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Preliminary Mission Studies for a Small Nuclear Engine

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scientific laboratory
of the University of California
LOS ALAMOS, NEW MEXICO 87544

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Preliminary Mission Studies for a Small Nuclear Engine

by

J. D. Balcomb

PRELIMINARY MISSION STUDIES
FOR A SMALL NUCLEAR ENGINE

J. D. Balcomb

ABSTRACT

Preliminary mission studies were made at the Los Alamos Scientific Laboratory for a small nuclear engine. Single-burn, multiple-burn, and multiple-stage planetary probes, as well as lunar and geosynchronous shuttle vehicles, were considered. Results indicate that a 300-MW engine with a thrust of ~ 15 000 lbf, an engine mass of 6000 lbm, and a specific impulse of 825 to 875 sec would be very attractive for a wide variety of advanced missions and payloads, including many missions that have previously been considered for the much larger NERVA engine. The small engine is particularly suitable for unmanned missions to the outer planets.

INTRODUCTION

The Los Alamos Scientific Laboratory has proposed the development of a small nuclear rocket engine to obtain the maximum benefit from the accomplishments attained to date in the nation's nuclear rocket program. The small-engine program would provide continued development of nuclear rocket technology through flight testing and would result in a useful space propulsion system.

The small-engine mission study presented in this report indicates that such a device is very attractive for a number of advanced space applications, with a growth potential that is clearly superior to that of chemical propulsion systems. The engine would be useful initially for early unmanned missions to the outer planets (for example, a 5000-lb-payload direct flight to Neptune) and would provide the capability both for much larger unmanned payloads to the outer planets and for carrying heavy payloads to and from the moon and/or geosynchronous space bases. Another attractive feature is that a complete nuclear stage capable of useful missions can be launched from a single Earth Orbital Shuttle vehicle. Early flight testing from a safe earth orbit could be accomplished with either the Earth Orbital Shuttle or a Titan III launch vehicle.

A small-engine definition study will be performed at LASL, beginning February 1, 1972 and continuing through Fiscal Year 1973. In support of the mission-study work that will be performed concurrently within NASA, it has seemed appropriate to publish the results of the preliminary mission analysis already done at LASL while investigating the small-engine concept. These results are of limited scope, covering only a single power density and few performance variations; the results will be superseded as the study progresses in a more comprehensive effort.

Data generated in the study are intended to serve as a guide to the general ranges of thrust and corresponding engine mass that would be of greatest utility. Three classes of mission were considered: (1) unmanned planetary probes that would be representative of early flight tests and would demonstrate the nuclear engine, with constraint on total initial mass in earth orbit; (2) more advanced unmanned planetary probes in which staging or multiple-burn strategies are allowed, and (3) reusable shuttle missions carrying heavy payloads to and from lunar and/or geosynchronous orbit.

A brief description of the engine on which most of this analysis is based is included in an Appendix.

CONCLUSIONS

Major conclusions of the study are:

- The power range of 150 to 400 MW brackets the optimum engine size for single-burn probe missions out of earth orbit for an initial mass in earth orbit of 30 000 to 100 000 lbm and a wide range of payloads. In this range, the nuclear engine is competitive with a chemical stage for Jupiter swingby missions.

- Advanced probe missions characteristic of the 1980's (e.g., 5000 lbm payload, 2800-day direct flight to Neptune) can be launched with either a two-burn single stage or two single-burn stages based on the use of a 300-MW/6000-lbm engine (15 000 lbf thrust) and a total initial mass in earth orbit of 65 000 to 75 000 lbm.

- The same 300-MW/15 000-lbf thrust engine, upgraded for long life, multiple starts, and reusability--shielded for manned use--is quite suitable for shuttling large payloads to a lunar or geosynchronous orbit from a safe base earth orbit and back. A double perigee burn is recommended to avoid excessive gravity losses for the initial burn out of earth orbit if the total vehicle mass exceeds ~ 200 000 lb. Although the reusability of the small engine will be decreased due to the longer required burning time, this disadvantage is more than offset by the smaller hydrogen and stage masses associated with the reduction in engine mass.

GENERAL ASSUMPTIONS FOR ALL CASES STUDIED

Initial Orbit Parameters (Circular Orbit):

Perigee = 4260 miles (255 n-miles altitude).
Velocity = 25 039 ft/sec.

Tank Mass (M_s):

M_s Nuclear = 0.16 (propellant mass).

M_s Chemical = $\frac{\text{propellant mass}}{9}$, (i.e., $\lambda' = 0.90$).

Engine Mass (M_e):

M_e Nuclear = 1500 + 15 (power, MW), lbm*.
M_e Chemical = Engine mass is included in tank mass.

*See Appendix.

Engine Thrust, Nuclear:

$$\text{Thrust (lbf)} = \frac{41\ 250\ (\text{power, MW})}{(I_{sp},\ \text{sec})}$$

Flow Rate, Nuclear:

Flow rate (lbm/sec) = (thrust, lbf)/(I_{sp}, sec).

Specific Impulse of Chemical Stages: 460 sec.

Equivalent Impulsive ΔV:

Results are presented in terms of an equivalent impulsive ΔV applied in the initial orbit. In terms of the final orbit achieved, this ΔV is given by

$$\Delta V_{eq} = \sqrt{2V_o^2 + (e-1)\ K/r_p} - V_o$$

where: e = eccentricity of orbit,
r_p = perigee of orbit,
K = earth gravity constant, 95 800 miles^3/sec^2,
V_o = velocity in initial orbit, 25 039 ft/sec

The ΔV is equivalent in the sense that ΔV_{eq} applied impulsively in the initial orbit will result in the same hyperbolic excess velocity as the orbit defined by e and r_p.

SINGLE-BURN PLANETARY PROBES

General

For early planetary probes it is assumed that the nuclear-engine operation will be constrained to a single burn, without cooldown, and that the total initial mass in earth orbit will be constrained by launch-vehicle capabilities. Three cases were treated as follows:

Initial Mass, lbm	Launch Mode
• 30 000	Titan III launch
• 50 000	Earth Orbital-Shuttle (EOS) launch
• 100 000	Orbital assembly of two EOS launches

Results

Results of these studies are presented in Figs. 1 through 5, as discussed in the following paragraphs.

Figures 1 and 2: The ΔV_{eq} is given as a function of payload plus engine mass for cases of initial mass equal to 30 000 lbm (Fig. 1) and 50 000 lbm (Fig. 2) for different thrust levels. The

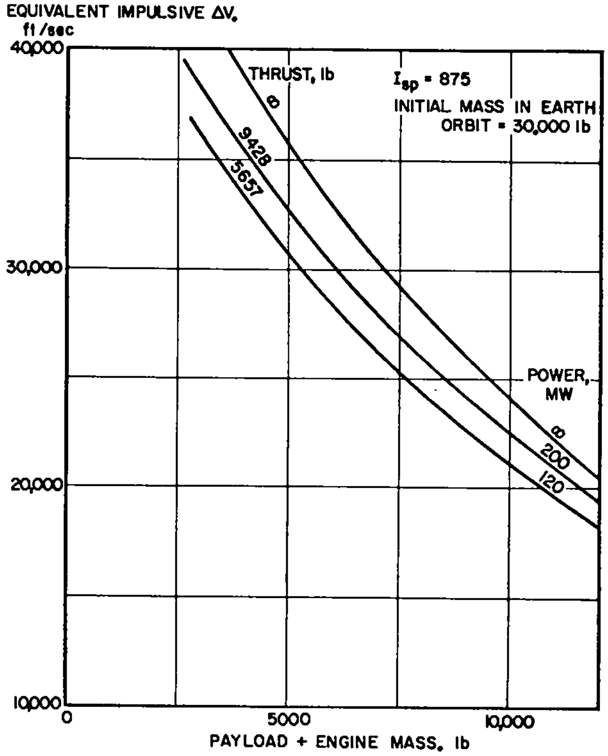

Fig. 1. Equivalent ΔV for an initial mass of 30 000 lbm.

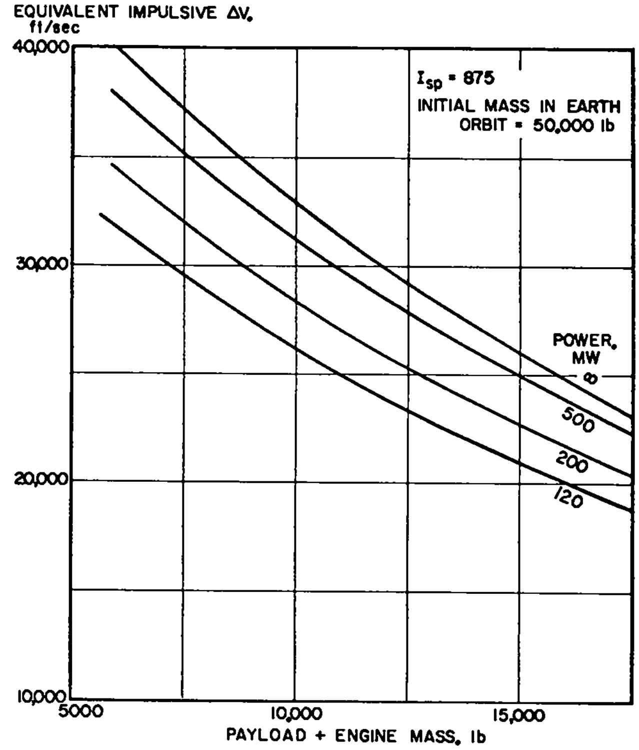

Fig. 2. Equivalent ΔV for an initial mass of 50 000 lbm.

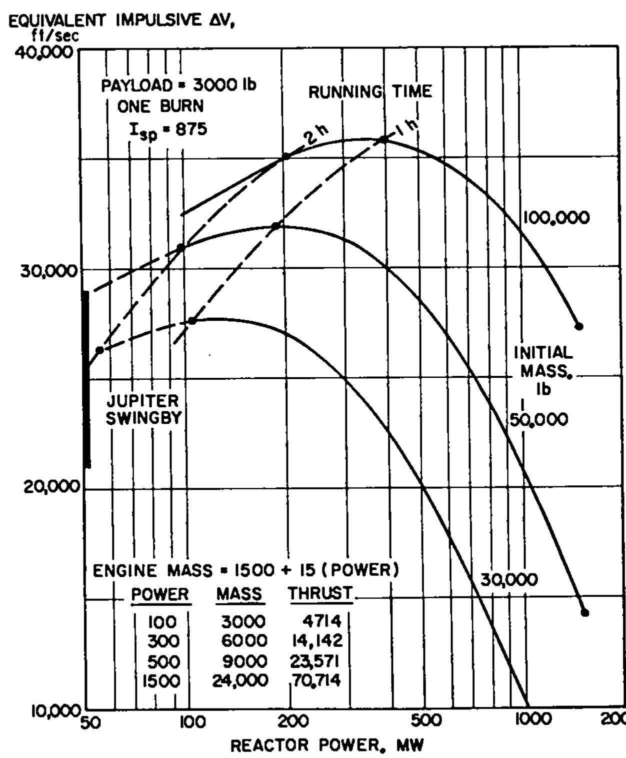

Fig. 3. Equivalent ΔV as a function of power for a 3000-lbm payload.

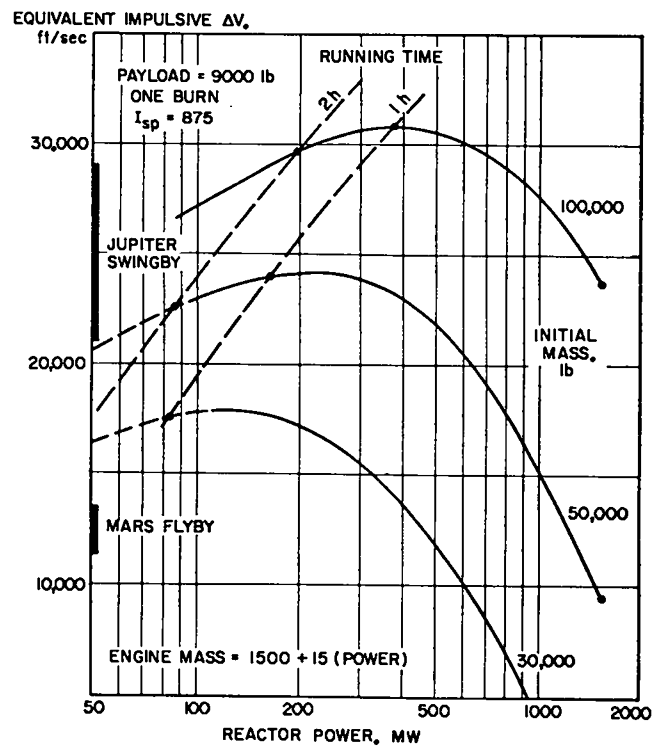

Fig. 4. Equivalent ΔV as a function of power for a 9000-lbm payload.

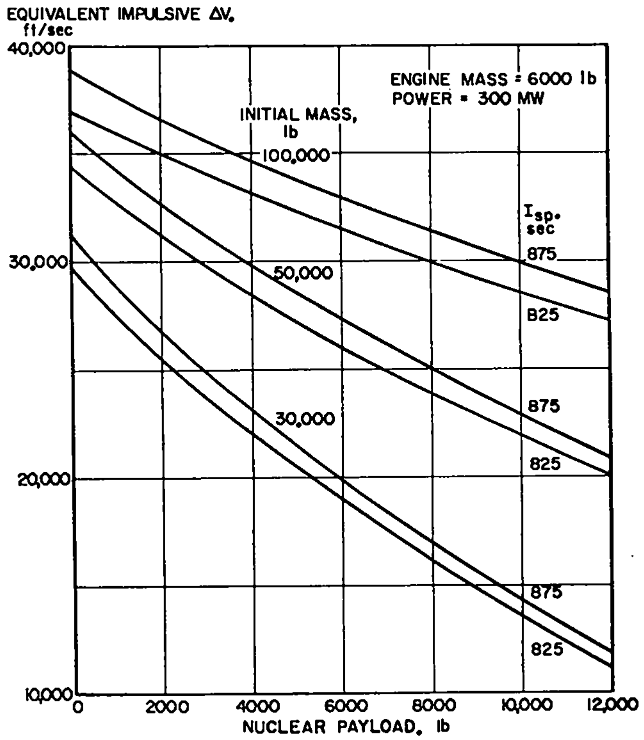

Fig. 5. Equivalent ΔV for a 300 MW/6000 lbm engine.

gravity loss is the ΔV difference between the finite-thrust ΔV_{eq} and the infinite-thrust ΔV_{eq}. Note that these figures contain no implied relationship between engine weight and thrust.

Figures 3 and 4: The data of Figs. 1 and 2 are replotted vs power for two fixed payloads of 3000 lbm (Fig. 3) and 9000 lbm (Fig. 4) with the assumed engine weight-power scaling law. Curves for an initial weight of 100 000 lbm are also presented. These curves point out the existence of an optimum power level for a given initial mass. Note that the optimum power level is not sensitive to payload (compare Figs. 3 and 4). A power level in the range of 200 to 300 MW is near optimum for the range of parameters assumed.

Figure 5: Data from Figs. 3 and 4 are extended for the particular case of 300 MW power (6000-lbm engine) to show a range of payloads. Corresponding results for I_{sp} = 825 sec are also shown indicating a small degradation in performance. Because power is held constant, the thrust is slightly higher (15 000 lbf vs 14 143 lbf for I_{sp} = 875 sec) partially offsetting the I_{sp} increase.

Comparison with Chemical Engines (Fig. 6)

The nuclear engine would generally be operated with an upper chemical stage to maximize the total ΔV achieved. The mass of the chemical stage can be optimized for each case. However, in many missions a Jupiter swingby should be utilized to increase the effectiveness of the chemical-stage impulse by burning at perijove. The nominal required ΔV_{eq} from earth orbit, to reach Jupiter in ~ 500 days within the 1982-1989 time period, is 24 000 ft/sec. Therefore, for comparison purposes, it is useful to calculate the initial mass required in earth orbit as a function of the payload for ΔV_{eq} = 24 000 ft/sec for both nuclear and chemical stages. For the nuclear stage, the data can be taken from Fig. 5. These data have been extended and are plotted in Fig. 6. For the chemical stage the mass ratio is constant at 9.22 and this straight line is also plotted in Fig. 6. No gravity losses have been charged to the chemical system because it is assumed that the thrust would be comparable to the initial mass. It can be seen from the intersection of the curves that a saving in initial mass is obtained with a nuclear stage for payloads exceeding 3100 lbm and that the saving is greater than 50% for payloads exceeding 7100 lbm.

Comparison with a 1500-MW Nuclear Stage (Fig. 6)

For the scaling law used, a 1500-MW nuclear stage would have an engine mass of 24 000 lbm. This is consistent with estimates of a modified

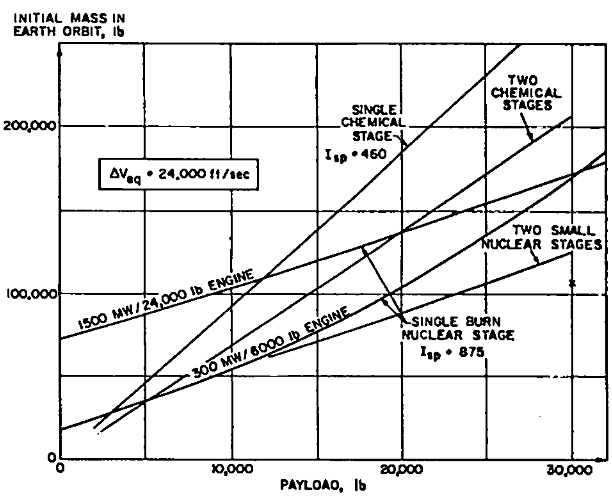

Fig. 6. Vehicle mass requirements for a 24 000 ft/sec mission.

NERVA engine operating on one turbopump, without a shield. Thus the extension of the scaling law previously used up to this power level is reasonable. Figures 3 and 4 indicate that, for an initial mass in earth orbit in the range of 30 000 to 100 000 lbm, the 1500-MW-engine with a mass of 24 000 lb is far beyond optimum. This same power-engine mass combination has been calculated for the ΔV_{eq} = 24 000 ft/sec case for a range of payloads, with the results presented in Fig. 6. It can be seen that the small nuclear engine requires a smaller initial mass in earth orbit than the large nuclear engine for payloads up to 30 500 lbm (corresponding to 170 000 lbm in earth orbit). At this point, however, staging or multiple-burn strategies can be used with the small engine to substantially improve performance as will be shown in the next section.

MULTIPLE-BURN AND MULTIPLE-STAGE PLANETARY PROBES

General

The gravity losses associated with a single-burn use of the nuclear engine become excessive for either large payloads or large-ΔV missions. An effective approach for overcoming these losses is to use either multiple burns on a single nuclear engine or to use multiple nuclear stages rather than resort to a heavier, higher-thrust engine.

As mentioned in the previous section, Jupiter swingbys can be used very effectively to reach the outer planetary regions. For example, a 2800-day mission to Neptune requires a ΔV_{eq} of 34 000 ft/sec for a direct flight, whereas a ΔV_{eq} of 25 000 ft/sec is sufficient with a Jupiter swingby. However, this applies only if Neptune is at the correct point in its orbit as the probe passes. The best opportunities occur in 1979 and thereafter at 12.7-year intervals. The situations for Saturn and Uranus are similar. Interest in missions to the outer planets will continue during the 1980's even though the propulsion requirements are high; thus direct flights to the outer planets will be prospective missions for a small nuclear engine. As an example of such a mission, a ΔV_{eq} of 34 000 ft/sec has been selected. This corresponds to a hyperbolic excess velocity of 47 241 ft/sec for the simple coplanar case analyzed.

Two-Burn Planetary Probes

The large gravity losses associated with a single long burn out of earth orbit can largely be overcome by multiple burns using the perigee-kick technique. Most of the advantage to be gained for the small engine can be obtained by using two burns rather than one. Disadvantages of this technique are that the operation is more complicated and that the nuclear engine must be cooled down after the first burn. The following guidelines were used in analyzing this technique:

- The first burn is terminated on attaining a 24-h intermediate elliptical orbit ($\Delta V_{eq} \approx 8905$ ft/sec).

- Eight percent of the hydrogen consumed in the first burn is used for cooldown, with no impulse benefit. (The total hydrogen needed for shutdown and cooldown more nearly amounts to 12% and would, in fact, be used for useful thrust at about one-half the full-power specific impulse; however, the effectiveness of this impulse is diminished because it occurs in regions of decreased orbit velocity relative to perigee velocity.)

- The second burn is initiated 60 deg prior to the intermediate-orbit perigee.

- There is no cooldown after the second burn.

The results are presented in Fig. 7, in which the initial mass in earth orbit is plotted as a function of the payload delivered. A typical case is as follows:

	Vehicle Mass, lbm	Δ Mass, lbm
Initial total vehicle	80 000	-
Burn 1, 23.15 min	-	22 464
Burn-1 cooldown	-	1 797
Mass at start of Burn 2	55 739	-
Burn 2, 35.96 min	-	34 872
Mass at burnout	20 867	-
Drop engine and tank	-	15 461
Payload	5 406	-

A single case with an actual cooldown profile has been calculated to check the validity of the assumptions and to determine the effect of the cooldown on the intermediate orbit. The results are shown in Fig. 8. The burn intervals were

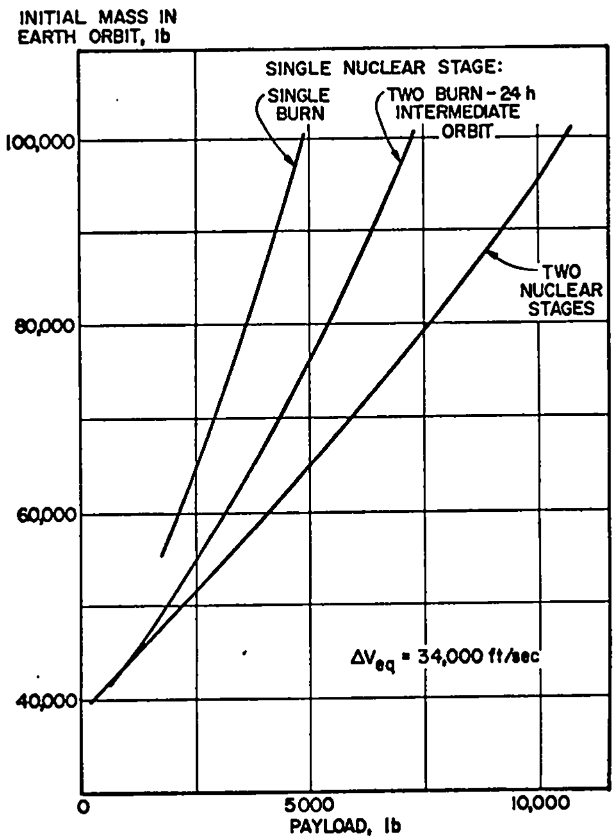

Fig. 7. Vehicle requirements for a 34 000 ft/sec mission.

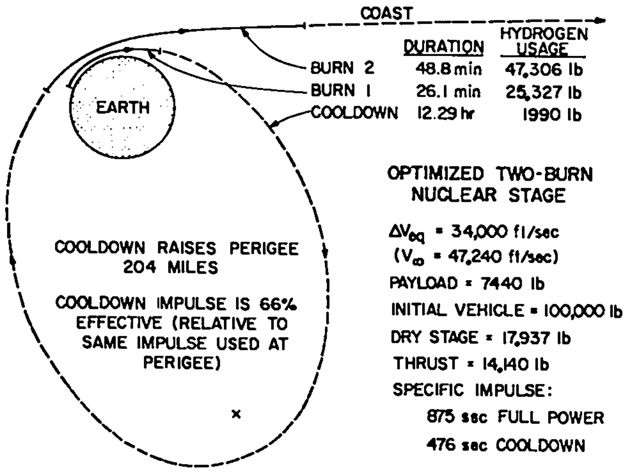

Fig. 8. Orbit schematic for a two-burn escape maneuver with cooldown considered.

optimized for maximum payload. All impulse during the two burns and during cooldown was applied tangentially (along the direction of the velocity vector). The payload achieved is 7440 lbm compared to 7270 lbm for the assumption made in arriving at Fig. 7. The effects of the cooldown are:

- The intermediate orbit apogee* is raised from 26 440 miles at the end of the first burn to 29 880 miles at the beginning of the second burn.
- The perigee* is similarly raised from 4527 to 4731 miles.
- The total impulse is 44% less effective (in increasing ΔV_{eq}) than the same impulse applied at perigee.

Two-Stage Planetary Probes

An advantage over the two-burn case can be obtained by staging, despite the added mass of an extra engine, provided the staging is performed prior to earth escape so that the second-stage impulse can be used at an intermediate orbit perigee. Thus no cooldown is required for either engine, and the perigee-kick technique is utilized.

The results, plotted in Fig. 7, clearly show an improvement over the two-burn case.

A typical two-stage case would have the following characteristics:

	Vehicle Mass, lbm	Δ Mass, lbm
Initial total vehicle	65 000	-
Burn Engine 1, 19.64 min		19 048
Mass at burnout	45 952	-
Discard tank and engine		9 048
Mass in 48-h orbit	36 905	-
Burn Engine 2, 23.02 min		22 326
Mass at burnout	14 579	-
Discard tank and engine		9 572
Payload	5 006	-

The first stage is left in a 48-h orbit. It may be desirable to carry enough extra hydrogen to boost the used first stage to earth-escape conditions after staging; this would increase the total initial mass in earth orbit by 620 lbm in order to achieve the same delivered payload as in the example given above.

*Measured in statute miles from earth center.

SHUTTLE VEHICLES

General

Although the first small nuclear engines would presumably not be built for long life, reusable stages, or be man-rated, it is useful to ascertain how a small nuclear engine, upgraded to fit these criteria, would perform in missions that have historically been postulated for the 75 000-lbf NERVA engine. The missions are shuttle or logistics round trips to synchronous or lunar orbit. The first obvious difference is the low thrust, 15 000 lbf, of the small engine. If the missions were to be performed as postulated for NERVA, i.e., with a single burn out of earth orbit, then the factor-of-five reduction in thrust would lead to large gravity losses (particularly for the lunar mission). However, as indicated earlier, there is a straightforward way to overcome these losses by the perigee-kick technique. In this technique the thrust is divided into short segments taken near orbit perigee. The result is a series of elliptical orbits, each with nearly the same perigee but increasing apogees. In this way thrust is utilized near maximum orbit velocity and is therefore maximally effective. By limiting thrusting periods to 0.5 h maximum, the gravity losses in earth orbit can be kept to a few percent.

This, then, corresponds to the situation postulated for a NERVA-powered nuclear stage in which the first burn out of earth orbit lasts roughly 0.5 h and the corresponding gravity loss is roughly 6% (for the lunar mission). If a nuclear stage powered by a small nuclear engine would use this perigee-kick technique, the mission ΔV's would be comparable to those for NERVA; however, because the small engine weighs much less the stage weights would be appreciably less. The main penalty, compared to NERVA, would be an operating time roughly four times longer than that of a NERVA engine, restricting the use of the small engine to fewer missions. However, the savings in stage weight easily justify this reduction in engine lifetime.

Basis for Comparison with NERVA

Years of work were spent in the Nuclear Flight System Definition Study performed for NASA by three contractors. One of these studies, all of which went into great detail of stage design, is used herein as the basis for predicting stage weights for the nuclear shuttle. The stage design referred to is the Lockheed Missiles and Space Company (LMSC) modular-tank concept described in the Phase-III performance review document LMSC-A981482, December 17, 1970. This particular design is suitable as a basis for comparison because it incorporates a "propulsion module" which includes a medium-capacity tank (37 371 lbm of LH_2) to which additional propellant modules are clustered to make up the large stage. Stage-weight differences can be predicted by accounting for known differences between the LMSC stage design based on NERVA propulsion and a stage design based on a small nuclear propulsion engine.

LMSC Stage Weights

The reference total stage weight is 81 241 lbm (dry inert weight) plus 269 477 lbm of LH_2 and 2200 lbm of reaction-control-system (RCS) propellant. The weight of the propulsion module alone is 42 080 lbm (dry inert weight) plus 37 371 lbm of LH_2 propellant. Major subassembly weights are 27 798 lbm for the NERVA engine, 5900 lbm for the external shield, and 2034 lbm for the astrionics and auxiliary propulsion in the control and assembly module.

Stage Weight Scaling

The following formula is used for stage-weight scaling:

	Weight, lbm	
	NERVA	Small Nuclear Engine
Engine	27 798	6 500
Tank	2 403 + 0.16 M_ℓ	2 403 + 0.16 M_ℓ
Astrionics + auxiliary propulsion	2 035	2 035
Shield	5 900	2 000
Total	38 136 + 0.16 M_ℓ	12 938 + 0.16 M_ℓ

M_ℓ = hydrogen capacity in lbm.

This basis for scaling gives both the correct propulsion-module weight and the correct total stage weight for the LMSC design. The 0.16 factor for scaling on hydrogen capacity includes all weights not specifically itemized, e.g., tank structures, meteoroid and thermal protection, assembly and docking structures, propellant loading and tank controls, and tank-feed systems, all of which are presumed to scale with the propellant load. The reduction from 38 136 to 12 938 lbm in basic stage weight is attained specifically as a result of reductions in engine and shield mass. The mass of the small nuclear engine of 6500 lbm is obtained by adding 500 lbm to the weight of the engine, to account for the insulators required for long life. The shield mass of the small nuclear engine is an educated guess based on the scaled-down engine diameter and pump weights. Actual shield weights will depend strongly on stage configuration: A long array of tanks would obviate the need for supplemental external shielding, but a single squat tank might require some extra shielding amounting perhaps to > 2000 lbm. This shielding requirement would be a consideration in the final choice of stage configuration. Errors made in the given assumptions will tend to be self-compensating because reductions in shield weight will be accompanied by required increases in tank weight due to the poorer configuration for hydrogen storage.

Lunar Mission ΔV's

The lunar shuttle selected as the baseline NERVA mission will be considered first. The payload is taken from the base orbit of 255 n-mile to lunar polar orbit and is there exchanged for a return payload.

The required velocity increments are taken from the LMSC reference lunar mission. Eight engine burns are required as follows:

Maneuver	ΔV, ft/sec
Translunar insertion (includes 666 ft/sec gravity loss)	10 705
Midcourse correction	68
Lunar orbit insertion	2 909
Total	13 682
Transearth injection	
Burn 1 - 36 h ellipse	1 960
Burn 2 - plane change	910
Burn 3 - leave moon	1 596
Midcourse correction	68
Earth orbital insertion (includes 79 ft/sec gravity loss)	10 213
Total	14 747

These ΔV's are based on a 60 n-mile lunar polar orbit and on leaving the moon at the least favorable time. For our purposes these eight maneuvers can be consolidated into two main intervals---outbound and return, with ΔV's of 13 016 and 14 668 ft/sec, respectively, plus the appropriate gravity losses.

Payloads

Payloads are taken from the reference mission used by LMSC. For our purposes the payloads include the actual delivered payload plus all ancillary material unloaded in the same orbit.

	Mass, lbm
Unloaded in Lunar Orbit	
Actual payload	85 965
Ullage gas dropped	1 141
Boiloff in orbit	1 173
RCS consumption	550
Total	88 829
Loaded in Lunar Orbit	
Total	20 114
Unloaded in Earth Orbit	
Actual payload	20 114
Residuals	2 012
RCS propellant	550
Total	22 676

Equivalent Specific Impulse

The LMSC study used an engine specific impulse of 825 sec at full power. However, their study also accounted for reactor startup, shutdown and cooldown, dropping of ullage gas, and boiloff and RCS propellant usage in transit. All these impulse degradations can be lumped together and a total effective specific impulse can be calculated, based on the actual mass ratio and actual total ΔV's for the entire outbound and return portions of the trip. This process leads to the following effective specific-impulse values:

Outbound = 783.7 sec
Return = 784.9 sec.

By using a value of 784.2 sec for both legs of the trip, one can duplicate the actual final mass. This value of 784.2 sec will be used for this study.

Lunar Shuttle Based on Small Nuclear Engine

The above procedure provides a straightforward way of calculating the stage weight for a small nuclear engine that would accomplish the same mission as the NERVA with the same payloads and using the same effective specific impulse. The calculation is iterative because the gravity losses on both the translunar insertion (TLI) and earth-orbital insertion (EOI) are dependent on stage mass.

Results of this comparison are shown in Table I. As indicated in this table, large savings in stage and hydrogen mass can be realized with the small engine by using two or more burns for the translunar insertion, as compared with the heavier NERVA-based stage. This benefit is obtained at the expense of longer engine running time and a somewhat more complex mission.

The trajectories for the one-burn and two-burn translunar-injection cases are shown in Figs. 9 and 10, and the total vehicle mass comparison is shown in Fig. 11.

Synchronous-Orbit Shuttle

The shuttle of large payloads to and from earth synchronous orbit is basically similar to the lunar-orbit shuttle. The total round-trip ΔV is very nearly the same, but is divided differently resulting in lower net gravity losses.

TABLE I

COMPARISON OF LUNAR SHUTTLE BASED ON SMALL ENGINE WITH SHUTTLE BASED ON NERVA

	Small Nuclear Engine			NERVA
Number of burns for translunar insertion	1	2	4	1
Initial mass in earth orbit, lbm	464 040	365 760	327 340	438 832
TLI gravity loss, ft/sec	4 850	1 990	500	666
EOI gravity loss,* ft/sec	1 070	800	700	79
Hydrogen capacity, lb	312 839	228 110	194 989	269 477
Dry stage mass, lb	62 999	49 442	44 136	81 241
Total full-power running time, h	4.46	3.24	2.76	0.75
Intermediate orbit period(s), h	none	3.49	2.13 3.39 7.81	none
Intermediate orbit apogee(s), miles	none	9 464	6 131 9 841 20 358	none

*Single burn in all cases.

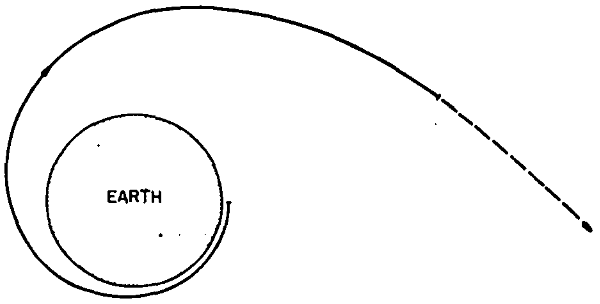

Fig. 9. Single-burn translunar injection for a lunar shuttle mission.

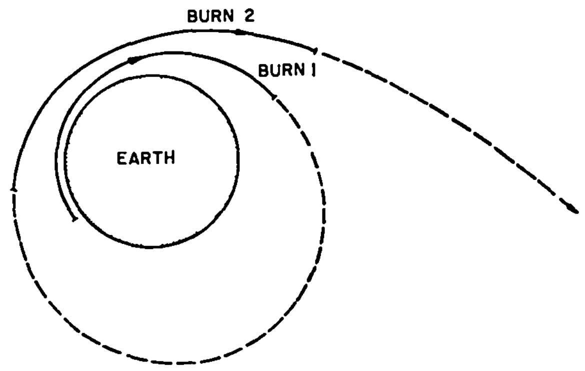

Fig. 10. Two-burn translunar injection for a lunar shuttle mission.

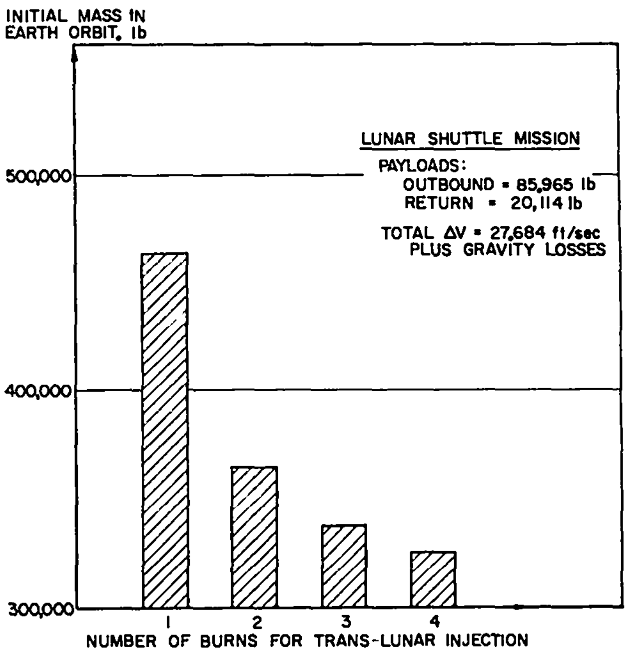

Fig. 11. Initial mass requirements for a lunar shuttle vehicle using a small nuclear engine.

The total stage scaling law, the overall engine thrust, and the engine performance are the same as those used for the lunar-orbit shuttle described earlier. A shielded, reusable vehicle is assumed.

The ΔV's without gravity losses are:

Maneuver	ΔV, ft/sec
1. Out of 255 n-mile orbit	7808
2. Into synchronous orbit*	5970
3. Out of synchronous orbit*	5970
4. Into 255 n-mile orbit	7808

*Includes 28.5 deg plane change

Calculations for finite thrust were performed as follows. For a given mass in earth orbit a tangential-thrust burn was performed until an orbit was achieved tangent to the synchronous orbit (orbit apogee equal to synchronous altitude). The mass on this Hohmann transfer orbit was then multiplied by the appropriate mass ratio for the impulsive maneuver* required to inject into the synchronous orbit (including a 28.5-deg plane change) at the transfer-orbit apogee. This yields the mass injected into synchronous orbit. The mass leaving synchronous orbit was calculated similarly by performing Maneuvers 3 and 4 in reverse order, assuming some final mass in earth orbit. Each such pair of calculations corresponds to a single pair of outbound and return

*The gravity loss for Maneuvers 2 and 3 is small and is neglected.

payloads, which are calculated from the difference between the synchronous-orbit mass and the final mass (after accounting for the dry-stage mass which depends on total hydrogen burned). A number of such calculations were made and have been used to construct Fig. 12. From this figure it is possible to determine the vehicle requirements for any combination of outbound and return payloads. The hydrogen requirement for any particular case can be determined from the formula

$$\text{Hydrogen Capacity} = \frac{\begin{bmatrix}\text{Initial mass} \\ \text{in earth orbit}\end{bmatrix} - \begin{bmatrix}\text{Outbound} \\ \text{payload}\end{bmatrix} - 12\,938}{1.16}$$

The gravity loss is significant only for Maneuver 1 and only for an initial mass in earth orbit greater than 200 000 lbm. This gravity loss can be reduced by splitting Maneuver 1 into two separate burns in the now-familiar manner. The vehicle requirements for this procedure are presented in Fig. 13.

Thrust Vector Optimization

The results presented are based on tangential thrust, i.e., the thrust vector is aligned with the velocity vector throughout the burn. For a finite-thrust burn it is always possible to achieve a greater equivalent ΔV by steering the thrust along some optimum program. The benefits to be gained by this process increase as the gravity losses increase. To ascertain the order-of-magnitude of the benefit achievable for the small engine, two cases have been studied that are representative of earth-escape and shuttle types of missions.

The first case considered a 100 000-lbm initial vehicle propelled by a 15 000-lbf-thrust engine to an effective hyperbolic excess velocity of 33 700 ft/sec in a single burn. Thrust-vector optimization resulted in a thrust-vector program beginning 23 deg below the velocity vector and slowly rotating tangent to the velocity vector with an approximately exponential shape. The gain produced by this optimization corresponded to an increase of 192 ft/sec or 0.57% in hyperbolic excess velocity.

The second case considered a two-burn lunar-shuttle translunar injection. Optimization produced not only a thrust-vector program but also defined optimum start and stop times for both burns. The benefit of the thrust vector considered alone was small, resulting in a 22.3-sec reduction in total burn time compared to the same burn intervals with tangential thrust. This burn-time reduction represents a saving of 0.32%.

Results of thrust-vector optimization during cooldown phases are not yet available.

ACKNOWLEDGEMENT

The thrust-vector optimization calculations were performed by J. R. Streetman, LASL, using the TOPCAT code supplied by Princeton University.

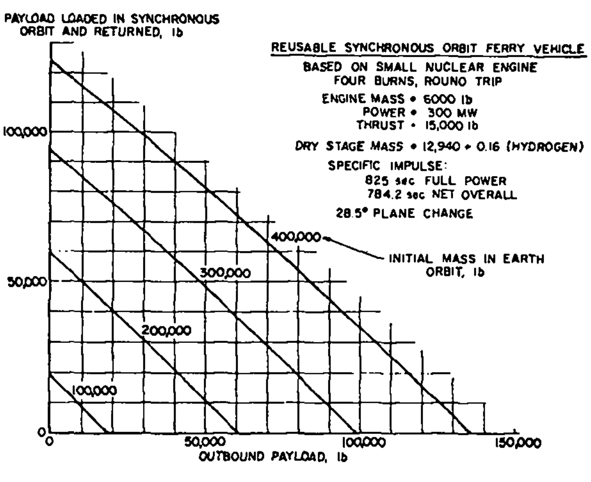

Fig. 12. Initial mass requirements for a synchronous orbit ferry vehicle.

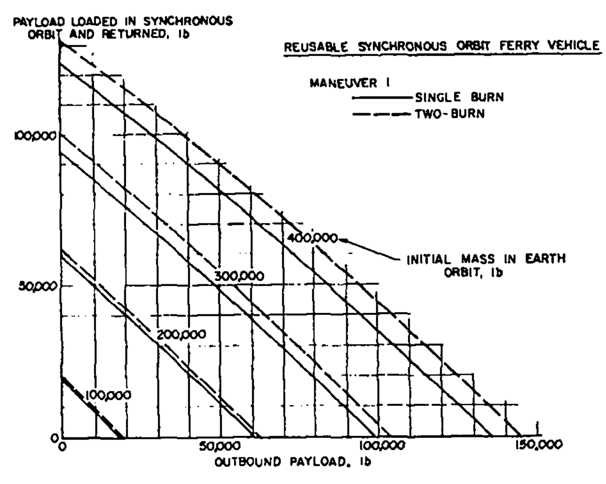

Fig. 13. Effect of two-burns for Maneuver 1.

APPENDIX

SMALL ENGINE DESCRIPTION

F. P. Durham

The small-engine reactor requires a large number of hydride-containing support elements to meet the nuclear criticality requirements. This results in a much higher ratio of support elements to fuel elements than in the NERVA-size reactor and thus provides a larger energy source from the tie-tubes to drive the turbine in a topping cycle. It has been determined that half the total reactor flow in the 2:1 (fuel element/support element) core pattern is adequate to cool the support elements and to drive the turbine. Preliminary estimates also indicate that half the total reactor flow would be sufficient for adequate nozzle cooling. By splitting the total flow equally between the support structure and the nozzle and using only the tie-tube flow to drive the turbine, the system pressure level can be held at a lower value than for NERVA. In addition, the pressure-vessel pressure is only slightly higher than the core-inlet pressure, which obviates the need for a separate inner pressure vessel as in NERVA, thus effecting both a neutronic and weight saving in the smaller reactor. The engine schematic for this system is shown in Fig. A.1.

The support-element heat pickup, including the pickup by regeneratively cooled beryllium slats, was calculated for a 300-MW reactor having a 2:1 core pattern with half the propellant flow used for support cooling. The calculations indicated a coolant outlet temperature of 850°R and a pressure drop of 100 psi for the support system. Cycle pressure and temperature calculations were made assuming a combined turbopump efficiency of 40%, a turbine bypass flow of 10% at full power, and a turbine outlet pressure of 650 psi. The results of these calculations, shown in Fig. A.2, indicate that a pump discharge pressure between 1100 and 1200 psi should provide comfortable margins of pressure drop for cooling both the support system and the nozzle.

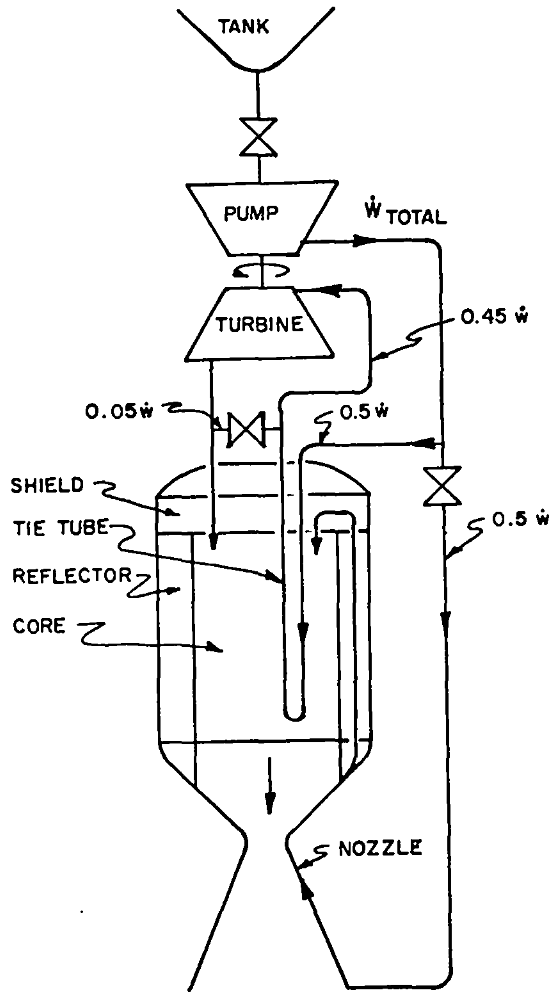

Fig. A.1. Engine cycle schematic.

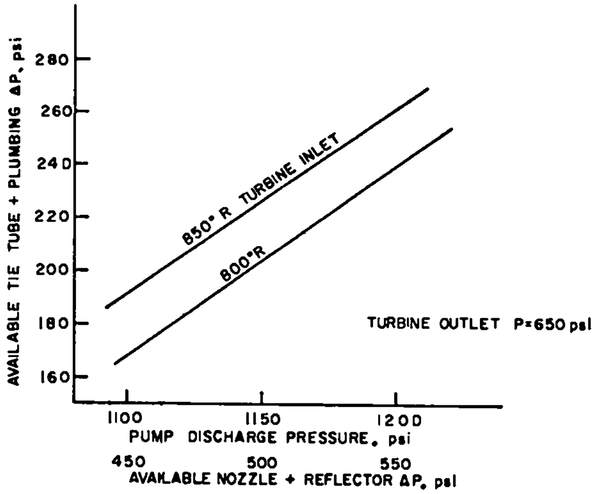

Fig. A.2. System pressures and pressure drops.

Weight estimates were made for a 300-MW engine having the following major parameters:

Power, MW	300
Chamber temperature, °R	4500
Chamber pressure, psia	400
Pump discharge pressure, psia	1150
Flow rate, lb/sec	17
Core pattern, (fuel element/support element)	2:1
Core length, in.	52

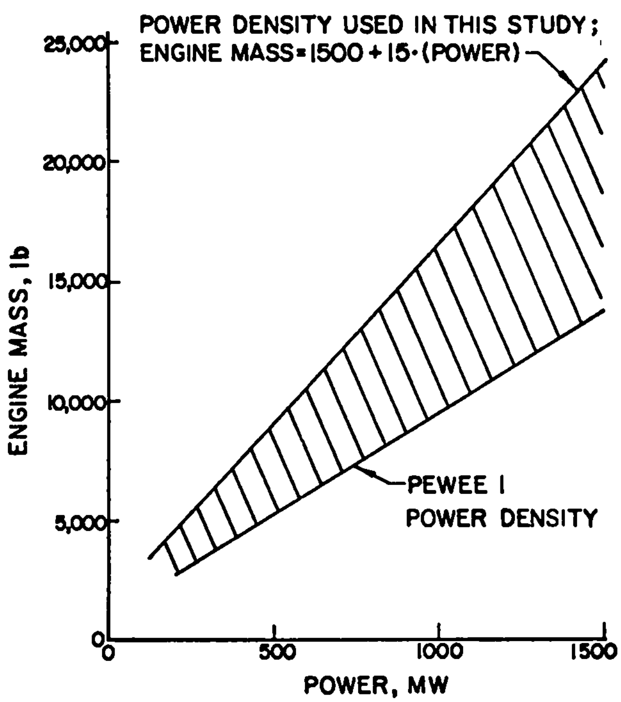

Fig. A.3. Estimated engine mass versus power.

Two insulating materials were considered: pyrographite and non-corroding, low-density ZrC. The resulting engine weights were 6000 and 6250 lb, respectively. A detailed weight breakdown is given in Table A-I.

TABLE A-I

SMALL ENGINE WEIGHT ESTIMATES
300 MW, 2:1 CORE PATTERN

	Engine Weight, lb	
	Pyrographite Insulators	Porous ZrC Insulators
Reactor		
Core	1410	same
Periphery	205	same
Reflector	1020	1255
Other hardware	285	same
Pressure vessel	240	255
Nozzle and skirt	500	same
Shield	915	same
Feed system	600*	same
Thrust structure and gimbal	300*	same
Actuators, instrumentation, and control	525*	same
	6000	6250

*Estimates supplied by SNSO.

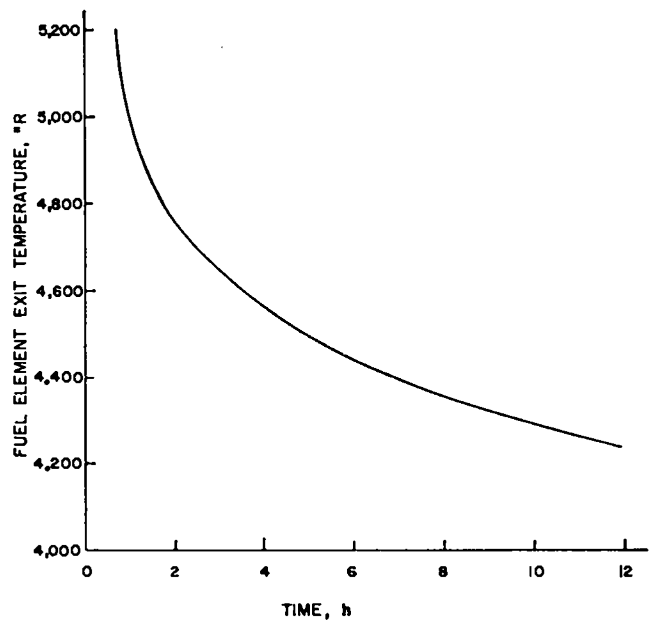

Fig. A.4. Estimated fuel-element time-versus-temperature performance.

The estimated dependence of engine weight on power is shown in Fig. A.3 for a range of fuel-element power densities. The estimated relationship of operating time to fuel-element exit temperature is shown in Fig. A.4. Indications are that a chamber temperature of 4800°R, which corresponds to a specific impulse of ~ 890 sec, is attainable for an operating time of 1 h.

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