately 570 secs., while a hydrogen arc-jet should operate at a specific impulse of approximately 1250 secs. Ascent time would be about 70 days for ammonia and 220 days for hydrogen.

More generally, the results of the study indicate that the optimum power level for this class of engines is between 1.5 and 3 KW, based on projected power supply specific weights, and further indicate potential advantages of ammonia over hydrogen as the propellant, provided that the required engine life can be obtained on ammonia. Principal potential advantages of ammonia over hydrogen include the much shorter ascent time required and the more practical design and size of the propellant storage system, while the principal disadvantage involves the more severe thruster development problems.

Finally, a procedure has been outlined for selecting design and development goals for arc-jet thrusters, including a procedure for taking into account the thruster life vs. specific impulse characteristics. This information should be included in the design point selection process when adequate experimental data is available.

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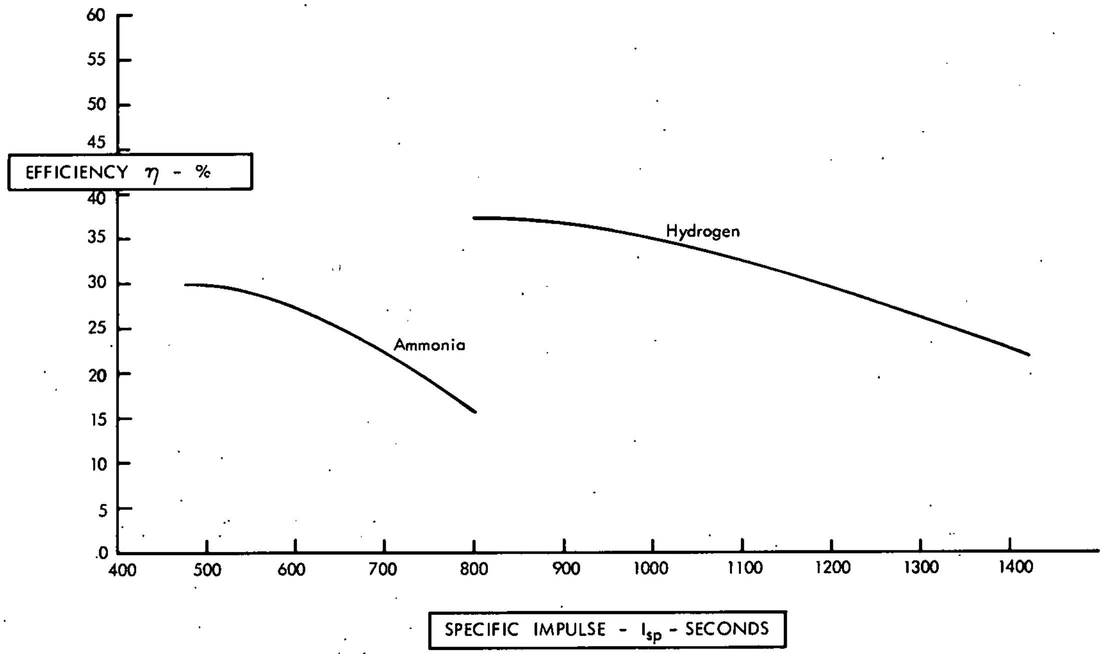

Figure 1 ASSUMED EFFICIENCIES FOR LOW POWER HYDROGEN AND AMMONIA ARC-JET THRUSTERS
Electrical Conversion Efficiency = 95%

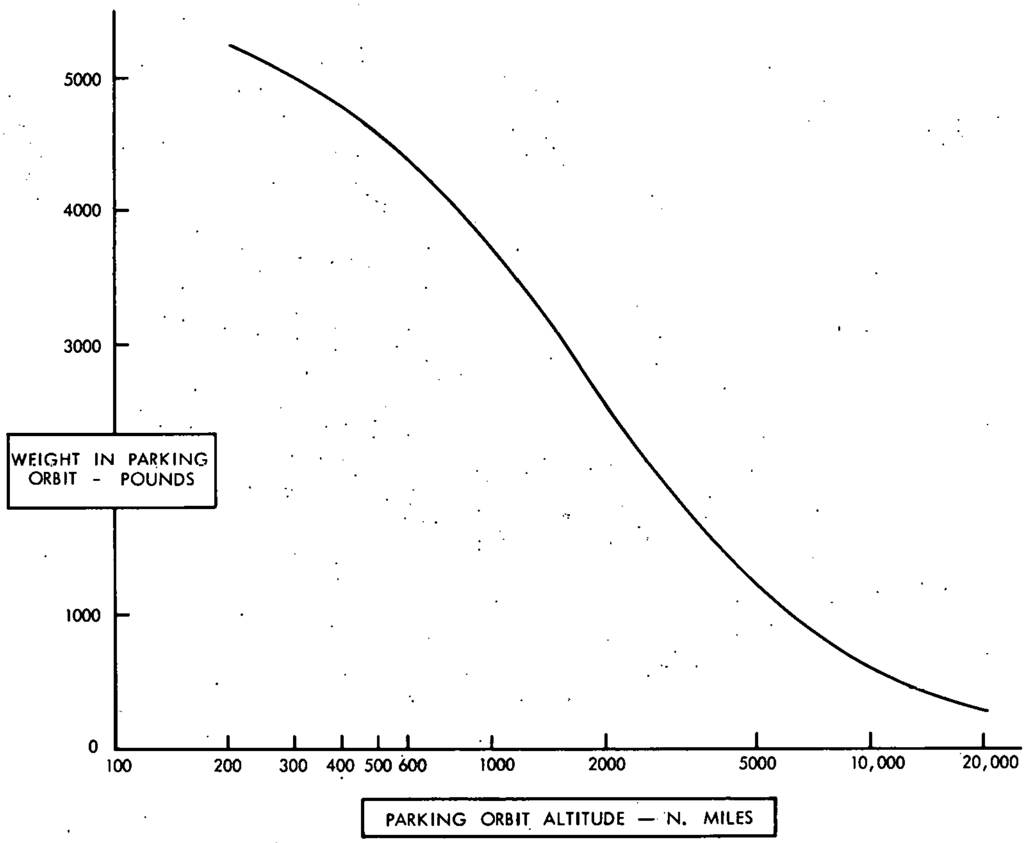

Figure 2 ASSUMED CHARACTERISTICS FOR TYPICAL BOOSTER CIRCULAR PARKING ORBIT

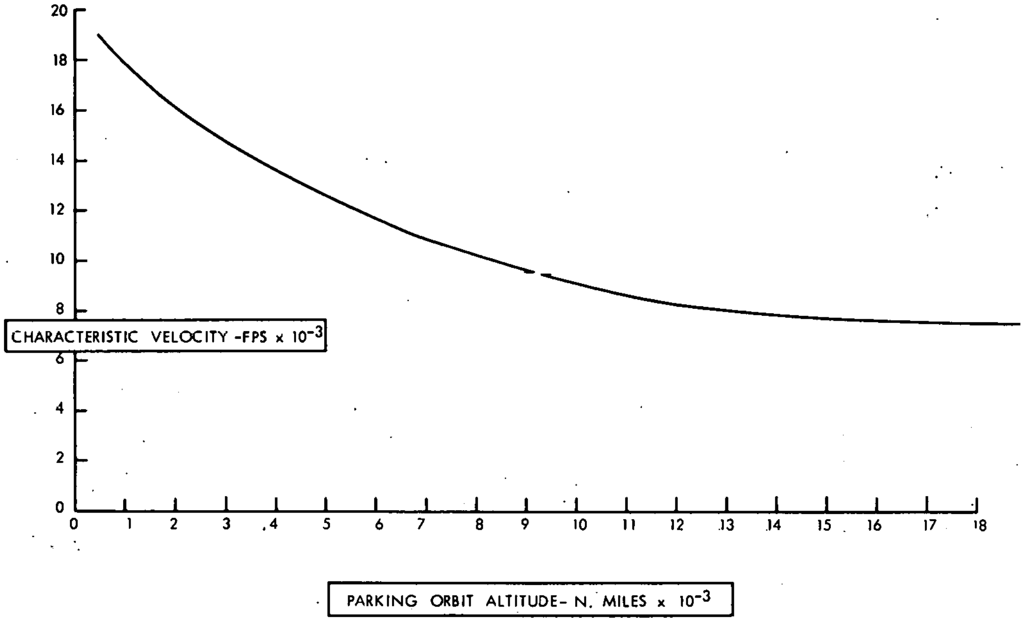

Figure 3 CHARACTERISTIC VELOCITY FOR LOW THRUST TRANSFER FROM INCLINED PARKING ORBIT (28.5°) TO SYNCHRONOUS EQUATORIAL ORBIT

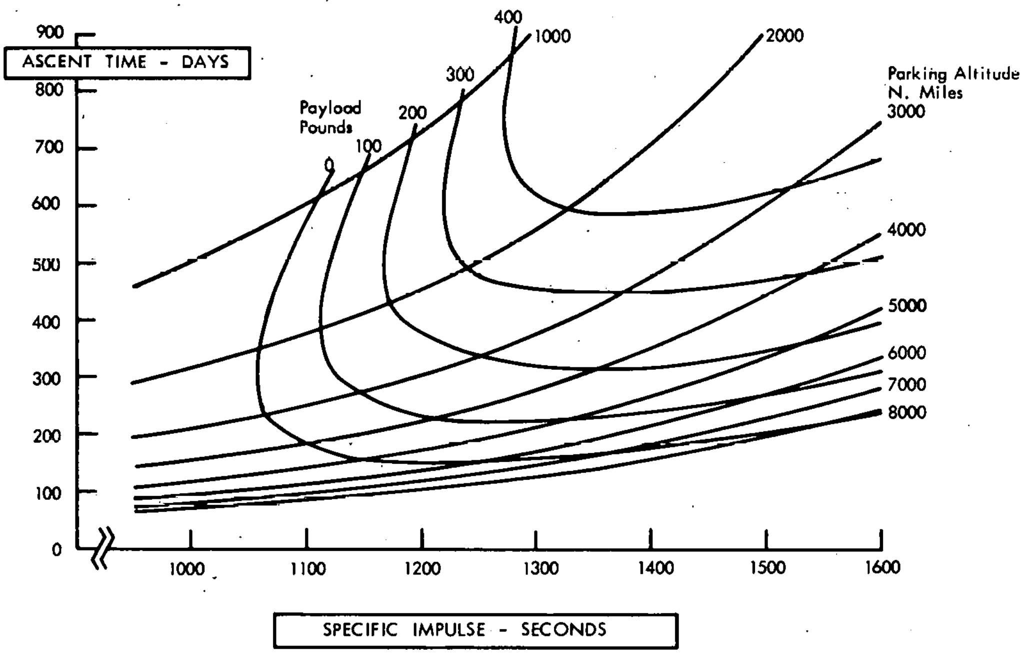

Figure 4 TRANSFER TO SYNCHRONOUS EQUATORIAL ORBIT
Propellant - Hydrogen
Power - 3 KW
Power Supply Weight = 100 #/KW

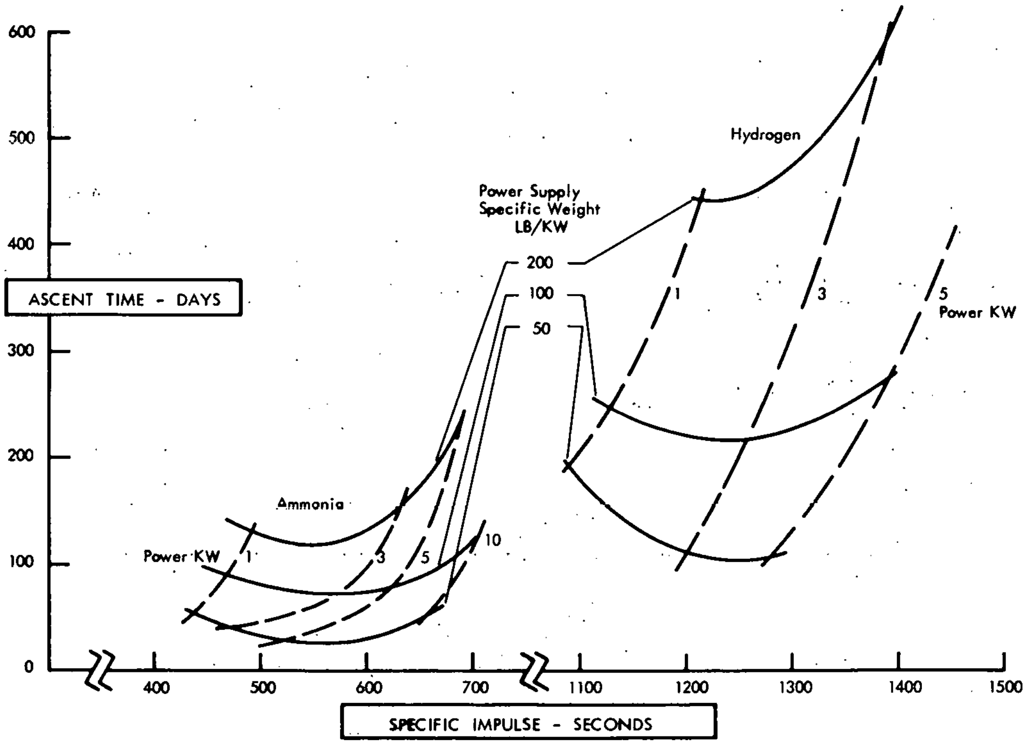

Figure 5 OPTIMUM TIME TO TRANSFER TO SYNCHRONOUS EQUATORIAL ORBIT
Payload - 100 Lbs. exclusive of Power Supply

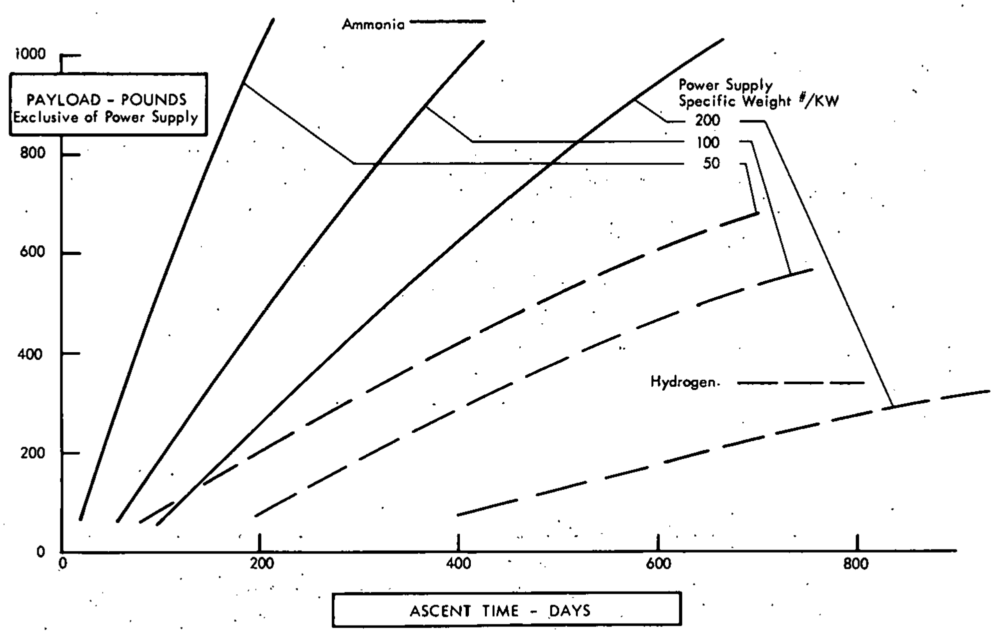

Figure 6 PAYLOAD IN SYNCHRONOUS EQUATORIAL ORBIT AS A FUNCTION OF MINIMUM ASCENT TIME

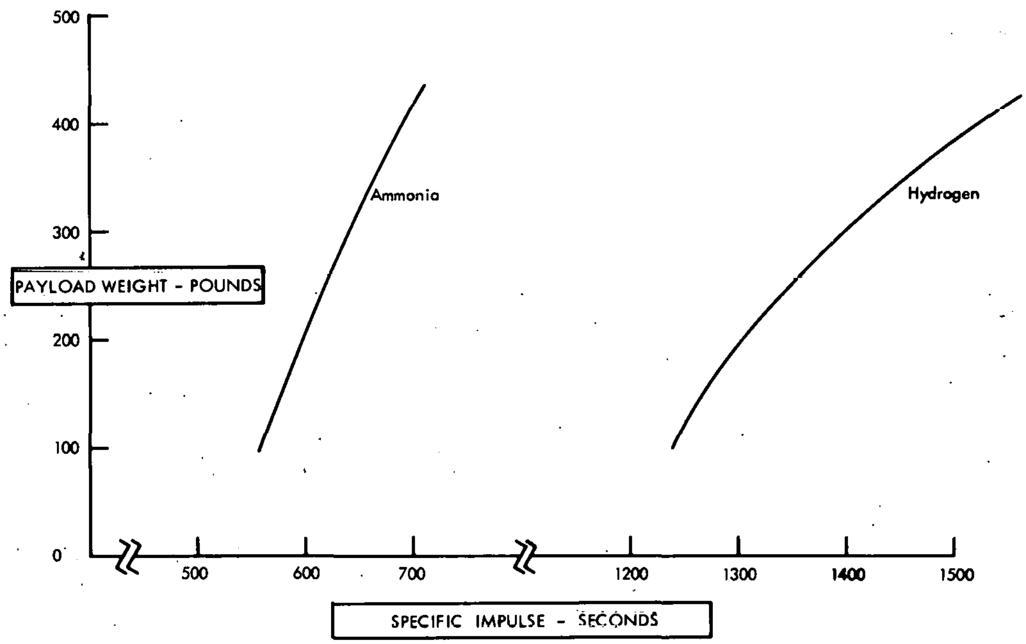

Figure 7 PAYLOAD IN ORBIT AS A FUNCTION OF OPTIMUM I_{sp} FOR MINIMUM ASCENT TIME
(Independent of Power Supply Specific Weight)

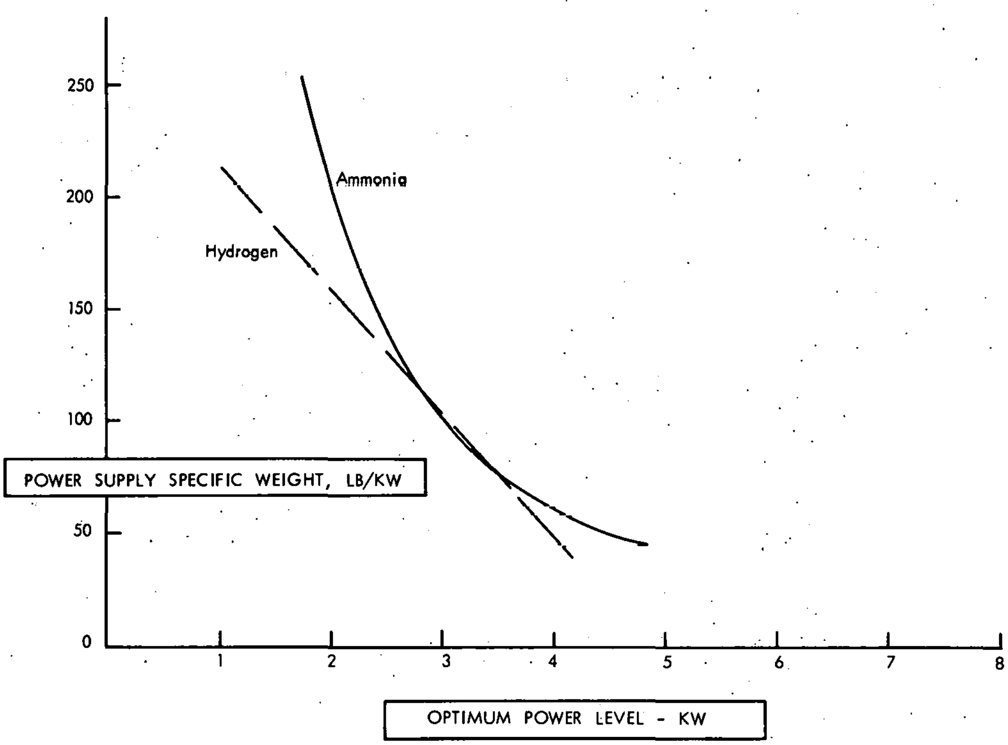

Figure 8 OPTIMUM POWER LEVEL AS A FUNCTION OF POWER SUPPLY SPECIFIC WEIGHT

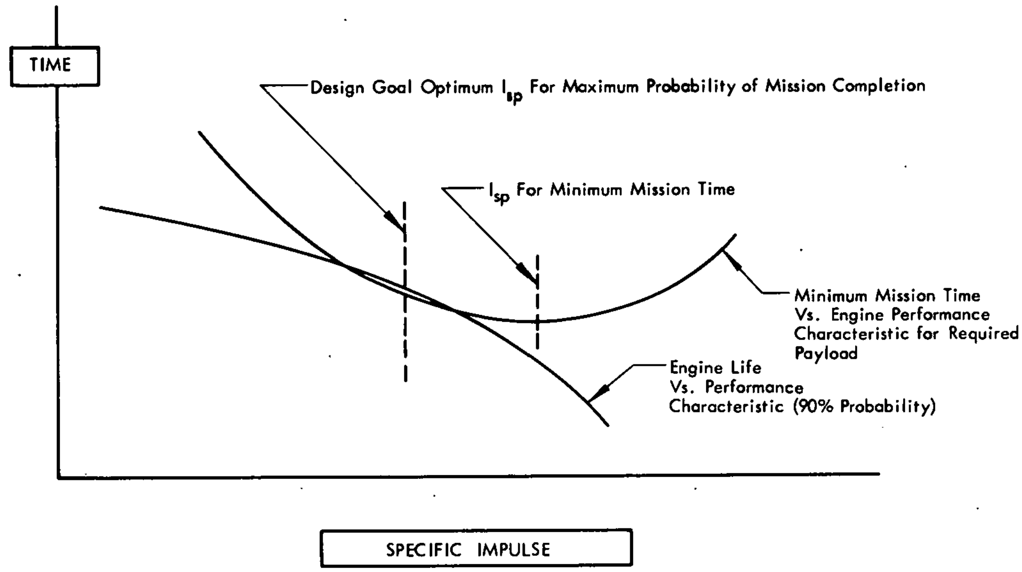

Figure 9 ILLUSTRATION OF METHOD FOR DETERMINING ENGINE DESIGN PERFORMANCE GOALS