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AEC RESEARCH AND DEVELOPMENT REPORT

# LOS ALAMOS SCIENTIFIC LABORATORY

OF THE UNIVERSITY OF CALIFORNIA ○ LOS ALAMOS NEW MEXICO

## MISSION STUDIES FOR NUCLEAR HEAT EXCHANGER ROCKETS

(Title Unclassified)



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# MISSION STUDIES FOR NUCLEAR HEAT EXCHANGER ROCKETS
(Title Unclassified)

by

Ralph S. Cooper

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By Marlene Lujan CIC-14 9-20-95



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## ABSTRACT

A study has been made of a wide variety of missions which might be performed by nuclear heat exchanger propelled rockets utilizing hydrogen propellant. The emphasis is on an early level of technology, such as might be expected from the Rover Project during the 1960's. Payload capabilities have been computed for single and two stage all nuclear vehicles, nuclear upper stages on Atlas class and Saturn chemical boosters, and on larger specially designed chemical boosters. Recoverable and reusable nuclear boosters also have been examined.

While the results are too numerous to quote in detail, we can cite a few of the more interesting examples. Half million pound nuclear single stage rockets are capable of placing ~50,000 pounds in low earth orbits. Orbital start vehicles of this weight allow low (~.2) values of thrust/initial weight and are capable of sending probes of the order of 10,000 pounds throughout the solar system. Manned exploration of the moon appears feasible with two stage nuclear rockets weighing about $10^6$ pounds. Lightweight reactors (of ~1000 Mw) would make 50,000 pound nuclear second stages for Atlas class boosters very attractive, while the conventional nuclear engines are more suitable to boosting by Saturn or a large nuclear stage.

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Introduction

The purpose of this paper is to discuss the capabilities and limitations of nuclear rockets for the technology which may exist in the 1960's from a natural development of the existing Rover program. Naturally, the more advanced goals of the program will influence its earlier phases so these will be considered at appropriate points. We first will present the generalized performance range for early Rover vehicles and then discuss in some detail some specific missions.

In the next decade, the exit gas temperature of operational solid core nuclear heat exchangers may be expected to be at least 2000$^{o}$C, corresponding to vacuum specific impulses ($I_{SP}$) of 780 seconds. Thus, the exhaust gas velocity ($v_e$) is comparable to the ideal velocity required for earth satellites (~27,000 ft/sec). Nuclear vehicles will have greater dry weight fractions owing to the larger engine weight and the greater tank fraction required by the low density of hydrogen. Perhaps the greatest uncertainty in the capabilities of early vehicles comes from uncertainties in the component weights rather than from the value of $I_{SP}$ expected. The state of knowledge concerning dry weights, combined with the low mass ratios allowed by the high specific impulse, leads to the conclusion that very simple methods are adequate at the present time for



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study of nuclear rocket missions. Thus, much of our work has been based on simple energy considerations corrected for losses (gravity, turbine, nozzle, atmospheric drag and pressure, etc.) by more exact methods.

## General Performance Analysis

Identifying various masses by subscripts $\ell$ for payload, b for burnout, t for tanks, o for total initial, p for propellant, and e for engine and associated miscellaneous items, we have

$$M_\ell = M_b - M_e - M_t. \tag{1}$$

And letting

$$\epsilon = \frac{M_e}{M_o} \qquad \text{(engine and miscellaneous weight fraction)}$$

and

$$f = \frac{M_t}{M_p} \qquad \text{(tank fraction)}$$

leads to

$$y = \frac{M_\ell}{M_o} = \frac{(1 + f)}{R} - (\epsilon + f), \tag{2}$$

where y is the payload fraction and R is the mass ratio,

$$R = e^{\Delta V/v_e}.$$



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This easily generalizes for n stages to

$$y_n = \frac{M_\ell^{(n)}}{M_o} = \prod_{i=1}^{n} \left[ \frac{1 + f_i}{R_i} - (\epsilon_i + f_i) \right]. \tag{3}$$

For $v_i$ and $f_i$ independent of i, $y_n$ is maximized for $R_i(\epsilon_i + f_i)$ equal for all i. Usually for cases of interest, $y_n$ is not very sensitive to the optimization.

With f, $\epsilon$, and $v_e$ dependent upon i, the payload is maximized if the quantity*

$$\frac{(1 + f_i)}{v_e^i \left[ (1 + f_i) - (\epsilon_i + f_i) R_i \right]} \tag{4}$$

is equal for all i.

With $\epsilon$, f, and $v_e$ independent of i, the optimum payload is obtained with all $R_i$ equal and thus equal velocity increments for each stage. Under this simple condition, we might seek some limiting case which eliminates staging effects also. Infinite staging leads to the result

$$\lim_{n \to \infty} y_n = \lim_{n \to \infty} (1 - \epsilon)^n \, e^{-\frac{1+f}{1-\epsilon} \frac{\Delta V}{v_e}}, \tag{5}$$

which is zero for non-zero $\epsilon$. A better method is to optimize staging considering n as continuous.

*Derived by Blair Schwartz, private communication.



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$$\frac{\partial y_n}{\partial n} = \frac{\partial}{\partial n}\left[(1 + f)e^{-\frac{\Delta V}{n v_e}} - \epsilon - f\right]^n = 0. \tag{6}$$

This can be solved by making judicious approximations (Appendix) to yield the result

$$y = \left[\frac{1 + f}{e(1 - b)} - f - \epsilon\right]^{(1+b)\frac{\Delta V}{v_e}}, \tag{7}$$

where

$$b = \frac{f}{e(f + \epsilon)}$$

$$e = 2.718 \ldots .$$

This is of the form of

$$\ell\, ny = -k\,\frac{\Delta V}{v_e} .$$

We can make an approximate evaluation of k in terms of f and $\epsilon$ to obtain

$$y_{max} \approx e^{-(1+1.42f+2.9\epsilon)\frac{\Delta V}{v_e}} \tag{8}$$

for optimum staging.

The equation (2) for a single stage is valuable for determining exchange ratios (the value of one parameter in terms of payload or other parameters). Of greatest interest is the worth of specific impulse $I_{SP}$ (= $v_e/g$).



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$$\frac{dy}{dI_{SP}} = \frac{d}{dI_{SP}}\left[(1 + f)e^{-\frac{\Delta V}{gI_{SP}}} - \epsilon - f\right]$$

$$= (1 + f)e^{-\frac{\Delta V}{gI_{SP}}}\left(\frac{\Delta V}{gI_{SP}^2}\right) \tag{9}$$

letting $x = \frac{\Delta V}{gI_{SP}}$

$$\frac{dy}{dI_{SP}} = \frac{(1 + f)}{I_{SP}} xe^{-x}. \tag{10}$$

Since $xe^{-x}$ is a slowly varying function ($\simeq e^{-1}$) in the range of interest (15,000 ft/sec $\leq \Delta V \leq$ 40,000 ft/sec or $.5 \leq x \leq 1.5$), we can make a very simple approximate generalization ($f \ll 1$ in practice also),

$$\frac{\Delta M_L}{M_o} \approx \frac{M_o}{e} \frac{\Delta I_{SP}}{I_{SP}} . \tag{11}$$

Where no dissociation occurs in $H_2$

$$I_{SP} \propto T^{1/2} \qquad \text{(T being absolute temperature),}$$

thus

$$\frac{\Delta M_L}{M_o} \approx \frac{M_o}{2e} \frac{\Delta T}{T} . \tag{12}$$

Returning to the original equation (2) and computing the effect of changing the velocity requirement, we have



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$$\frac{\Delta M_L}{M_o} = \frac{(1 + f)}{R} \frac{\Delta(\Delta V)}{v_e} , \tag{13}$$

and for component weights

$$\Delta M_L = M_o(\frac{1}{R} - 1)\Delta f$$

$$= -M_p \Delta f = -\Delta M_t \tag{14}$$

and

$$\Delta M_L = -M_o \Delta\epsilon = -\Delta M_e . \tag{15}$$

The problem of choosing component weights is difficult and the crucial one for such studies; but for definiteness, we will pick a single set for ground launched vehicles and one for upper stages. They differ because relatively smaller engines are required for upper stages as a first nuclear stage usually will have a burnout velocity near orbital speed. Our assumptions are listed in Table 1, and we will examine the effect of variation of the parameters later.

Table 1

COMPONENT WEIGHTS AND PERFORMANCE PARAMETERS

| | NUCLEAR | | CHEMICAL | |
|---|---|---|---|---|
| | First Stage | Upper Stages | LOX-$H_2$ | LOX-RP |
| Tank fraction $M_t/M_p = f$ | .07 | .07 | .04 | .02 |
| Miscellaneous weight fraction | .02 | .02 | .015 | .01 |
| Engine weight fraction | .06 | .03 | .015 | .01 |
| Engine fraction $\epsilon$ (including misc. wt.) | .08 | .05 | .03 | .02 |
| $I_{SP}$ (VAC) | 760-900 | 760-900 | 430 | 300 |
| Fuel density lbs/ft$^3$ | 4.34 | 4.34 | 17.3 | 63 |



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The results for one and two stage nuclear rockets for a range of $I_{SP}$ are given in Figure 1, along with the performance of some chemical rockets for comparison. We see that on a gross weight or thrust basis, the nuclear rockets are generally superior.

Note that as the mission velocity requirement increases, the higher $I_{SP}$ of the nuclear system causes it to become increasingly superior to the chemical systems on a gross weight per payload basis. It can be seen from Eq. (8) and the assumed values of the parameters that the gross weight ratio for a fixed payload of a LOX-$H_2$ compared to nuclear propulsion system will be

$$\frac{M_o^{CHEM}}{M_o^{NUC}} = e^{+\frac{\Delta V}{8.0\ \text{km/sec}}} = e^{\frac{\Delta V}{26,000\ \text{ft/sec}}} . \tag{16}$$

Gross weight is an important factor in the case of upper stages on existing boosters, or if the vehicle is to be first placed in orbit. However, approximate cost comparisons are frequently based on the dry (manufactured) weight of the vehicles. These will naturally be much more sensitive to the choice of values for $\epsilon$ and f and, recognizing the great uncertainty in these values, an adequate approximation for the dry weight is simply

$$M_d \simeq (\epsilon + f)M_o . \tag{17}$$

Combining this with Eq. (5), we get the results of Figure 2. This is still a crude comparison for costs, since there are no developmental or ground operational costs, and nuclear hardware may cost more per pound



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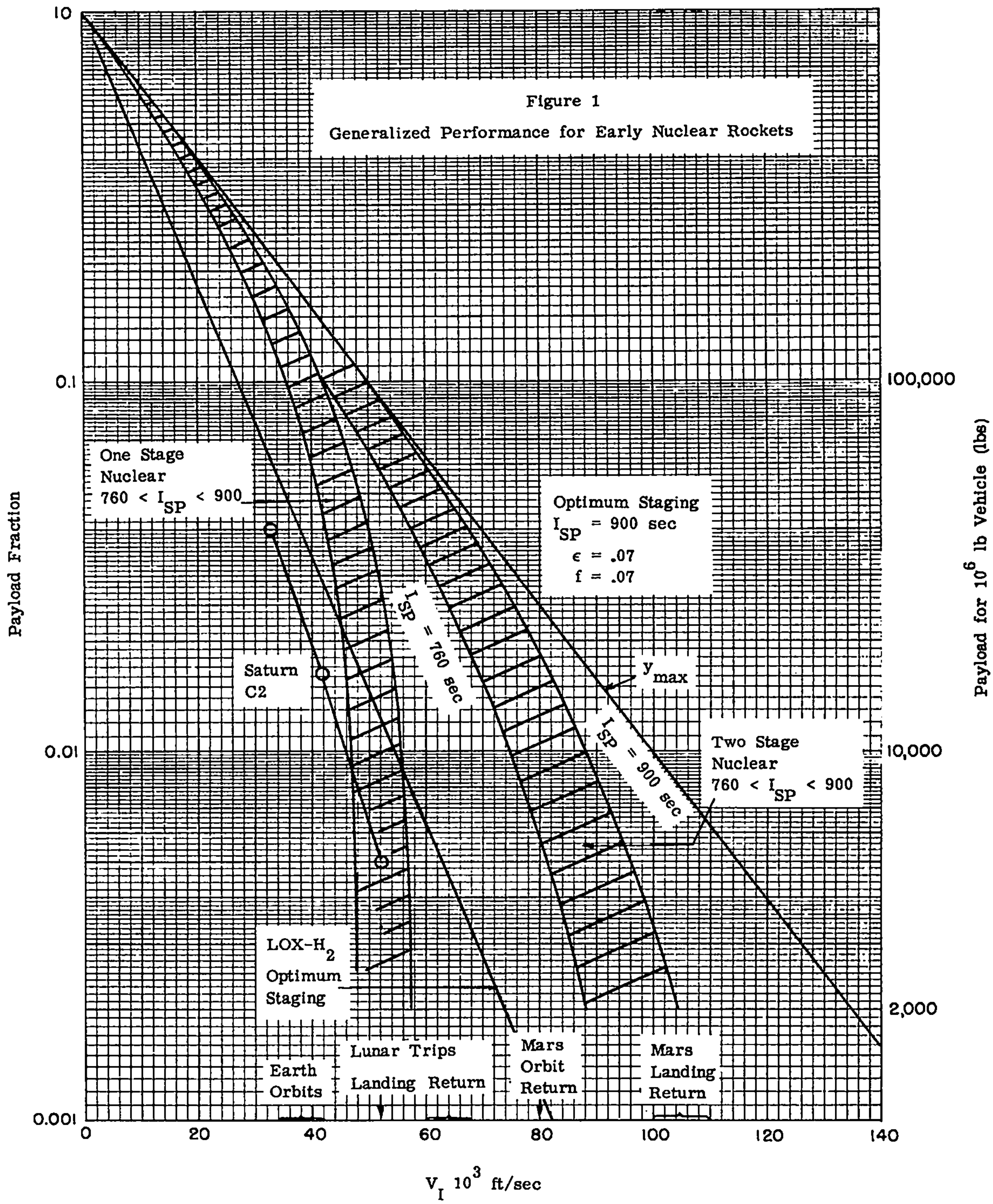


Figure 1
Generalized Performance for Early Nuclear Rockets



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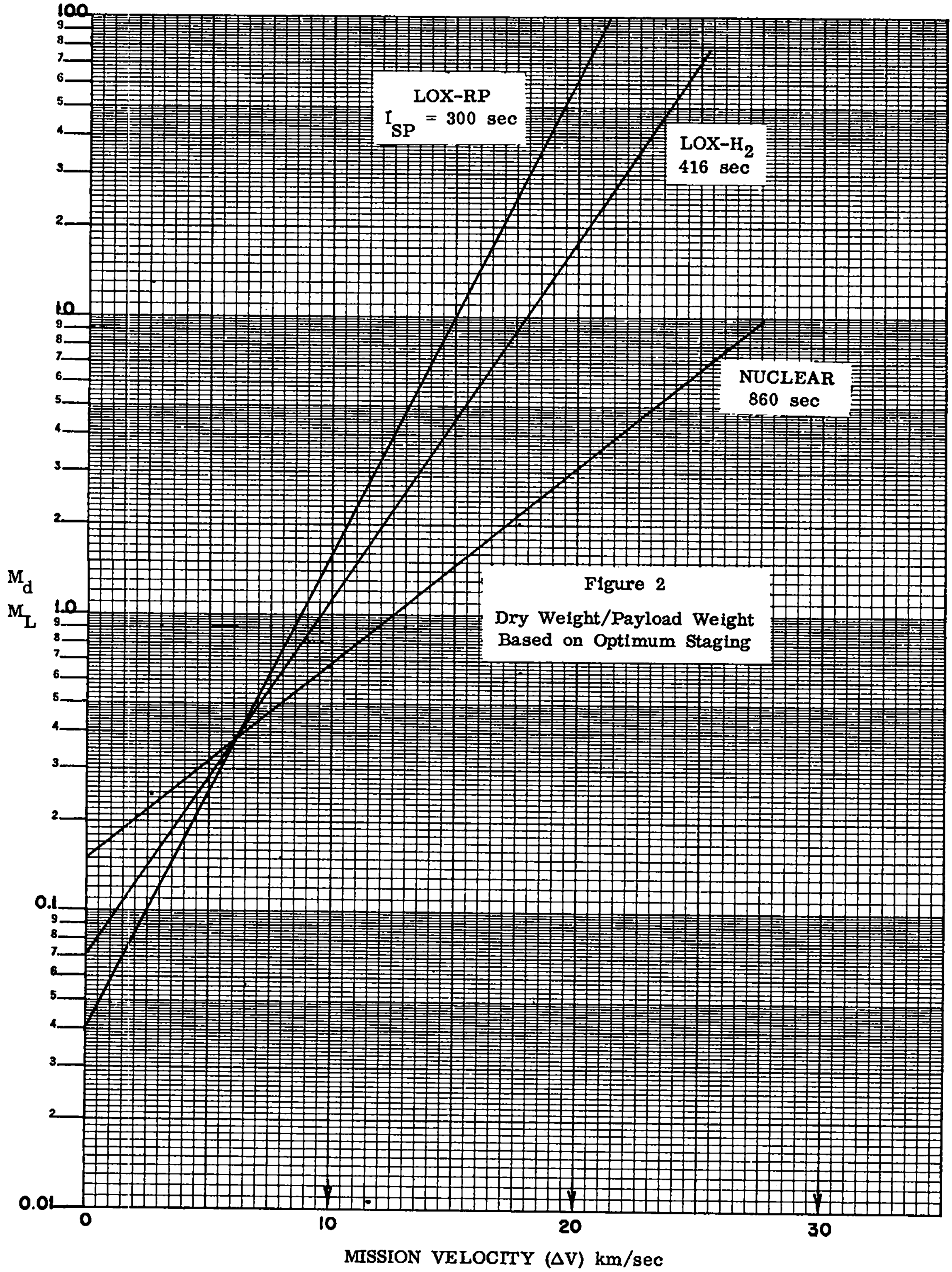


Figure 2

Dry Weight/Payload Weight Based on Optimum Staging



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than conventional engines. Still, the higher $I_{SP}$ always is best for sufficiently large $\Delta V$. There is little else in sight for high thrust applications other than chemical and nuclear systems and the Orion scheme for using nuclear explosions. "Exotic" chemicals such as $F_2$-$N_2H_4$ or nuclear-$NH_3$ systems might offer advantages in size, dry weight, etc., but they are presently further from the operational stage than their counterparts discussed here. Where milli-g accelerations are permissible, higher impulse electrical propulsion (plasma or ion jets) will be competitive only when low weight electrical power systems are developed (presumably based on nuclear energy also). However, for manned interplanetary trips, the desire for quick passage through the Van Allen belts and high thrust for landing favors the direct nuclear propulsion scheme, though it is too early to decide the issue conclusively.

Assuming a fixed engine fraction as we have done applies only for sufficiently large engines, as there is a lower limit to reactor weight as determined by criticality. Furthermore, the effect of altitude upon thrust and the acceptable thrust/initial weight ratio affect the value of $\epsilon$ slightly. The important parameter is the ratio of engine weight to power, which we shall label $\mu$ (in pounds/megawatt). With $H_2$ at temperatures of interest, one megawatt of power will deliver ~45 pounds of thrust in vacuum (~40 lbs at sea level). Thus the engine fraction (.06) which we have chosen for first stages corresponds to about 2 lbs/Mw, assuming an initial thrust to weight of about 1.2 for the vehicle. Let us



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now examine the existing reactor designs and concepts to define the region of applicability of the foregoing generalizations.

Existing Engine Designs and Concepts

The only concept in the hardware stage is Kiwi, which is expected to lead to an engine in the 1000-1400 Mw power range (40,000 to 60,000 lbs thrust). Its weight is estimated to be about 7000 pounds, and therefore the engine will have a high specific weight of 5 to 7 lbs/Mw. At this power level, the Kiwi concept is basically neutronically limited (as opposed to heat transfer limited), and at larger powers (~10,000 Mw) values of 2 lbs/Mw seem feasible.

A new reactor core concept, Phoebus, is undergoing design studies which indicate that it can produce about three times the Kiwi power in the same engine size and engine weight, i.e., 3000 to 4000 Mw (120,000 to 160,000 lbs thrust). This would be about 2 lbs/Mw, which might not be reduced much on increasing the power level. Thus either Kiwi or Phoebus might be extrapolated to a large size engine (≥ 10,000 Mw, 400,000 lbs thrust) at ~2 lbs/Mw. This large engine has been named Condor, although there are no designs at present. Considering the reactor only, power densities as high as 5 Mw/lb (0.2 lbs/Mw) seem possible, so there is considerable room for eventual improvement.

Some effort has been made to reduce the minimum weight (of ~7000 lbs for Kiwi engines) and to achieve higher power densities in low power engines (100-1000 Mw). At Los Alamos, work is underway on two reactor



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concepts (Roc) which may lead to engines in the 100-1000 Mw class weighing from 500 to 4000 pounds.[1] Other laboratories (UCRL-Livermore and NASA-Lewis[2]) have considered other low power heat exchangers and their applications. While it would be premature to state specifications even approximately, we will occasionally examine the usefulness of hypothetical 1000 Mw engines weighing 2000 and 4000 pounds, and a 200 Mw engine weighing 1000 pounds, for the purpose of establishing the worth of effort in this direction. Very large engines (> 10,000 Mw) have received little practical effort thus far (although they have obvious applications), mainly since they represent a great extrapolation. We shall assume 2 lbs/Mw for such engines.

We can make a general comparison of Kiwi and Phoebus powered stages with a LOX-$H_2$ chemical stage. Assuming a $T/W_o$ (thrust/initial weight) of 1.0 for the nuclear stages, since they will be assumed to be upper stages, we find that a Kiwi powered 60,000 pound stage is only marginally better than a chemical stage on a gross weight basis. A 120,000 pound stage, using a higher powered Kiwi or a Phoebus engine, is markedly superior to a chemically propelled stage. Thus, 60,000 pounds is a break-even point in stage weight above which Kiwi nuclear propulsion is superior. Naturally, should the lighter weight reactors prove feasible, this break-even point will become lower. Assumptions are listed in Table 2, and results shown in Figure 3 as payload vs. stage velocity increment. Note that the dry weights represent quite a large percentage (27% and 19%) for the nuclear stages, which results largely from the large minimum weight of Kiwi



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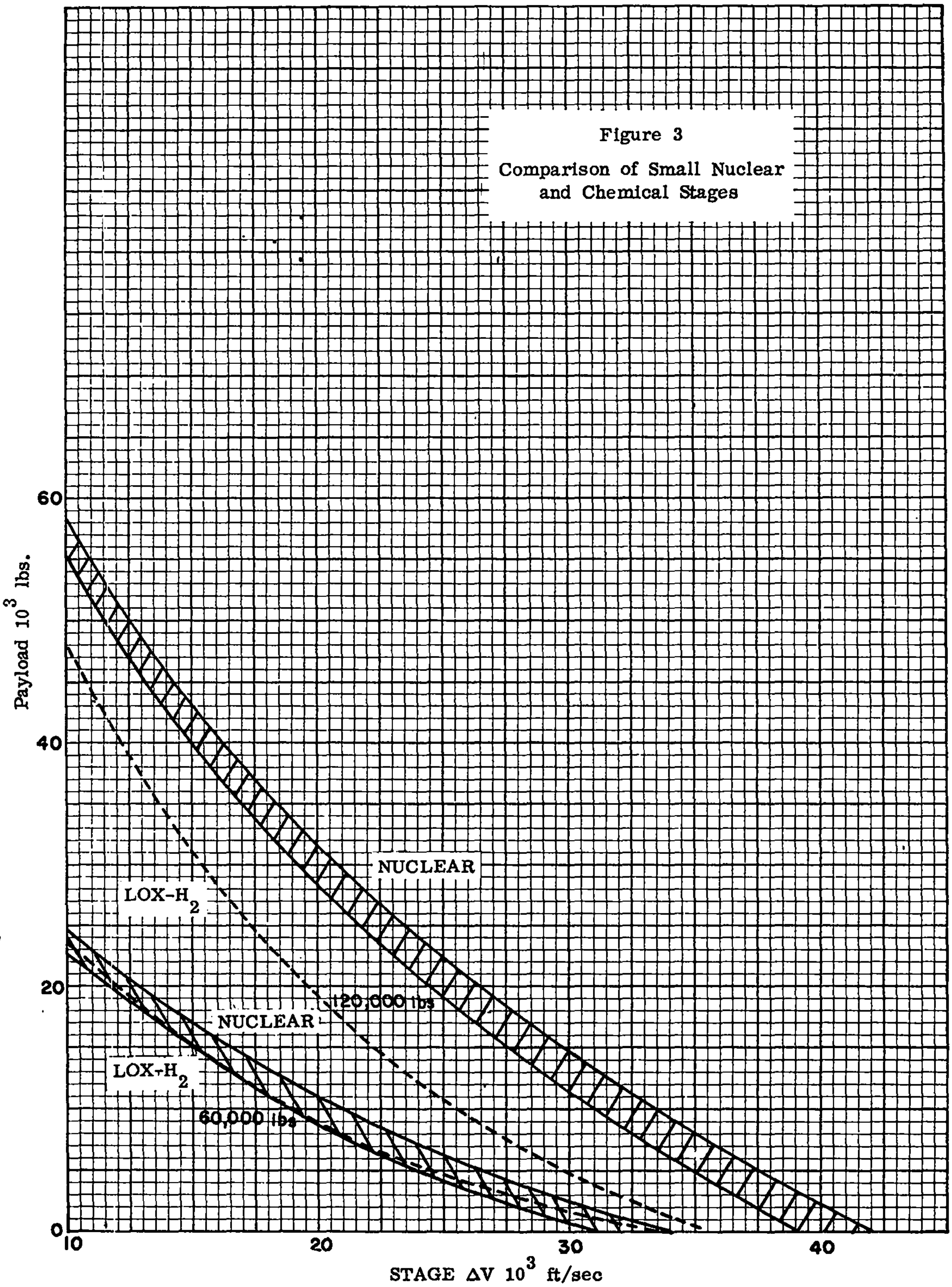




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engines and the relatively large tank fraction (~.15) for small $H_2$ tanks.

Table 2

SMALL STAGES

| | | |
|---|---|---|
| Total Weight | 60,000 lbs | 120,000 lbs |
| Nuclear propulsion 2150°C gas | Kiwi | Enlarged Kiwi or Phoebus |
| Power ($I_{SP}$ = 800 sec) | 1400 Mw | 3000 Mw |
| Engine weight | 8-9000 lbs | 10,000-12,000 lbs |
| Misc. equipment | 3000 | 4000 |
| Tanks | 5-6000 | 9000-10,000 |
| TOTAL DEAD WEIGHT | 16-18,000 lbs | 23,000-26,000 lbs |
| Chemical propulsion | | |
| LOX-$H_2$ $I_{SP}$ = 430 | | |
| TOTAL DEAD WEIGHT | 5000 lbs | 9000 lbs |

Very little improvement can be expected either in component weight or $I_{SP}$ for the chemical system. The nuclear stages assume a very early state of the art and are capable of considerable improvement. The assumed gas temperature (~2150°C) seems reasonable for an early engine, but the payload is not very sensitive to this parameter, changing 300 lbs/100°C for the 60,000 pound stage. This makes a difference of ± 1000 pounds for the range of temperatures which might be expected for an early engine.



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At the 120,000 pound stage level, there is considerable difference in the two propulsion systems, ranging from 6000 to 12,000 pounds in payload or 3000 to 6500 ft/sec in velocity increment. This difference exists independent of staging effects, as the additional (presumably chemical) stage can be placed on the nuclear stage as well as on the chemical one. The thrust to weight ratio for upper stages which must do much work against gravity will probably lie in the range of .8 to 1.2. For stages already in orbit or with largely horizontal trajectories, the power level can be further reduced, which is frequently helpful for nuclear stages where the engine weight can be reduced with power level.

Single Stage Nuclear Rockets

As can be seen from Figure 1, ground launched nuclear single stages can carry payload fractions of 0.01 or more up to ideal velocities of about 50,000 ft/sec, sufficient for soft lunar landing missions. However, because of fixed minimum component weights, especially for the smaller reactors and possibly tanks as well, there is a minimum vehicle size, at least for early vehicles, to which Figure 1 applies. The Kiwi engine, operating at 1400 Mw (~60,000 lbs thrust) is not in this class and, in particular, is not capable of placing itself in orbit from the ground. The maximum capability of the Kiwi engine (at 5 lbs/Mw) in a specially designed vehicle would be to carry the vehicle to about a 1000 mile altitude with negligible payload. Alternatively, for



[redacted]

demonstration or test purposes, one could use standard ICBM tankage (which would hold 16,000 pounds of $H_2$) and fly to 50 to 200 mile altitudes.

A Kiwi type engine of higher power, 2000 Mw, in a larger vehicle (~75,000 lbs) might be capable of placing itself in orbit with 0 to 4000 pounds of payload, but this still is a marginal case. The Phoebus engine concept (2 lbs/Mw at 4000 Mw) could power a single stage (130,000 lb gross weight) into orbit with a 10,000 pound payload. Its dead weight fraction is low enough for it to reach escape velocity with a 1000 pound payload. Table 3 gives some details of the vehicle. This performance is equivalent to that of the Atlas-Centaur vehicle which has twice the gross weight but a smaller dry weight. Since the Phoebus powered vehicle performance can be duplicated by an existing chemical vehicle, it would not represent an end itself, but it could be valuable for gaining experience in ground launching techniques, testing, etc. This size is not attractive for a booster unless a high power density, low power upper stage nuclear engine exists. Thus engines of power level up to 4000 Mw seem best suited to upper stage applications.

Perhaps the smallest size of practical interest for a nuclear booster is the half million pound thrust level. A series of accurate trajectory studies[3] (including gravitational and atmospheric effects, etc.) indicated low orbit payloads of the order of 50,000 pounds for single stage ground launched vehicles of ~400,000 pound take-off weight (~12,000 Mw). This is equivalent to the three chemical stage, $1.5 \times 10^6$ pound thrust Saturn



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Table 3

A PHOEBUS POWERED SINGLE STAGE ROCKET VEHICLE CONFIGURATION FOR LOW ORBIT MISSION

| | |
|---|---|
| Stage weight | 130,000 lbs |
| Reactor power | 4,000 Mw |
| Exit gas temperature | 2,170°C |
| $I_{SP}$ (VAC) | 807 sec |
| Component weights | |
| Reactor + pressure shell | 8,000 lbs |
| Turbopump + plumbing | 1,200 |
| Nozzle | 700 |
| Tanks (~8% of $H_2$) | 7,600 |
| Structure | 2,000 |
| Holdover (propellant) | 1,000 |
| Total dead weight (15.8% $W_o$) | 20,500 |
| Weight at burnout (300 miles) | 33,200 |
| Payload (fixed earth) | 12,700 |
| Payload (rotating earth) | 14,500 |
| Thrust/initial weight | 1.35 |
| $\dot{\omega}$ - $H_2$ flow rate | 256 lbs/sec |
| Tank diameter L/D = 6 | 17.6 |
| Tank length (including endcaps) | 125 ft |
| Missile length (including payload and engine) | ~150 ft |

| Orbit altitude | Payload (± 1000 lbs) | |
|---|---|---|
| | Fixed earth | Rotating earth |
| 300 mi | 12,700 lbs | 14,500 lbs |
| 1000 | 10,000 | 11,600 |
| 5000 | 2,300 | 3,600 |
| ∞ | 800 | 1,900 |



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rocket performance. Above ideal velocities of 40,000 ft/sec, the payload fraction decreases rapidly and is very sensitive to the component weights.

Since single stage vehicles are of interest due to their inherent simplicity and possible reusability, it is worth briefly examining more advanced technology to determine the limitations of solid core nuclear heat exchanger rockets, so as to evaluate them as a goal affecting the direction of the program. For this type of propulsion system (including a high thrust requirement), an estimate indicates the ultimate specific impulse would be about 1290 sec (3500°C $H_2$, dissociation effects included) and that 1075 sec is a more likely practical limit for a recyclable, reusable reactor. An engine weight of 3% of gross take-off weight seems to be the limit, plus 1/2% for miscellaneous structure, landing and re-entry devices, etc. 5% to 6% appears optimistic for metal propellant ($H_2$) tanks which are capable of sustaining re-entry and landing maneuvers. These assumptions lead to the performance curves shown in Figure 4 for "intermediate" and "ultimate" technologies. The maximum "range" (in terms of ideal velocity) is given by

$$\Delta V_{max} = gI_{SP} \ln\left(\frac{f + \epsilon}{1 + f}\right). \tag{18}$$

The "early" technology rocket is limited to ~50,000 ft/sec (with a 5% payload), sufficient only for round trips to earth or lunar orbits with atmospheric braking upon re-entry and for one-way trips to the lunar surface. For the "intermediate" technology (which might be



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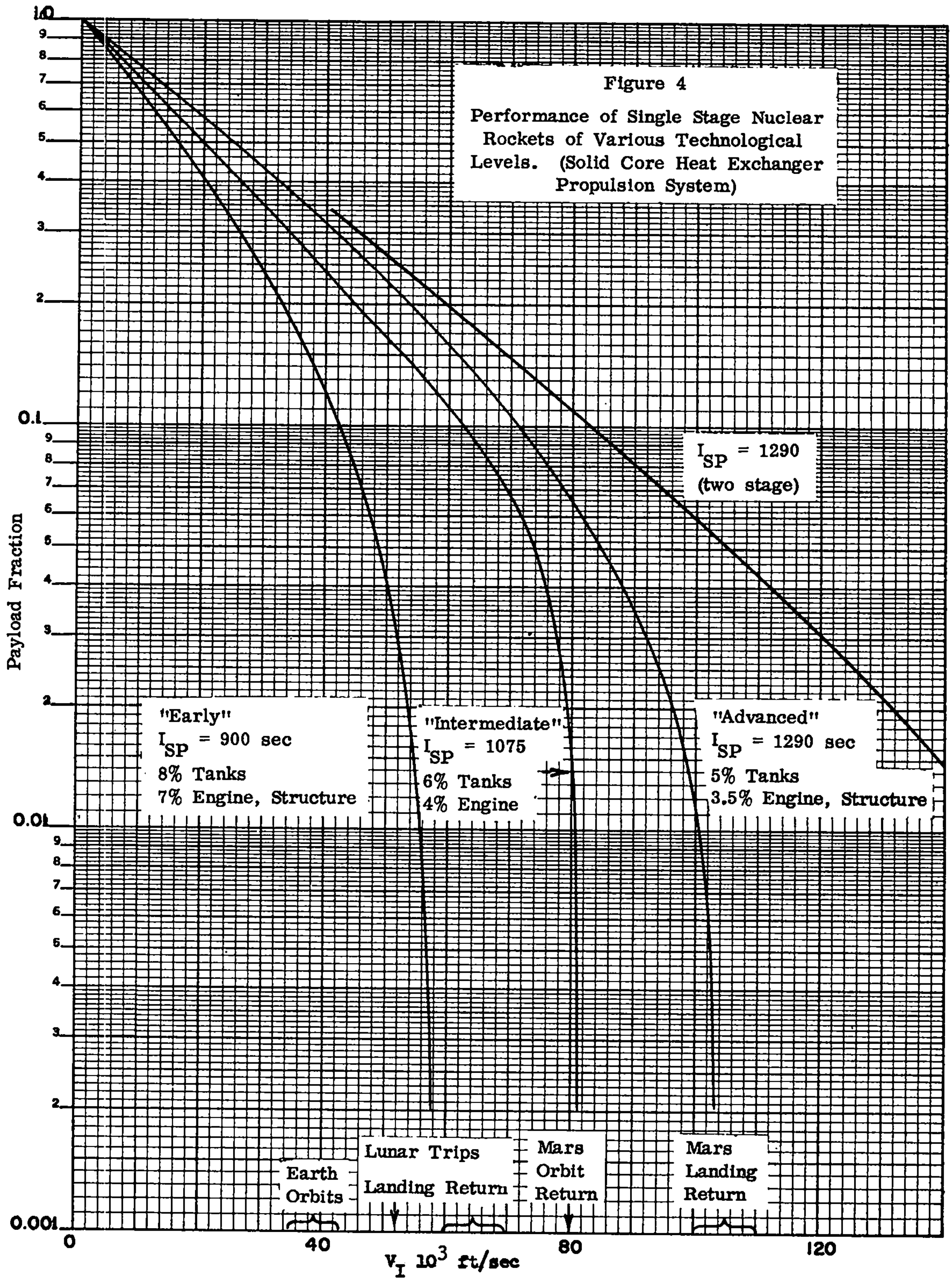


Figure 4

Performance of Single Stage Nuclear Rockets of Various Technological Levels. (Solid Core Heat Exchanger Propulsion System)



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developable in this decade with a very vigorous program), round trips to the moon or to Martian satellites are possible but not to Mars' surface. Minimum energy Martian expeditions appear to be the limit for reusable single stage, solid core, nuclear rockets. One additional nuclear stage (and even merely dropping fuel tanks) makes most solar system voyages practical with payload fractions of .1% or more.

Since even the early models of single stage nuclear rockets can place themselves and considerable payload in orbit, this opens the possibility for a reusable booster which places itself and its payload in earth orbit in which separation may take place at leisure, whereupon the booster is safely returned to earth under a combination of atmospheric and powered braking. (Considerably larger payloads can, of course, be delivered by having the booster burnout at lower than orbital velocity and using a second stage engine to place the payload in orbit; but the simplicity of the first scheme favors it for early developmental work, and the margin of safety it provides may make its value enduring, especially with higher performance vehicles.)

Let us immediately see what might be done with current nuclear technology, an average $I_{SP} = 800$, 7.5% $H_2$ tanks, a 5% engine, and liberal structure to allow for re-entry and landing. For definiteness, we will consider a 500,000 pound vehicle, though weights are linearly scalable at this level. The engine required will deliver 12,000 Mw (600,000 pound sea level thrust), and we assume 25,000 pounds (2 pounds/Mw) for it. We temporarily assume the weight of tanks and structure to be 10% of the

[redacted]

gross take-off weight and will compute them separately later on the basis of 7.5% of the propellant weight for the tanks. Thus, the total dead weight ($M_d$) is 15% of the gross weight $M_o$.

The first step is to place the booster and payload in a low earth orbit, which requires 27,000 ft/sec + 6,000 ft/sec as a reasonable allowance for gravitational and drag losses. The burnout weight is then

$$M_b = M_o e^{-\frac{33,000}{25,600}} = .276\ M_o = 138,000 \text{ pounds.}$$

Subtracting the dead weight leaves the sum of the payload ($M_\ell$) and the reserve propellant $M_{rp}$ required for return of the booster.

$$M_\ell + M_{rp} = M_b - M_d = .276\ M_o - .15\ M_o = .126\ M_o = 63,000 \text{ pounds.}$$

The next step is to return the booster to earth for which we must assume some aerodynamic braking and, let us say, 5000 ft/sec of powered deceleration. For this phase,

$$M_{rp} = (R_2 - 1)M_d = e^{\frac{5,000}{25,600}}\ M_d = (.216)M_d = 17,000 \text{ pounds}$$

and

$$M_\ell = M_b - M_d - M_{rp} = 46,000 \text{ pounds.}$$

Thus, the payload in orbit is 46,000 pounds, about 10% of the gross vehicle weight.

The total propellant weight is

$$M_p = 362,000 \text{ pounds} + 17,000 \text{ pounds} = 379,000 \text{ pounds.}$$



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This would occupy 88,000 $ft^3$ and fill a cylindrical tank 29 feet in diameter and 145 feet tall (L/D = 5).

The tank mass is

$$M_t = .075\ M_p = 28{,}000 \text{ pounds,}$$

leaving structure and miscellaneous weight of

$$M_s = M_d - M_e - M_t = 22{,}000 \text{ pounds.}$$

This is 4.4% of the gross take-off weight and 29% of the dry booster weight and should be more than ample for re-entry devices. Improving the art increases the payload considerably as can be seen from Table 4.

Table 4

RECOVERABLE ORBITING BOOSTER

| Technology | Early | Intermediate |
|---|---|---|
| Specific Impulse | 800 sec | 1,000 sec |
| Gross Take-off Weight | 500,000 pounds | 500,000 pounds |
| Payload (low earth orbit) | 46,000 | 110,000 |
| Propellant | 379,000 | 330,000 |
| Tanks | 28,000 | 20,000 |
| Structure + Re-entry Devices | 22,000 | 20,000 |
| Engine (12,000 Mw) | 25,000 | 20,000 |

Should we desire to leave the large $H_2$ tank in orbit as part of a space station or for refueling and return to earth with propellant in a small tank, less structure (say ~10,000 pounds) and return propellant



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(8000 pounds) would be required. As much as 70,000 pounds of payload plus 25,000 pounds of tankage could then be placed in orbit. The resultant reusable portion of the vehicle would be easily transportable.

Alternatively, one could have a non-orbiting recoverable nuclear booster with a full second stage. We can briefly examine results for a typical non-orbiting booster which achieves only 10,000 ft/sec at burnout, requiring a second stage engine to supply about 16,000 ft/sec additional to orbit the payload (Table 5). Because of the lower final velocity for the booster, we assume only 3500 ft/sec of powered deceleration. The second stage when in orbit will include an engine which would be useful if further powered maneuvers are contemplated. On this basis, twice the payload of an orbiting booster can be placed in a low earth orbit at the expense of staging complications.

Table 5

RECOVERABLE, NON-ORBITING BOOSTER

$I_{SP}$ (avg) = 800, Booster Burnout at 10,000 ft/sec

| | | |
|---|---|---|
| | Gross Take-off Weight | 500,000 pounds |
| Booster | Propellant | 248,000 |
| | Tanks | 19,000 |
| | Structure (4%) | 21,000 |
| | Engine (12,000 Mw) | 25,000 |
| | Booster Payload = Second Stage Initial Weight | 187,000 |
| Second Stage | Propellant | 87,000 |
| | Tanks | 7,000 |
| | Structure (4%) | 8,000 |
| | Engine (5000 Mw) | 10,000 |
| | Payload (low orbit) | 75,000 |
| Useful load (sum of above 3 items) | | 93,000 |



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## Two Stage Vehicles

The general capabilities have already been presented in Figure 1, and we can be more specific for cases of interest. 50,000 pounds can be orbited either by a single stage (12,000 Mw, 430,000 pounds) nuclear rocket or by the Saturn C2, and thus is a good value to consider for an orbital start vehicle. We will give results for various missions using chemical, Kiwi, and lightweight reactor propulsion (Table 6 and Figure 5). We will also consider a two stage nuclear rocket with suborbital startup of the second stage. While we cannot optimize the staging for all missions, we can pick a representative one and use it. For a 430,000 pound take-off weight where the first stage supplies 24,000 ft/sec of ideal velocity, a second stage of 124,000 pounds results. This stage has a residual mass of 90,000 pounds at orbital velocity, including a reusable engine (~3000 Mw is required) and 2500 pounds of now useless tankage. The payload of this vehicle for a series of missions is also presented in Table 6 and Figure 5. For orbital startup, the thrust/initial weight ratio may be of the order of 0.2 or lower without seriously degrading performance. Thus, power levels of 5 Mw/1000 pounds are suitable, and 1000 Mw reactors are sufficient for 200,000 pound stages.

The real forte of nuclear rockets is in meeting the large payload and high velocity requirements of manned space exploration. While these missions will probably begin in the 1970's, the long lead time required for development and reliability of the vehicles used will affect the



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Table 6

CAPABILITY OF UPPER STAGES

(Boosted by Saturn C2 or 430,000 lb Nuclear Booster)

| Mission | $\Delta V_I$ ft/sec (from orbit) | Payloads in $10^3$ lbs: $LOX\text{-}H_2$ | 1000 Mw Kiwi | Lightweight Nuclear 400 Mw | 3000 Mw Kiwi* or Phoebus |
|---|---|---|---|---|---|
| Moon Hit or Pass | 10,500 | 18.0 | 19.0 | 25.0 | 40.5 |
| Escape | 11,000 | 17.1 | 18.3 | 24.3 | 39.3 |
| Low Lunar Orbit or 24 Hr. Orbit | 13,000 | 14.0 | 15.7 | 21.7 | 34.6 |
| Soft Lunar Landing | 18,600 | 7.4 | 9.5 | 15.5 | 23.6 |
| Probes: | | | | | |
| Venus (Min. E) | 11,500 | 16.3 | 17.6 | 23.6 | 38.0 |
| Mars (Min. E) | 12,000 | 15.5 | 17.0 | 23.0 | 37.0 |
| Mercury Probe or Mars Satellite | 18,600 | 7.4 | 9.5 | 15.5 | 23.6 |
| Jupiter 2.8 Yrs. | 20,500 | 5.7 | 7.7 | 13.7 | 20.2 |
| Saturn 6 Yrs. | 24,000 | 3.4 | 4.7 | 10.7 | 14.6 |
| Solar Escape or Jupiter 1.2 Yrs. or Saturn 2.7 Yrs. or Direct Solar Probe, $18.5 \times 10^6$ mi. perihelion | 29,000 | 0.8 | 0.9 | 6.9 | 7.8 |
| Mars Satellite and Return | 32,200 | --- | --- | 5.0 | 4.0 |
| **Assumptions** | | | | | |
| $I_{SP}$ | | 416 sec | 800 sec | 800 sec | 800 sec |
| Engine Thrust | | 40,000 lbs | 50,000 lbs | 20,000 lbs | 150,000 lbs* |
| Engine Weight | | 1,000 lbs | 8,000 lbs | 2,000 lbs | 10,000 lbs |
| Misc. Dead Weight | | 2,500 | 4,500 | 4,500 | 5,000 |
| Tanks (Approx.) | | 1,500 | ~3,000 | ~3,000 | ~8,000 |
| Initial Stage Weight | | 50,000 | 50,000 | 50,000 | 124,000 |
| Start | | Orbital | Orbital | Orbital | Suborbital |

*Nuclearly boosted only.



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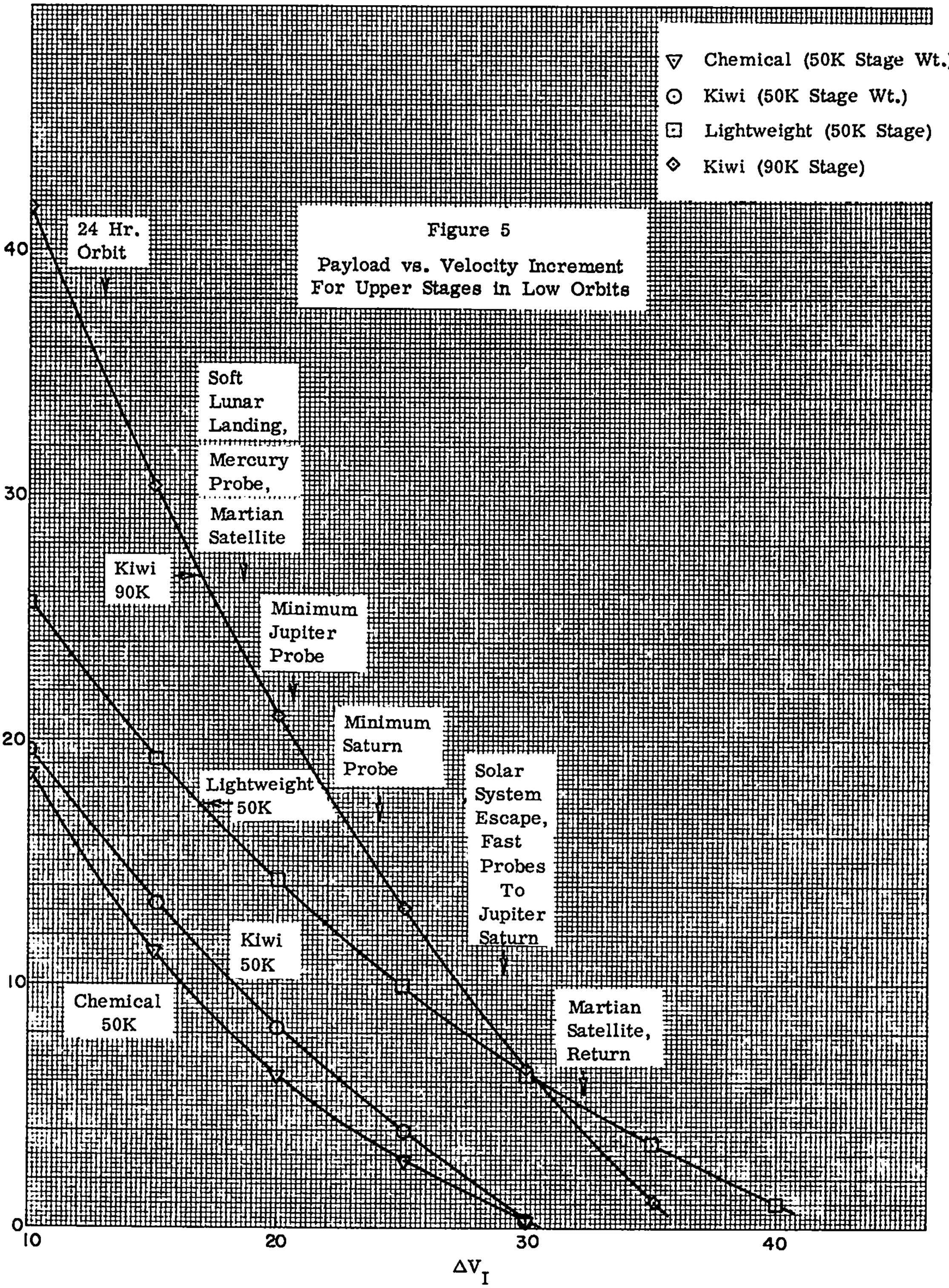


Figure 5

Payload vs. Velocity Increment For Upper Stages in Low Orbits



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early stages of the nuclear rocket program, and thus such mission studies are germane here.

Manned earth satellites and lunar circumnavigation missions can be accomplished with Saturn, and as minimum weight missions are probably too small and too early to have serious competition from nuclear rockets. Lunar exploration thus seems to be the first new mission appropriate for nuclear rockets.

The mission weight requirements are as yet poorly defined. In 1959, 8000 pounds was thought sufficient for a return capsule. A 6.7 million pound gross weight, four stage chemical rocket (NOVA), which could soft land a 36,000 pound payload on the moon, was contemplated as the vehicle.[4] Since then, however, the estimate for the capsule's weight has doubled and may increase further. We will, therefore, consider two stage nuclear rockets which can soft land various sized payloads on the moon. We assume the payload consists partially of a storable chemical return stage whose propellant also acts as a radiation shield. We see from Table 7 that the NOVA mission can be accomplished with two nuclear stages having about one-tenth the weight of a chemical rocket. It is clear that at least a 20,000 Mw reactor will be required for the booster engine. In a later section, we consider the effect of using a large (e.g., $3 \times 10^6$ pound thrust) chemical booster as a first stage with nuclear upper stages. The weights given in Table 7 may be scaled upward proportionally for larger payloads. Inner solar system reconnaissance missions without landing, such



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Table 7

TWO-STAGE NUCLEAR ROCKETS FOR LUNAR SOFT LANDING MISSION

| Technology | "Early" | "Intermediate" (Late 60's) |
|---|---|---|
| Gross Vehicle Weight | 955,000 lbs | 633,000 lbs |
| <u>1st Stage</u> | | |
| $I_{SP}$(VAC) (with 6% pump and nozzle losses) | 760 sec | 800 sec |
| Temperature | 2200°C | 2500°C |
| ΔV | 22,000 ft/sec | 22,000 ft/sec |
| Mass Ratio | 2.45 | 2.36 |
| Propellant Weight | 565,000 lbs | 365,000 lbs |
| Tank Weight (7% of $H_2$) | 40,000 | 25,000 |
| Structure and Misc. Wt. (3% of gross wt.) | 29,000 | 19,000 |
| Engine Weight (7% or 5% of gross wt.) | 66,000 | 32,000 |
| Engine Power | 24,000 Mw | 16,000 Mw |
| Thrust/Initial Weight | 1.25 | 1.25 |
| <u>2nd Stage</u> | | |
| $I_{SP}$ (6% pump and nozzle losses) | 760 sec | 860 sec |
| Temperature of Exit Gas | 2200°C | 2800°C |
| ΔV | 30,000 ft/sec | 30,000 ft/sec |
| Mass Ratio | 3.39 | 2.96 |
| Total Weight | 255,000 lbs | 192,000 lbs |
| Propellant Weight (+4000 lbs extra) | 184,000 | 131,000 |
| Tank Weight | 13,000 | 9,000 |
| Structure Wt. (Misc.) 5% or 3% of Stage Wt. | 12,000 | 6,000 |
| Engine Weight | 10,000 | 10,000 |
| Engine Power | 2,000 Mw | 2,000 Mw |
| Thrust/Initial Weight | .51 | .67 |
| Payload | 36,000 lbs* | 36,000 lbs* |

*By dropping a portion of the second stage tankage during the coast between the earth and the moon, one can increase both payloads to 38,000 pounds.



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as Hohmann transfer to Mars, are also appropriate missions for similar but larger two stage vehicles. For larger and more difficult missions, orbital assembly techniques will probably be preferable.

Chemically Boosted Nuclear Rockets

Since the early nuclear rockets will be relatively small in weight compared to existing or anticipated chemical boosters, it is natural to consider the early nuclear vehicles as upper stages. We do not feel that this situation is necessarily the logical one ultimately, but it does reflect the advanced state of the chemical propulsion art. It is also influenced by the maxim for chemical staging which declares that high energy fuels are more useful in upper stages and by many semi-technical and political considerations such as problems of ground take-off and of radioactive debris falling on populated areas. We shall not discuss these questions here except to comment that the most reasonable situation technologically in the long run appears to us to be relatively large nuclear rockets (one or two stages) with possibly a final small chemical stage which acts as a radiation shield, an atmospheric re-entry stage, and an accident escape system. While we do not see nuclear rockets as a justification for larger chemical boosters, we should make use of those already existing for the early development of useful nuclear stages. Chemical boosting alleviates the problems of air-scattered radiation to the launch facilities and payload, and many of the fallout problems. Studies have shown that in terms of useful payload, the best

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application of a nuclear engine of 4000 Mw or less is as an upper stage. Since the earliest nuclear engines are likely to be small, this leaves chemical boosting as the only choice for best utilizing the first nuclear engines. In using nuclear upper stages on chemical boosters (not specially designed for this purpose), we shall see that volume rather than weight considerations are usually limiting.

Atlas Class Boosters

Because of the relatively large engine weight involved in nuclear vehicles, Atlas and Titan may be the smallest single first stages appropriate to boost a nuclear stage. These chemical rockets can give appreciable velocities to upper stages of 30,000 to 60,000 pounds, for which reactors of ~1000 Mw (50,000 pounds thrust) are of interest. A limiting factor may be the 10 foot diameter of the present boosters. Forty thousand pounds of hydrogen would require a 10 foot diameter tank to be 125 feet long, which would make an unwieldy vehicle with an L/D $> 25$. If the hydrogen tank could be of a larger diameter (e.g., 13 feet), the length would be reduced (75 feet in this case); and a structurally and aerodynamically stable vehicle might result. This would require a detailed analysis to determine. A 30,000 pound nuclear stage seems clearly feasible but only marginally useful in terms of payload. However, this might represent a useful engine test flight vehicle for reactors up to 4000 Mw (or more depending upon their weight). The engine would be required to start at suborbital velocity (6000-10,000 ft/sec)



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and would place itself in a relatively low (~300-500 mile) orbit. The larger upper stages ($\leq$ 60,000 pounds) can place fair sized payloads in low orbits but are limited to these missions by the relatively large dead weight.

There are two possibilities worth mention here that might alter the above situation. They are the construction of a somewhat larger booster with a larger diameter tank (~13 feet) and/or the development of a lightweight nuclear engine of 30,000-60,000 pounds thrust. Either of these developments could extend the mission capability to lunar or 24 hour earth orbits, and both together could extend it to soft lunar landings. A lightweight engine would result in a valuable two stage vehicle upon retirement of the Atlas and Titan squadrons. Results are summarized in Table 8.

Table 8

CAPABILITY OF NUCLEAR SECOND STAGES BOOSTED BY ICBM CHEMICAL BOOSTERS

| Gross Wt. Including Upper Stage or Payload | Low Earth Orbit | Lunar Orbit 24 Hr. Orbit | Soft Lunar Landing, Mars Capture |
|---|---|---|---|
| 220,000 LOX-RP<br>30,000 Nuclear | 0- 5,000 | -- | -- |
| 220,000 LOX-RP<br>60,000 Nuclear | 10,000-15,000 | -- | -- |
| 360,000 LOX-RP<br>60,000 Nuclear | 15,000-20,000 | 0-3,000 | -- |
| 220,000 LOX-RP<br>60,000 Light Nuclear | 15,000-20,000 | ~5,000 | 0-2,000 |
| 360,000 LOX-RP<br>60,000 Light Nuclear | 20,000-25,000 | 5,000-10,000 | 2,000-6,000 |



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Saturn as a Booster

The Saturn vehicle, being the largest chemical rocket under development, is a natural candidate for boosting a nuclear vehicle. Studies have indicated that the most useful (in terms of payload capability) application of an early nuclear rocket of 4000 Mw or less is as an upper stage on Saturn if no nuclear booster is available. The best method of use, stage, size, power, etc. is difficult to gauge at this time, since the Saturn configuration is a function of time and is not being designed with the nuclear rocket as its objective; limitations of volume rather than weight are important, and what reactor power levels might be available is still in doubt at the time of writing. Nevertheless, we can make some representative calculations which will illustrate the areas of interest and capabilities of various nuclear stages for use with the Saturn vehicle.

We have concentrated our efforts on nuclear second stages in the range of 130,000-260,000 pounds, with particular attention to the question of the capability and desirability of various reactor power levels. After learning of the possibility of a 260-inch diameter second chemical stage, we have given some consideration to nuclear third stages. Nuclear fourth stages have been discussed under the topic of orbital start 50,000 pound vehicles.



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Saturn Second Stage Application

Our assumptions concerning the chemical and nuclear stages are listed in Table 9. The Saturn booster information is from a 1959 ABMA report[5] and corresponds physically to the C1 (interim) Saturn first stage except that the C1 engines might not be operated at the full thrust of 1,500,000 pounds. Our machine calculations (begun before information on the C1 was available) assumed a 130,000 pound nuclear second stage, which coincidentally is the same as the sum of the two chemical upper stages of the C1. For a typical case, the booster gave the 130,000 pound second stage a velocity of 8800 ft/sec at the booster burnout altitude of 40 miles, assuming a non-rotating earth. We have tried to be reasonably conservative with regard to all quantities associated with the nuclear stage. We have chosen a temperature of 2200°C exit gas and assumed 6% turbine and nozzle losses.

The machine calculations were made as follows: The first stage was flown vertically for 16 seconds, kicked over to an angle $\beta_K$ with the vertical and flown to burnout (and separation) in a gravity turn (thrust parallel to velocity), assuming a non-rotating earth. The second stage is allowed to coast for 10 seconds and then the full thrust is instantly turned on. An optimization routine directing the thrust vector flies the second stage into a 100 mile circular orbit, which is a convenient altitude for Hohmann transfer to higher orbits. Other cases were run with different values of $\beta_K$ to determine the best trajectory. The code determines the flight time, and thus one can compute the amount of



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Table 9

NUCLEAR SECOND STAGE ON SATURN, ASSUMPTIONS

| | |
|---|---|
| First stage assumptions from 1959 report on Saturn booster | $W_d$ = 100,000 pounds |
| | $W_p$ = 740,000 pounds |
| | T = 1,500,000 pounds |
| | $I_{SP}$(VAC) = 290 sec |

Nuclear Stage

Engine weight = $W_e \sim 7000 + 2$ (Mw) lbs; (e.g., 11,000 lbs/2000 Mw)
T = 2200°C $I_{SP}$(VAC) = 780 (after 6% turbine and nozzle losses)

Misc. equipment = 3000 lbs for all vehicles (Guidance, Vernier rockets, etc.)

Structure ~3% $W_o$ (4000 lbs/130,000 lbs gross weight)

Tanks ~10% $W_{propellant}$ (~8000 lbs typically)
10 sec coast period between first stage burnout and second stage thrust application

Typical Case -- 3050 Mw, T/W = 1.0

| | |
|---|---|
| Stage weight | $W_o$ = 130,000 pounds |
| Propellant weight | $W_p$ = 68,800 |
| Engine weight | $W_e$ = 12,000 |
| Structure weight ~3% $W_o$ | $W_s$ = 4,000 |
| Miscellaneous weight | $W_m$ = 3,000 |
| Tank weight ~10% $W_p$ | $W_t$ = 7,000 |
| Total dead weight | $W_d$ = 26,000 |
| Burnout weight | $W_b$ = 61,200 |
| Payload | $W_\ell$ = 35,200 |



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propellant required for each case. From the direct results for a second stage gross weight of 130,000 pounds and various thrust levels and kick-over angles, we can evaluate the minimum propellant required and thus the maximum payload for each thrust level. We thus obtain the results of Figure 6 which show that for the 130,000 pound stage weight, the payload is almost independent of thrust or power in the range of 2000 to 4000 Mw. We are able to extend the machine calculations to other vehicle weights rather simply to get the other results given in Figure 6. To do this, we note the computer calculation gives us the second stage mass ratio as a function of the second stage thrust to initial weight ratio $(T/W_o)$. We must correct this mass ratio to account for the different velocity and altitude given by the booster to upper stages of different weights, but this is a simple hand calculation for each case. This has been checked for two cases of 162,000 pounds thrust, 217,000 pound and 270,000 pound stage weights. With these results, we can examine the question from another view; i.e., given a reactor of a fixed power, what size vehicle should be designed for it. The C1 booster with 740,000 pounds of propellant and $1.5 \times 10^6$ pounds thrust is capable of carrying upper stages of 240,000 pounds or more, and a nuclear upper stage may be limited by volume considerations rather than weight. Figures 7 and 8 show that with reactor power (rather than stage weight) fixed, larger payloads can be achieved with $(T/W_o)$ values much less than 1.0, 0.6 being near optimum. This is quite understandable on the basis that the booster is capable of carrying larger upper stages with relatively little loss



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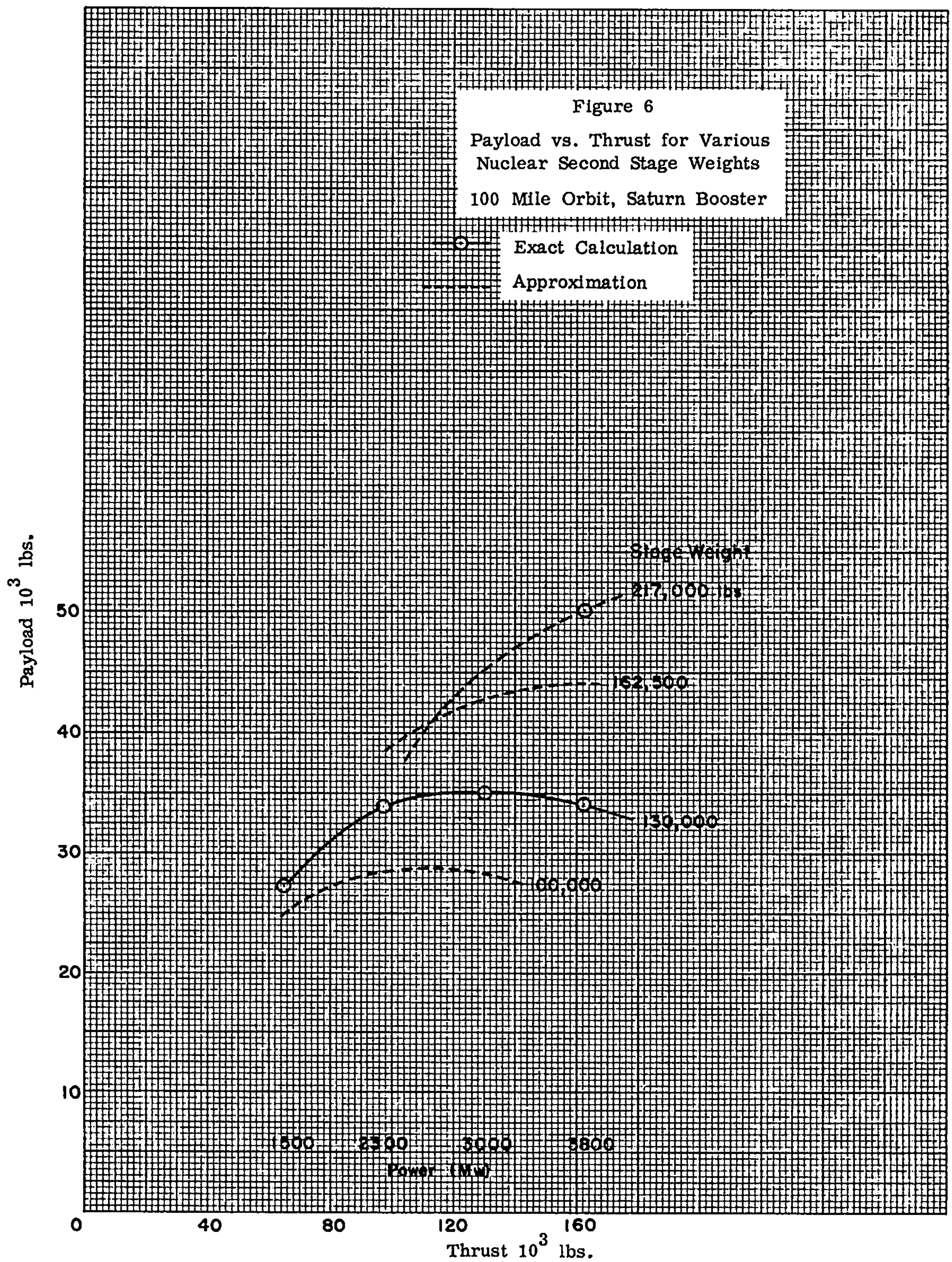




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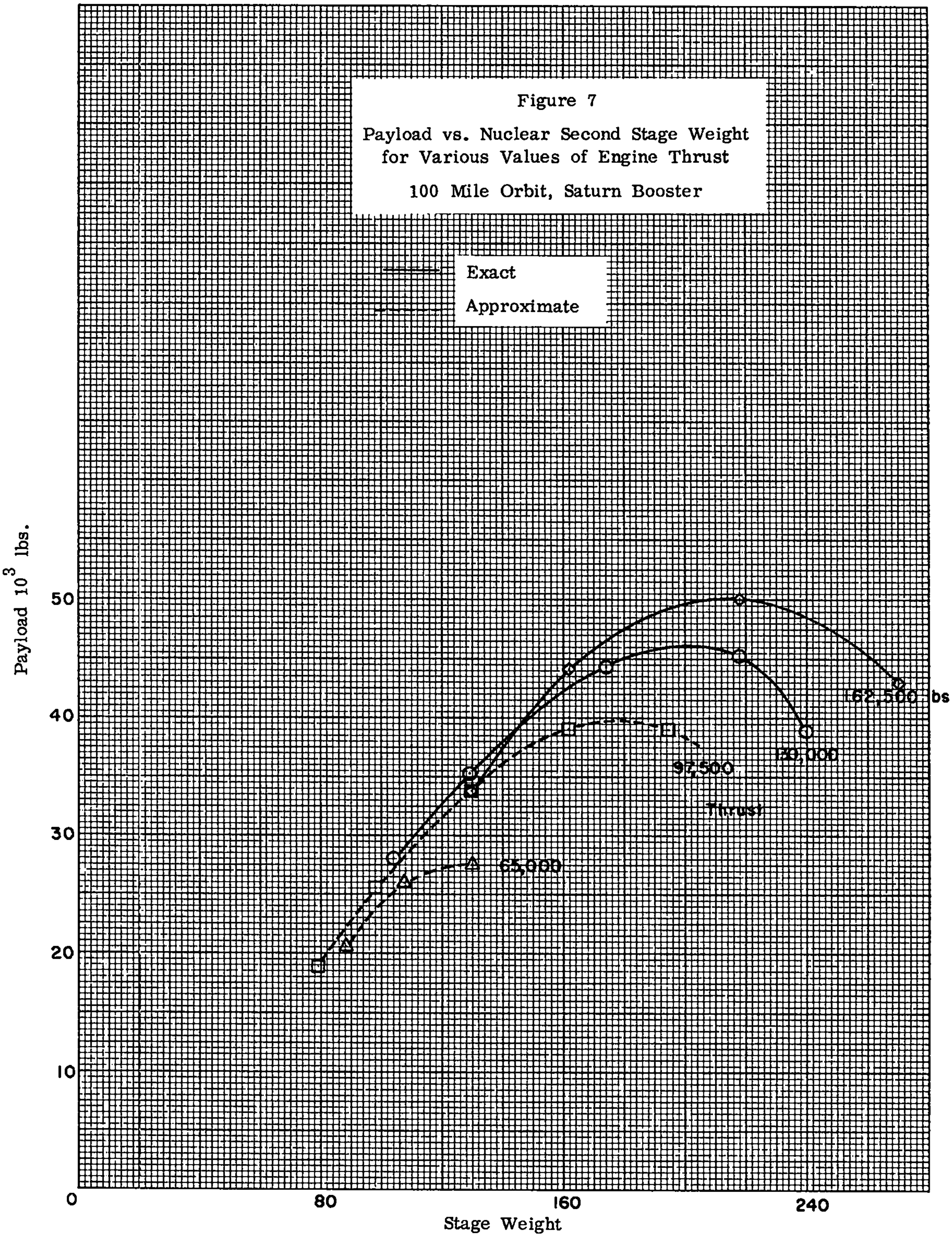


Figure 7

Payload vs. Nuclear Second Stage Weight for Various Values of Engine Thrust

100 Mile Orbit, Saturn Booster



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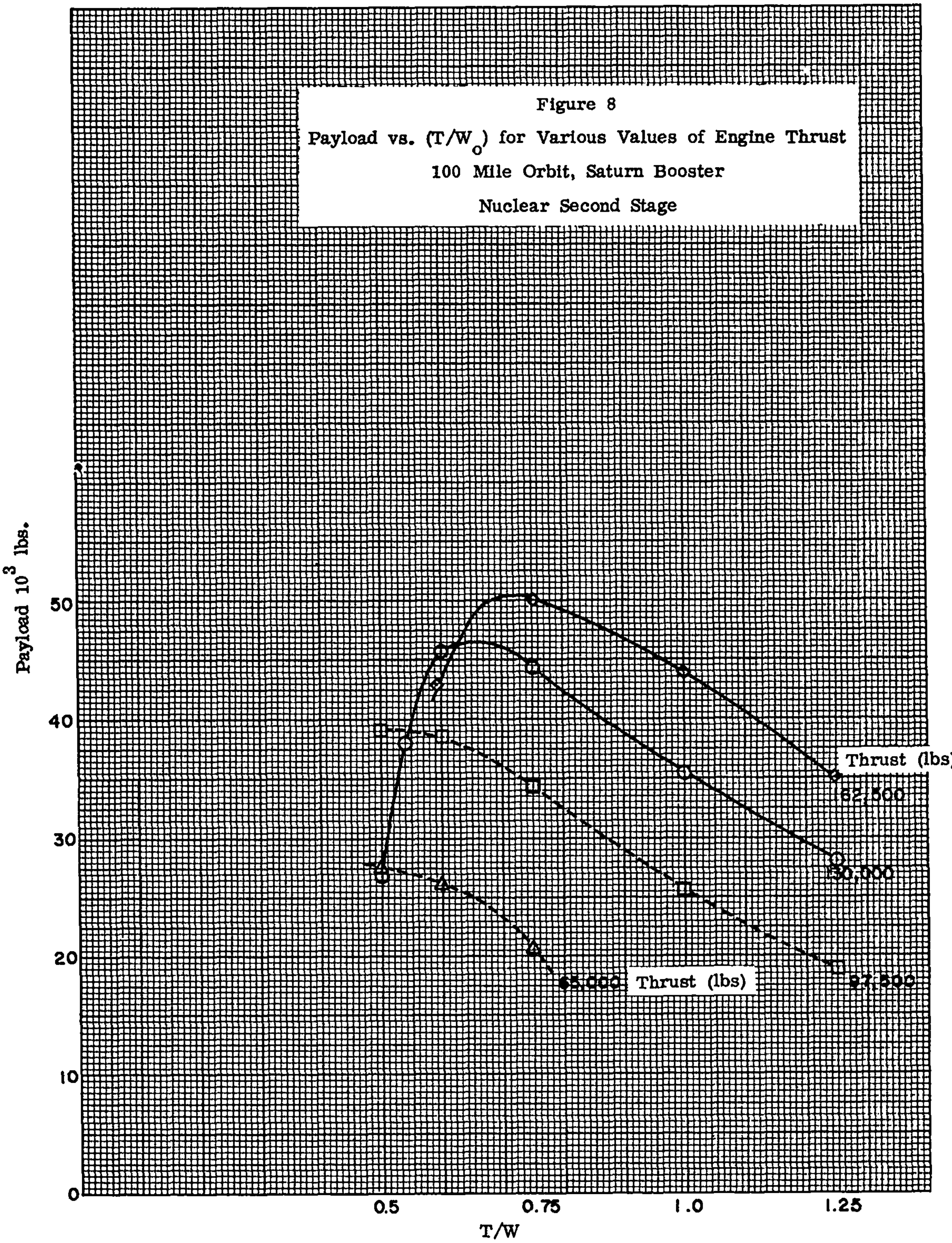




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in velocity and the fact that the second stage mass ratio (or payload) is not very sensitive to the $(T/W_o)$ value over the range of interest (.5 to 1.25). However, this result is indicative of the strong influence on results of mission studies of the type of constraints imposed on the system. These particular results are dependent upon the velocity imparted to the second stage by the booster. For example, should a larger nuclear stage (~400,000 pounds) be used on a C2 booster (which has less propellant), higher values of (T/W) will be more appropriate. On the other hand, starting from orbit, T/W values of ~.2 are acceptable. The cases which have been considered apply directly only to final missions of low earth orbits, which can be reached at small propellant cost by Hohmann transfer from the 100 mile orbit. We can re-examine these results to note the effect of changing the mission, e.g., to a Martian probe.* This would require an additional $\Delta V$ of 12,000 ft/sec at an appropriate time while the vehicle is in its 100 mile orbit. (This velocity requirement is approximately representative of many missions -- earth escape, Venus probes, lunar passes, lunar hits, and high lunar orbits.)

The results (Figure 9) show very much the same form as the low earth orbit case. For the more difficult mission, the smaller payloads are more sensitive percentagewise to reactor power and to the assumed value (of 2 pounds/Mw) for the increase in reactor weight with power.

*We choose this rather than the more difficult soft lunar landing for which many of the cases would yield negative payloads.



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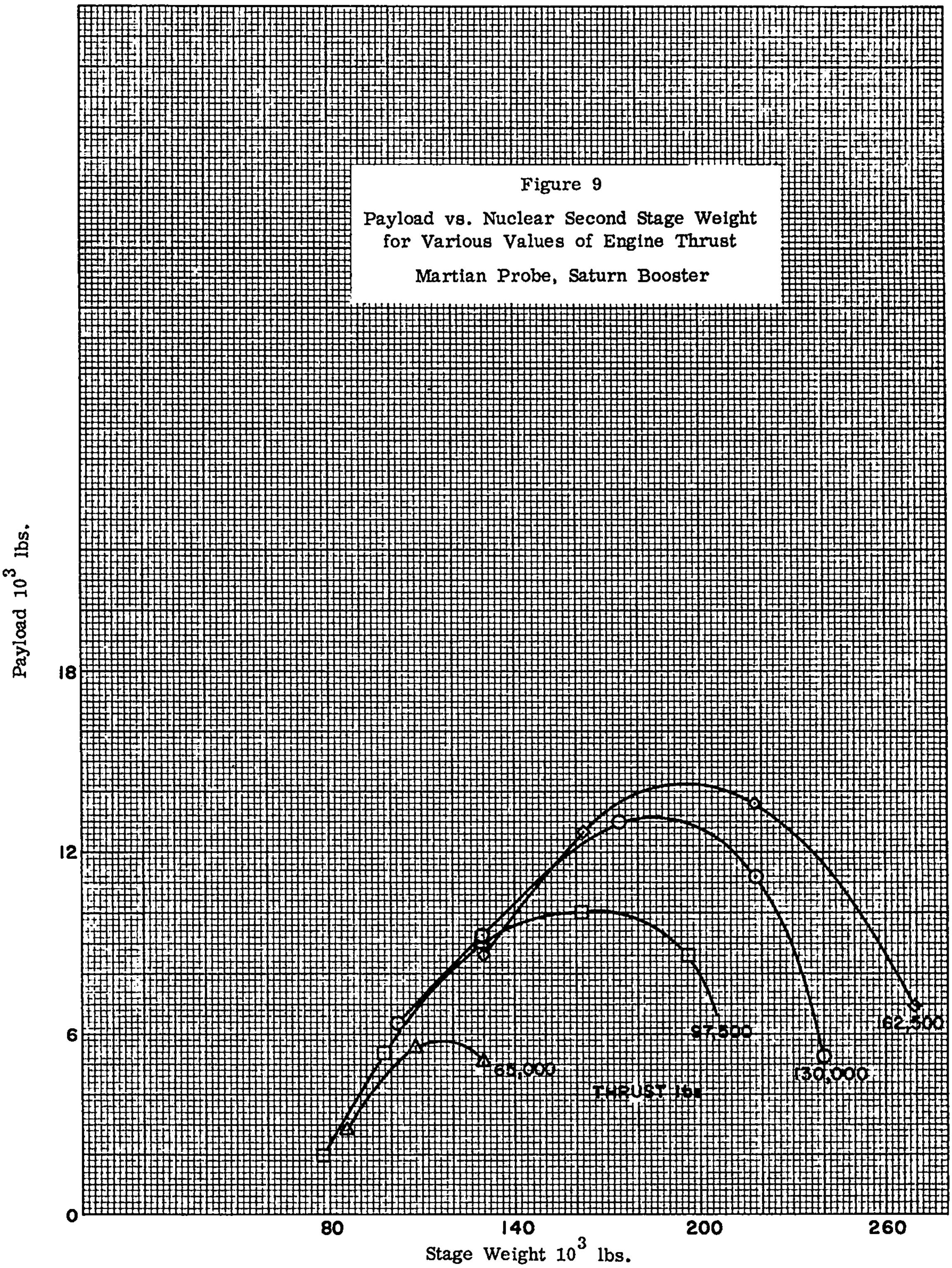


Figure 9

Payload vs. Nuclear Second Stage Weight for Various Values of Engine Thrust

Martian Probe, Saturn Booster



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The above results are about as much as can be gotten from the few machine calculations which have been made for the C1 booster. It might be noted that we have kept the booster fixed (particularly the propellant loading) for these cases, as we assume the nuclear rocket is incidental to the early ($\leq$ 1965) phases of the Saturn program, and thus the Saturn booster design will not reflect optimization to fit a nuclear stage. If the nuclear stage tank diameter is limited to 260 inches, then ~200,000 pounds is the maximum propellant load (~140 foot tank length). The C2 booster is capable of lifting a 500,000 pound nuclear second stage, but this would require a larger upper stage diameter which might result in an unstable vehicle. However, let us consider a C2-type booster with a nuclear second stage of 500,000 pounds (equal to the planned weight of the three LOX-$H_2$ upper stages) temporarily assuming this configuration is aerodynamically feasible. The booster will impart an ideal velocity of 6000 ft/sec and lift it to 20 miles or less at first stage burnout. We estimate losses will reduce the actual velocity to only 2500 ft/sec at burnout. Thus, the nuclear second stage will have to do much work climbing against gravity and an initial thrust to weight ratio of at least 1.0 is indicated, requiring a reactor in the range of 10 to 12 Bw. Making assumptions similar to those in the preceding section, we compute a payload of ~100,000 pounds in a low earth orbit (Table 10).

This low orbit payload can be improved only marginally (~15%) by using two nuclear stages in place of one. On the other hand, for more difficult missions (e.g., lunar orbit, requiring about 40,000 ft/sec from



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Table 10

$C_2$-TYPE BOOSTER WITH 500,000 POUND NUCLEAR SECOND STAGE
$I_{SP}$ = 760 (VAC)

| | | |
|---|---|---|
| $W_p$ | Propellant weight (16' x 150' tank) | 320,000 pounds |
| $W_b$ | Burnout weight in orbit | 180,000 |
| $W_t$ | 8% tank weight | 26,000 |
| $W_m$ | Miscellaneous | 4,000 |
| $W_s$ | Structure | 12,000 |
| $W_e$ | 10-12 Bw engine | 38,000 |
| $W_d$ | Dead weight | 80,000 |
| $W_\ell$ | Payload in low orbit | 100,000 ± 10,000 pounds |

the upper stages), the use of two upper stages will increase the payload from 20,000 pounds to at least 42,000 pounds. A sizeable payload (~30,000 pounds) could be soft landed on the moon.

Saturn Nuclear Third Stage

Now let us consider the C2 booster plus part or all of its planned chemical second stage. If we use the full second stage (330,000 pounds propellant and ~40,000 pounds dry weight), we can put up to 130,000 pounds of nuclear third stage on it. From knowledge of the capabilities of the all-chemical system, we can make rough estimates of the gravitational losses and the actual velocity given to the third stage. The latter turns out to be 15,000 ft/sec, which is large enough so that low thrust



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to initial weight ratios ($\lesssim .5$) should be quite adequate for the nuclear stage ($\lesssim$ 1500 Mw for this case of 130,000 pound stage weight). We conclude this configuration will place ~62,000 pounds in low orbit (compared to 47,000 pounds for a three stage all-chemical rocket) and can soft land 11,000 pounds on the moon, again assuming an $I_{SP}$ of 760 seconds (2200°C exit gas, 6% losses).

Finally, let us examine one intermediate case where the second chemical stage propellant load is reduced from 330,000 pounds to 200,000 pounds (keeping the dead weight fixed at 40,000 pounds) to allow for a larger (260,000 pound) nuclear third stage. The two chemical stages can give the 260,000 pound nuclear third stage a velocity of ~9000 ft/sec. Since this is about the same velocity as occurred for the small second stages considered earlier, 3000-4000 Mw would represent a useful power range for such a vehicle, which is the expected power for the Phoebus engine. This combination can put 94,000 pounds in a low orbit (exactly twice the three chemical stage capability) or soft land 19,000 pounds on the moon (compared to 3500 pounds for a four stage chemical Saturn). Off-loading the first stage propellant instead of the second stage would increase these payloads 8000 and 3000 pounds, respectively.

## Later Missions

Both the possible missions and vehicle configurations are unlimited in number, but there are some of particular interest for which nuclear rockets offer great advantages. Manned satellites and circumlunar flight



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can be achieved with chemical rockets already in the development stages. A manned lunar expedition, requiring ~20 km/sec total mission velocity, would require a very large multistage rocket or assembly and refueling in orbit if chemical propulsion alone were used. In a separate study[6] which will be summarized here, we find that both the gross weight and the manufactured weight can be drastically reduced by the use of one or more nuclear stages. The assumptions are listed in Table 11 and include three sets of values for tank and engine weight fractions to reflect the uncertainty in the extrapolation of the state of the art. We have assumed 55,000 pounds to be soft landed on the moon, including 10,000 pounds to be left there and a storable chemical stage which returns 15,000 pounds to earth with full atmospheric braking upon re-entry. The results are given in Figure 10, which depicts the vehicle configurations, and Figure 11, which shows the gross weight, dry (manufactured) weight, and propellant volume for the vehicles as functions of the state of the art. Case C, consisting of two nuclear powered stages on a chemical booster, appears to be the most interesting one for several reasons. The gross weight is relatively small and the dry weight and propellant volume are the smallest of those cases considered. The larger of the two nuclear engines is only about 10,000 Mw, and the chemical booster can be powered by only two F-1 ($1.5 \times 10^6$ pound thrust) engines (an $O_2$-$H_2$ or solid booster could also be used if available). The chemical booster relieves the radiation hazard problem, especially with regard to the air scattered dose to the payload and launch site. The two nuclear stage case D is half as heavy as Case C,



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Table 11

LUNAR MISSION ASSUMPTIONS

| | $\Delta V_I$ ft/sec | $\Delta V_I$ (with losses etc.) |
|---|---|---|
| Earth surface to low orbit | ~26,000 | 33,000 |
| Earth orbit to moon | ~10,500 | 12,000 |
| Lunar landing | 7,800 | 10,000 |
| Lunar surface to earth | 7,800 | 10,000 |
| Earth landing-atmospheric braking | | 0 |
| | | 65,000 ft/sec |
| | | or |
| | | 20 km/sec |

| System | $I_{SP}$(VAC) s | Propellant lbs/ft$^3$ | Tank Fraction f Opt. | Prob. | Pess. | Structure and Engine Fraction $\epsilon$ Opt. | Prob. | Pess. |
|---|---|---|---|---|---|---|---|---|
| LOX-RP | 300 sec | 63.4 | .01 | .02 | .04 | .01 | .02 | .04 |
| LOX-$H_2$ | 416 | 17.3 | .02 | .04 | .05 | .02 | .03 | .05 |
| Nuclear $H_2$ | 860 | 4.34 | .05 | .07 | .10 | .05* / .04 | .08* / .05 | .10* / .08 |

| | $I_{SP}$(VAC)$_{sec}$ | Ve, ft/sec | Fuel Density, lbs/ft$^3$ |
|---|---|---|---|
| LOX-RP | 300 | 9,660 | 63 |
| LOX-$H_2$ | 417 | 13,400 | 17.3 |
| Nuclear $H_2$ | 860 | 27,700 | 4.34 |

*Ground launched stages only.



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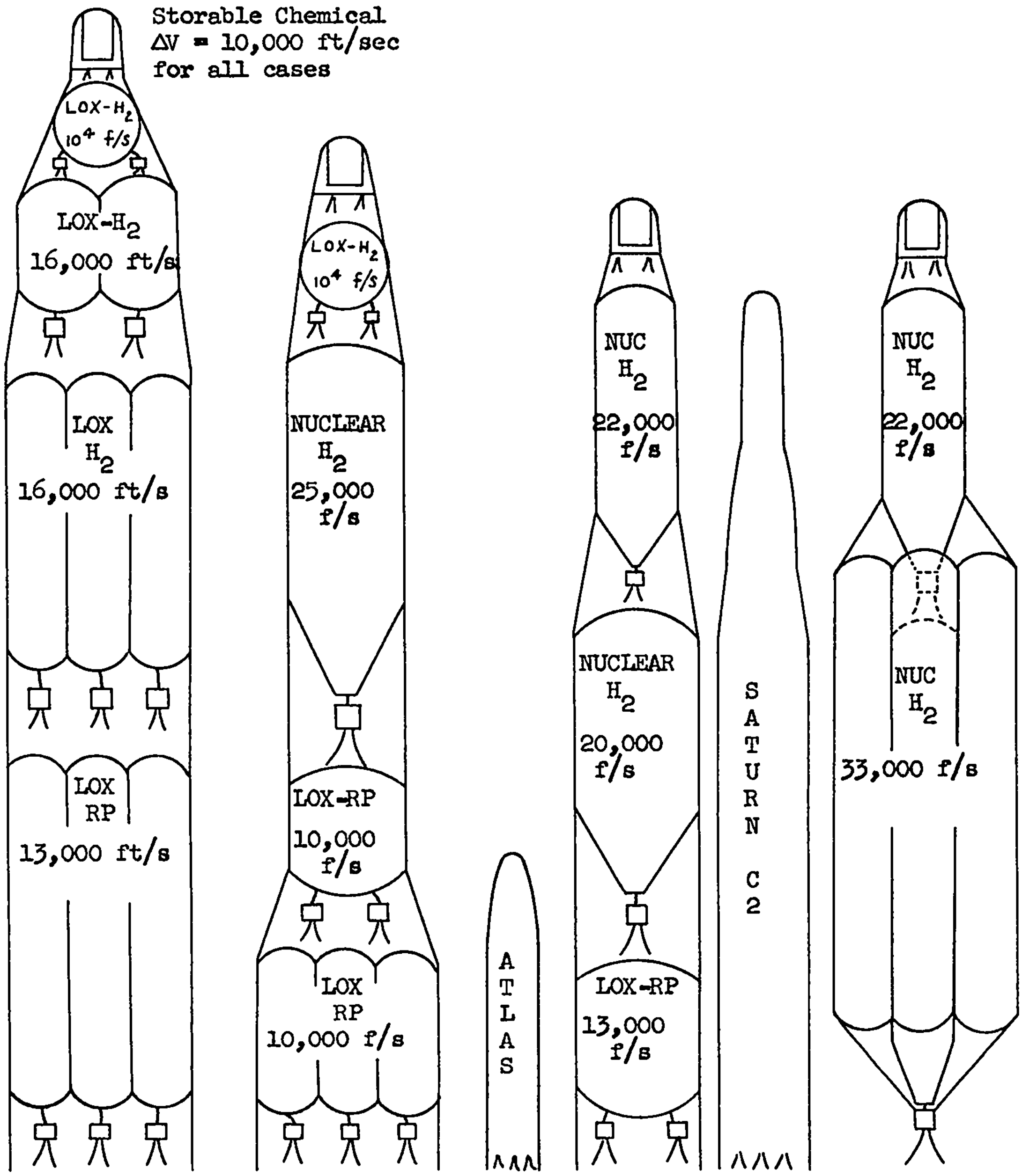


| CASE | A | B | C | D |
|---|---|---|---|---|
| $W_o$ | 9,650,000 | 3,980,000 | 1,660,000 | 860,000 lbs |
| $W_{dry}$ | 504,000 | 220,000 | 105,000 | 125,000 lbs |
| $V_p$ | 223,000 | 110,000 | 81,000 | 158,000 $ft^3$ |

Figure 10 Lunar Vehicle Configuration



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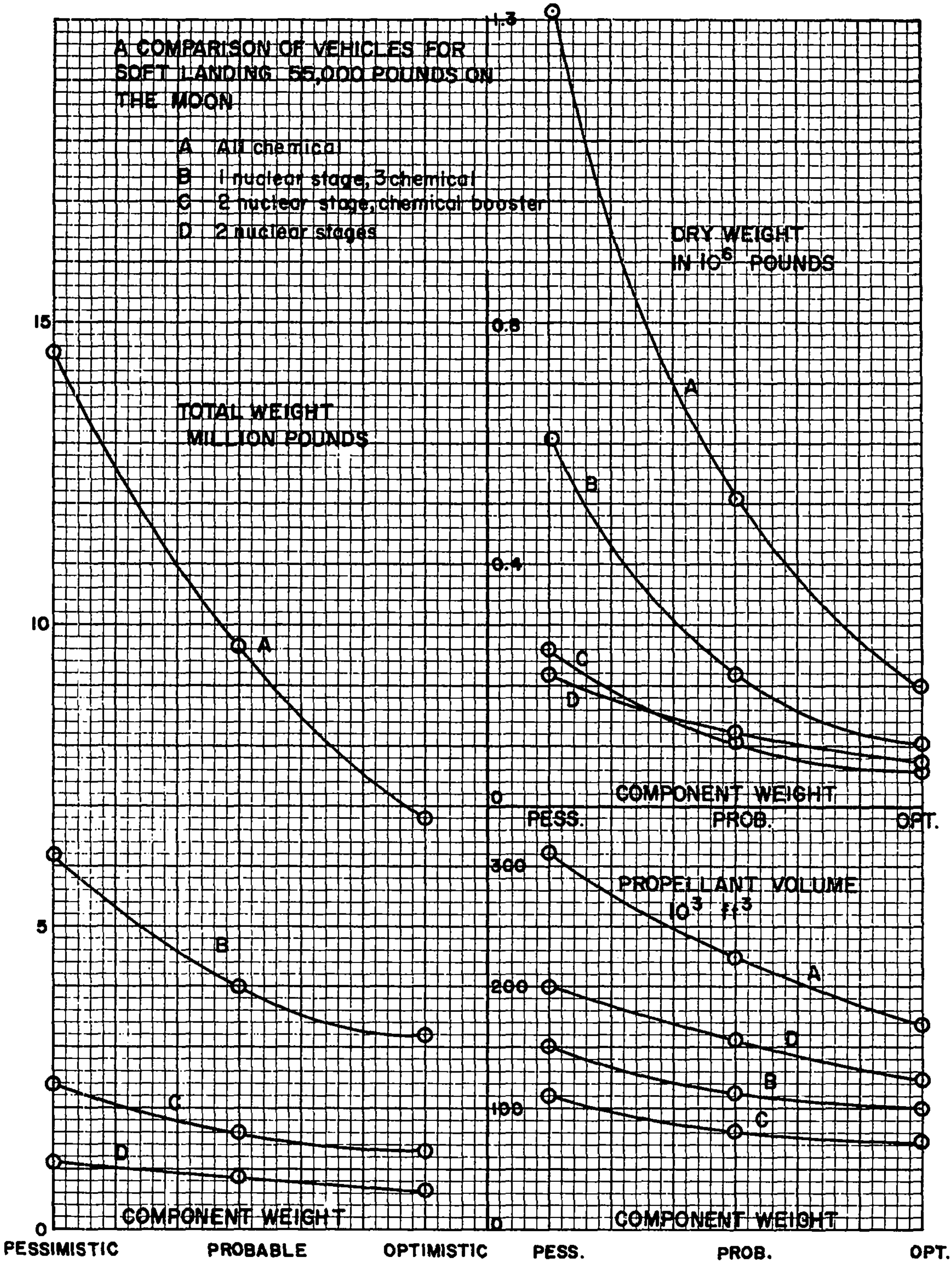


Figure 11



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but has twice the propellant volume. The first stage places itself in orbit where separation can take place slowly and safely, but an engine of 25,000 to 30,000 Mw is required.

There have been unclassified studies of manned reconnaissance of inner solar system planets using ground launched[7] or orbital start[8] nuclear rockets. On such long journeys, electric propulsion may prove competitive although nuclear propulsion appears closer to realization, and its high thrust may be essential in reducing the time spent in the radiation belts. Some earlier missions of scientific interest, i.e., fast, heavy ($\sim 10^4$ pounds), unmanned probes to the outer planets and to the vicinity of the sun, have been briefly discussed under Two Stage Vehicles. A solar probe, for example, would require high mission velocity capability to approach the sun closely, either by a direct orbit or by a "two-kick" transfer in which the vehicle first moves away from the sun and at apogee decelerates. Large payloads are required to provide thermal shielding and communications equipment. Nuclear propulsion is well suited to provide this combination of large payload and high mission velocity.

## Summary

We shall summarize the results in terms of the approximate capabilities of the reactors under consideration in the Rover Program.



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1. Kiwi -- UC-Graphite core, 50% unloaded graphite structure.

   ~1400 Mw (60,000 pounds thrust) $W_e$ = 8000 pounds.

   Status: Developmental

   A. Vertical flight test vehicle (~40,000 pounds) with Atlas tankage-altitude 50-200 miles. Special $H_2$ tankage-altitude 500-1000 miles.

   B. Insufficient for ground launch to orbit single stage. (~2000 Mw in same weight engine required.)

   C. On ICBM booster 5000-15,000 pounds in low orbit.

   D. On Saturn booster

      second stage -- not optimum, power too low; ~25,000 pounds in low orbit.

      third stage -- power somewhat low; 60,000 pounds in low orbit. 10,000 pounds soft landed on moon.

   E. Orbital start vehicles weighing from 50,000 to 300,000 pounds, including manned lunar vehicle upper stage.

2. Phoebus UC-Graphite core, fully loaded.

   ~4000 Mw (180,000 pounds thrust) $W_e$ ~ 8000 pounds.

   Status: Design

   A. Single stage ground launched vehicle (~130,000 pounds).

      10,000 pounds in earth orbit, ~2000 pounds to escape velocity.



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B. Power level too high for ICBM boosted stage, but engine may be tested in this configuration.

C. On Saturn booster

Good power level for volume limited second stage (50,000 pounds in orbit with 200,000 pound stage). Slightly overpowered, but useful for third stage (~150,000 pounds). Most promising for larger (~260,000 pound) third stage.

D. Orbital start vehicles weighing up to $10^6$ pounds especially where high thrust is desired, e.g., lunar landing stage.

3. Condor -- Large graphite reactor ~10,000 Mw (400,000 pounds thrust). ~20,000 pounds weight.

Status: Conceptual

A. Single stage ground launched vehicle. ~10% of gross weight into low earth orbit.

B. Too large for ICBM boosted stage, except for testing.

C. Too high power for 260 inch Saturn boosted stage, due to volume limitations, except for test purposes.

D. Appropriate for large diameter 1 to 2 x $10^6$ pound chemical booster, particularly for lunar mission.

E. Too low power for nuclear booster for manned lunar mission (at least 25,000 Mw required).



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4. Roc -- Low power, lightweight engine; ill-defined at present.

   100-2000 Mw range; 500 to 4000 pounds.

   Status: Conceptual

   A. At 1000-2000 Mw power level, all Kiwi missions with ~5000 pounds additional payload, of particular interest in 50,000 pound stages on ICBM boosters or orbital start stages.

   B. At 100-500 Mw power level, small upper stages of all kinds, particularly orbital start interplanetary probes.

Conclusions

On a gross weight basis, even the earliest nuclear engines will give higher performance than LOX-$H_2$ chemical rockets for stage weights over 60,000 pounds. With slightly more advanced or larger engines, nuclear propulsion is better than chemical propulsion on a manufactured weight per payload basis for missions requiring velocities of 7 km/sec (e.g., low earth orbits) or more. For the early engines, which will have powers of 4000 Mw or less (thrusts of 180,000 pounds or less), the most profitable use in terms of payload is as an upper stage. In such use, they can increase payloads by factors of two or more over all-chemical vehicles. Nuclear upper stage size is limited (owing to the low density of $H_2$) by the diameters rather than the thrusts of the existing chemical boosters (ICBM and Saturn). Finally, for the manned lunar expedition, the use of



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nuclear propulsion in only one stage can reduce vehicle gross and manufactured weights by a factor of two, while an additional nuclear stage can reduce them by a factor of four or more.



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# APPENDIX

We are looking for a simple approximate function for the payload fraction which is independent of staging effects. As we saw (Eq. (5)) with a non-zero value of $\epsilon$ (the engine fraction), the infinite staging limit leads to zero payload for all $\Delta V$. Instead, we shall assume the vehicle parameters ($f_i$, $\epsilon_i$, $v_e^i$) and the stage velocity increments ($\Delta V_i$) to be equal for all stages and determine the optimum staging limit, allowing n, the number of stages, to be a continuous variable. The payload for n stages is given by

$$y_n = \left[(1 + f)e^{-\beta/n} - f - \epsilon\right]^n = u^n, \qquad \text{(A1)}$$

where $\beta = \frac{\Delta V}{v_e}$ and u is the function in the square brackets. Maximizing $y_n$ with respect to n leads to

$$u \ln u + \frac{\beta}{n} (u + f + \epsilon) = 0 \qquad \text{(A2)}$$

or

$$u \ln u = (1 + f)a \ln a \qquad \text{(A3)}$$

where

$$a = \frac{u + f + \epsilon}{1 + f} = {}^{-\beta/n} = (\text{mass ratio})^{-1}. \qquad \text{(A4)}$$



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We solve (A3) approximately by letting $u = a - \lambda$ and expand the left side of (A3), giving

$$\lambda \cong - \frac{fa\, \ell n\, a}{1 + \ell n\, a}, \tag{A5}$$

whence

$$u \cong a + \frac{af\, \ell n\, a}{1 + \ell n\, a}; \tag{A6}$$

and using the definition of u, we have

$$a \cong \frac{(f + \epsilon)}{f} (1 + \ell n\, a). \tag{A7}$$

The analysis to this point is due to Dr. K. Brueckner.

Since "a" is the inverse mass ratio for each stage, we know from experience that it is about $e^{-1}$ in optimized systems; and so we let $a = e^{-1} + \delta$, leading to

$$\delta \cong \frac{f}{e[e(f + \epsilon) - f]} \tag{A8}$$

and

$$a \cong \frac{1}{e(1 - b)}, \tag{A9}$$

where

$$b = \frac{f}{e(f + \epsilon)} \leq \frac{1}{e}. \tag{A10}$$



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From (A4) and (A9),

$$n(\text{optimum}) = \frac{\beta}{\ln a^{-1}} = \frac{\beta}{\ln e(1 - b)} \cong \beta(1 + b + \tfrac{1}{2} b^2). \tag{A11}$$

This says $n \approx \beta$, which is to be expected when $a \simeq e^{-1}$. Thus

$$y(\text{optimum}) = \left[(1 + f)a - f - \epsilon\right]^n = \left[\frac{1 + f}{e(1 - b)} - f - \epsilon\right]^{\frac{\beta}{\ln e(1-b)}}. \tag{A12}$$

Note that this is of the form of

$$y = A^{-k\,\beta} \tag{A13}$$

or

$$\ln y = -k(f,\epsilon)\,\frac{\Delta V}{v_e}\,. \tag{A14}$$

We can evaluate k under various assumptions about f and $\epsilon$, always including $f \ll 1$ and $\epsilon \ll 1$. We get the following results:

$$\left\{\begin{array}{l} f = \epsilon \ll 1 \\ \\ k \cong (1 + 4.32f) = (1 + 4.32\epsilon) \end{array}\right\} \tag{A15.1}$$

$$\left\{\begin{array}{l} \epsilon \ll f \ll 1 \\ \\ k \simeq (1 + 1.33f + 3.18\epsilon) \end{array}\right\} \tag{A15.2}$$



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$$\begin{Bmatrix} f \ll \epsilon \ll 1 \\ \\ k \simeq (1 + 1.72f + 2.72\epsilon) \end{Bmatrix} \qquad \text{(A15.3)}$$

In view of the fact that these are only for an approximate limiting case, that they give very similar results, and that in practice $\epsilon \simeq f$ for both nuclear and chemical systems, we choose a compromise which is correct for $\epsilon = f$ and reflects the relative importance of the two parameters,

$$\ell n\ y = -(1 + 1.42f + 2.9\epsilon)\ \frac{\Delta V}{v_e}\ . \qquad \text{(A16)}$$