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LOS ALAMOS SCIENTIFIC LABORATORY

OF THE UNIVERSITY OF CALIFORNIA ○ LOS ALAMOS NEW MEXICO

APPLICATIONS OF LOW POWER NUCLEAR ROCKET ENGINES

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OF THE UNIVERSITY OF CALIFORNIA LOS ALAMOS NEW MEXICO

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ABSTRACT

The capabilities of low powered ($\leq$ 2000 Mw) nuclear rocket engines have been examined for a wide variety of missions, including orbital probes and ferries, maneuverable satellites, and small upper stages on ICBM boosters. Lightweight engines based on the fast reactor concept (ROC) are described and their performance compared to that of graphite (KIWI) reactor engines and of O_2-H_2 chemical propulsion. While results vary with the mission, for stage weights of the order of 50,000 pounds, the ROC stages have significantly better performance than KIWI or O_2-H_2 stages. For stages larger than 100,000 pounds, the difference between the two types of nuclear engines becomes less important, and both types are quite superior to chemical propulsion. The ROC reactors offer good performance in small (15,000 to 50,000 pounds) second stages on ICBM boosters and appear to offer a rapid avenue to small useful nuclear rocket engines. Their small physical size offers great advantages if much shielding is required as might be for a debarking crew. The effects of specific impulse, reactor weight, and of tank staging are illustrated analytically and by examples.

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Introduction

The purpose of this paper is to examine many possible applications for low power nuclear rocket engines, defining low power to be $\leq$ 2000 Mw or $\leq$ 100,000 pounds thrust. While this value is somewhat arbitrary, it is in the range where graphite reactors approach their minimum size and weight. Other reactor concepts which may lead to smaller engine weights in this power range can have a strong effect on the usefulness of nuclear propulsion and therefore should be considered. We shall place particular emphasis on the fast reactor concept to determine possible engine weight and power characteristics which might also be applicable to other reactor types.

The analysis is based on very simple methods which have been checked against more exact calculations. Furthermore, the uncertainty in component weights and performance limits the value of very detailed computations. Finally, because of the high development costs of engines, we feel that versatility will be generally more important than optimization for a particular mission, and thus a wide variety of missions have been examined. Results are based primarily on the equation

$$\frac{M_L}{M_o} = (1 + f)e^{-\Delta V/v_e} - f - \epsilon, \tag{1}$$

where

M_L = payload mass

M_o = gross mass

ΔV = stage velocity increment

v_e = exhaust velocity

f = (tank mass)/(propellant mass)

ϵ = (engine and miscellaneous mass)/(gross mass).

Equation (1) comes from a straightforward mass balance and the field free dynamics for the stage mass ratio $\left(R = e^{\Delta V/v_e}\right)$ in terms of stage velocity increment. Where necessary, gravity losses have been included in ΔV or (e.g., for low acceleration from orbit) more exact calculations have been used to obtain R directly.

Reactors

At present, only graphite rocket reactors have reached the hardware stage with the first engine prototype, KIWI, to have a nominal power of 1170 Mw (~55,000 pounds thrust), an I_{sp} of ~800 seconds, and a weight of 7000 to 8000 pounds. A more advanced core concept involving a fully loaded core (Phoebus) is expected to triple the power for the same engine weight and I_{sp}. This concept might lead to a minimum weight of 5000 to 6000 pounds for graphite reactor engines. The amount of uranium which can be loaded in the graphite without reducing its strength is the limiting factor in reducing reactor weight.

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Many suggestions have been put forward for small nuclear rocket reactors, including UO_2-BeO reactors and UO_2-W or UO_2-Mo fueled reactors moderated by ZrH, BeO, or even H_2O. Detailed discussion and comparison of them is beyond the scope of this report, and lack of experimental information would leave many points undecided. Thus we shall concentrate upon one kind, fast reactors (ROC)*, for this report and consider their weight-power relation as representative.

A fast (or unmoderated) assembly can be made critical with a total weight of 50 pounds or less, and thus weight itself is not the problem. One must find materials which can exist at very high temperatures in forms which allow efficient heat transfer to the propellant and satisfy a number of subsidiary conditions (neutronic, structural, chemical, etc.). Two classes of fuel elements, UO_2-W cermets and UC-metal carbide solid solutions, seem well suited to this purpose. The UO_2-W cermets may be able to operate up to the melting point of the UO_2 ($\sim 2800^\circ C$) and can be loaded with 40 or 50% by volume of UO_2. The UC solid solution melting points depend upon the concentration and melting point of the other metal carbide. UC itself melts at $2450^\circ C$ and thus might be of interest where low weight is desirable at the expense of high I_{sp}. Be is used for reflection of all reactor cores considered here. UC-ZrC solid solutions have been investigated experimentally (ZrC melts at $3500^\circ C$) and have also been considered for fuel elements. HfC and TaC, which melt at $3800^\circ C$,

*R. Cooper, "Lightweight Nuclear Rocket Reactors," Los Alamos Scientific Laboratory Report LAMS-2404 (December, 1959).

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are the highest melting solids, but further experimental work is required before preliminary reactor design can begin. Finally, one might use U^{233} which is twice as effective neutronically as U^{235} for fast spectra. It could be used to lower reactor weight, increase void (gas flow) volume, or reduce uranium loading in graphite, UO_2-W or UC-MC fuel elements. Figure 1 shows the general range of weight vs. power for various general reactor types, including the present graphite designs. Figure 2 illustrates the wide variety possible in the characteristics of fast reactors due to the choice of fuel element material, loading, and void fraction. One might approximate the engine weight-power relation as

$$M_e \text{ (pounds)} = 1000 + P \text{ (Mw)} \tag{2}$$

with the understanding that this is uncertain by 500 to 1000 pounds. This is, however, sufficiently precise for our purposes. The set of reactors of Figure 2 are generally designed to operate at high pressure (~500 psi) and deliver gas at 1500°C to 2500°C (I_{sp} in the range of 700-850 seconds). However, there is another reason (other than low weight) which might lead one to low power reactors. That is to operate at low pressure ($\leq$ 100 psi) in order to take advantage of dissociation of H_2 in raising the specific impulse. Figure 3 shows that exit gas temperatures over 3000°C are necessary for this effect to be appreciable. This might be achieved with a lightly loaded UC-ZrC fuel element, with fast UC-HfC "after-heaters" or with more radical reactors (e.g., dust bed or liquid

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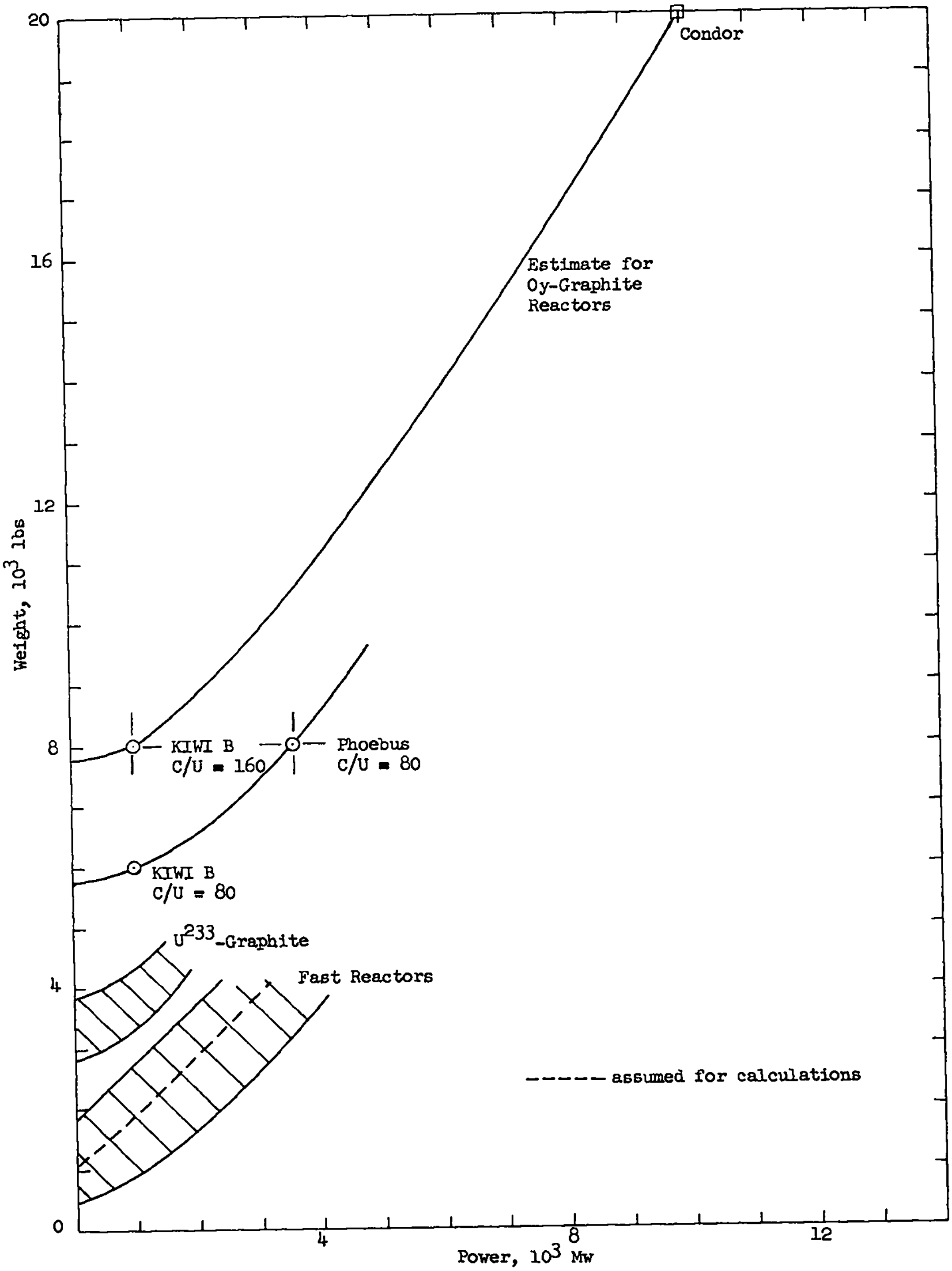

Figure 1 A comparison of the weight-power domains of fast reactors and graphite systems

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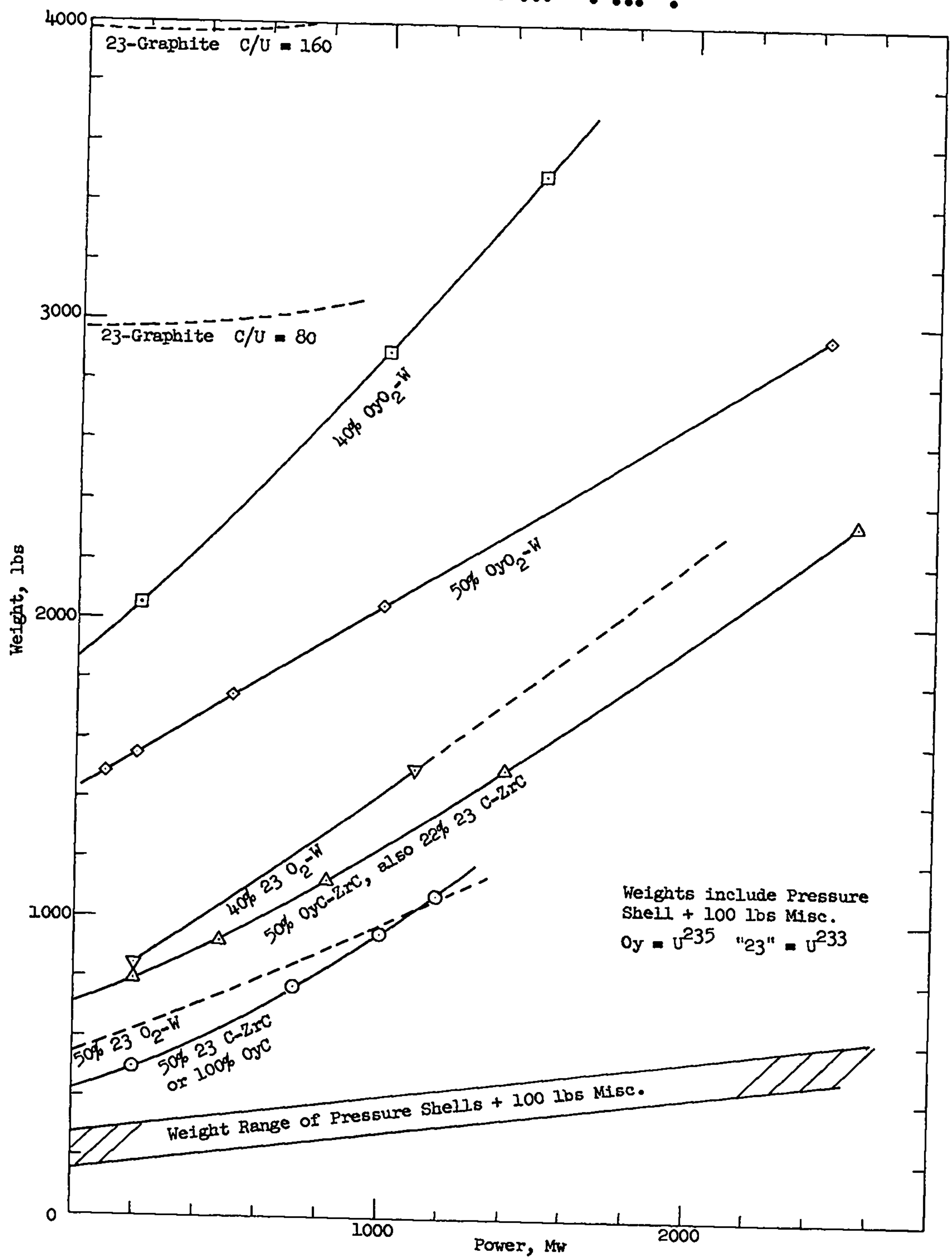

Figure 2 A comparison of weights for the various reactor types

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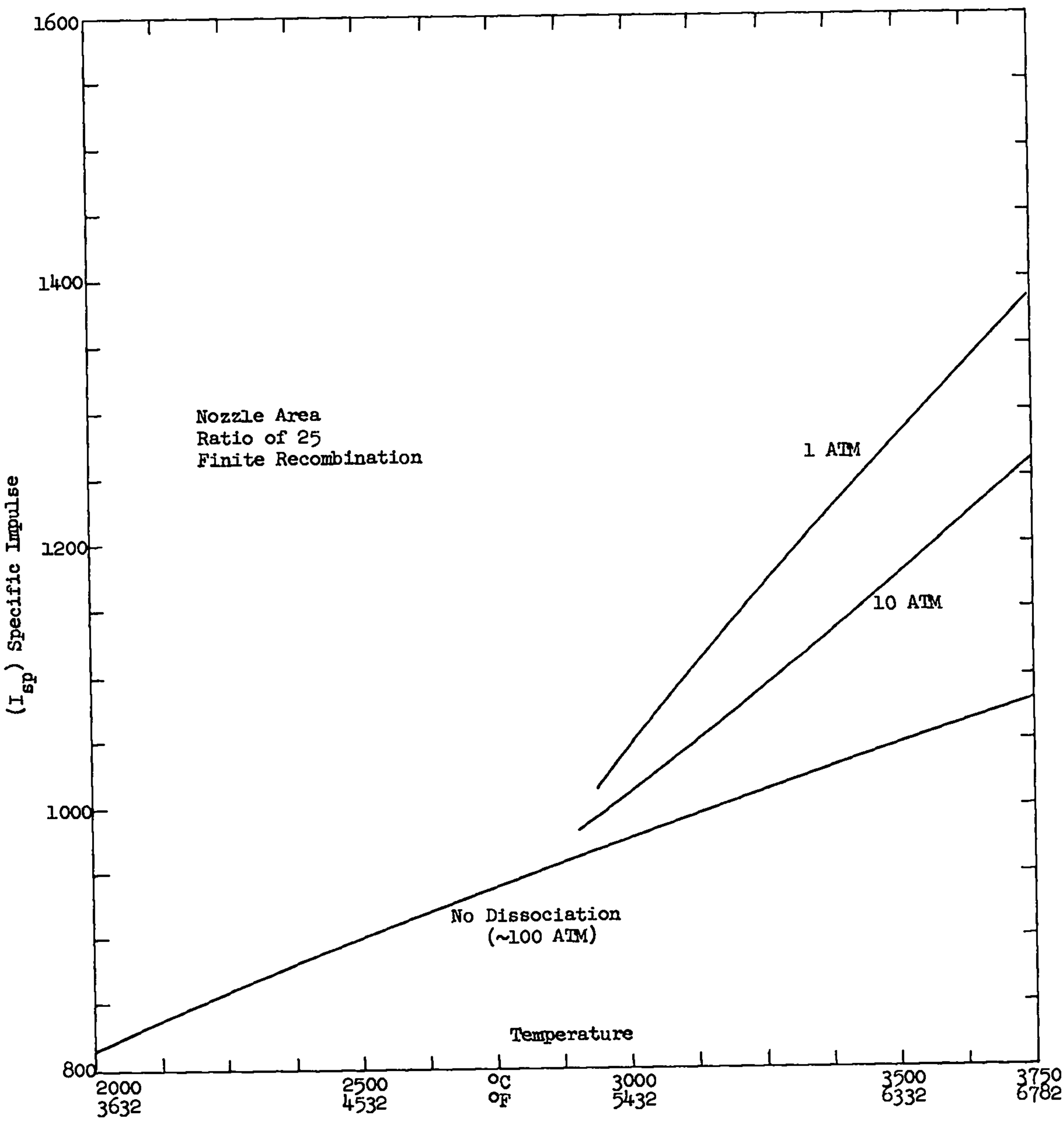

Figure 3 Specific impulse of hydrogen

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core). No specific designs exist, and thus one can only guess what weight penalty, if any, would be incurred in raising the I_{sp}. We do include cases with 1100 seconds I_{sp} to determine the desirability of effort in that direction. Reduction in structural and tank weights (for which we have assumed a value of 10%) would also produce significant improvements in performance.

Applications

Probes

We shall begin with one-way orbital start vehicles, a possible use for early, low power nuclear engines. The missions of interest would constitute fast exploratory probes to the inner and outer solar system, including capture at the target planet. We will consider single stages with restartable engines and for a few difficult missions, examine multiple staging. Three values for the initial weight in orbit will be taken corresponding to the approximate payload capabilities of Saturn C2 (50,000 pounds), C1 (20,000 pounds), and Atlas Centaur (10,000 pounds). The orbital start will allow the use of low accelerations (~.3g) with negligible payload penalty compared to impulsive thrust. An example of this effect, including the limiting case of zero thrust, is given in the appendix.

The case of 10,000 pounds in orbit is mainly of interest as an experimental or development tool, as the missions can be easily performed with comparable chemical rockets. A thrust of only 3000 pounds is required corresponding to 50 to 70 Mw. One thousand pounds should suffice for the power plant, and we shall assume 1000 pounds of tankage, insulation, and structure and 500 pounds of miscellaneous items (guidance, control, etc.). In such a small engine, very high performance will be difficult, and we will consider specific impulses of 700 seconds (1500°C) and 800 seconds (2200°C). The incremental payload for changes in specific impulse is given approximately by*

$$\frac{dM_L}{dI_{sp}} = \frac{M_o}{e\,I_{sp}} , \tag{3}$$

which is ~5 pounds/second for this case and is useful for interpolating or extrapolating the results given in Figure 4.

It is worth noting that even in this small size, with fast reactor engines, nuclear propulsion can equal or better chemical propulsion. The payload gains in themselves may be insufficient to justify a large nuclear engine development effort. If large-scale orbital operations make low power nuclear engines desirable, this type of vehicle is of value in the development phase for gaining operational experience. Because of the large development costs associated with each engine, one should try to develop as few different engines as possible. In this case, we are striving for very low engine weight at the cost of performance and power. This power (50 Mw) is probably too low to be of

*R. Cooper, "Mission Studies for Nuclear Heat Exchanger Rockets," Los Alamos Scientific Laboratory Report LAMS-2512 (December, 1960).

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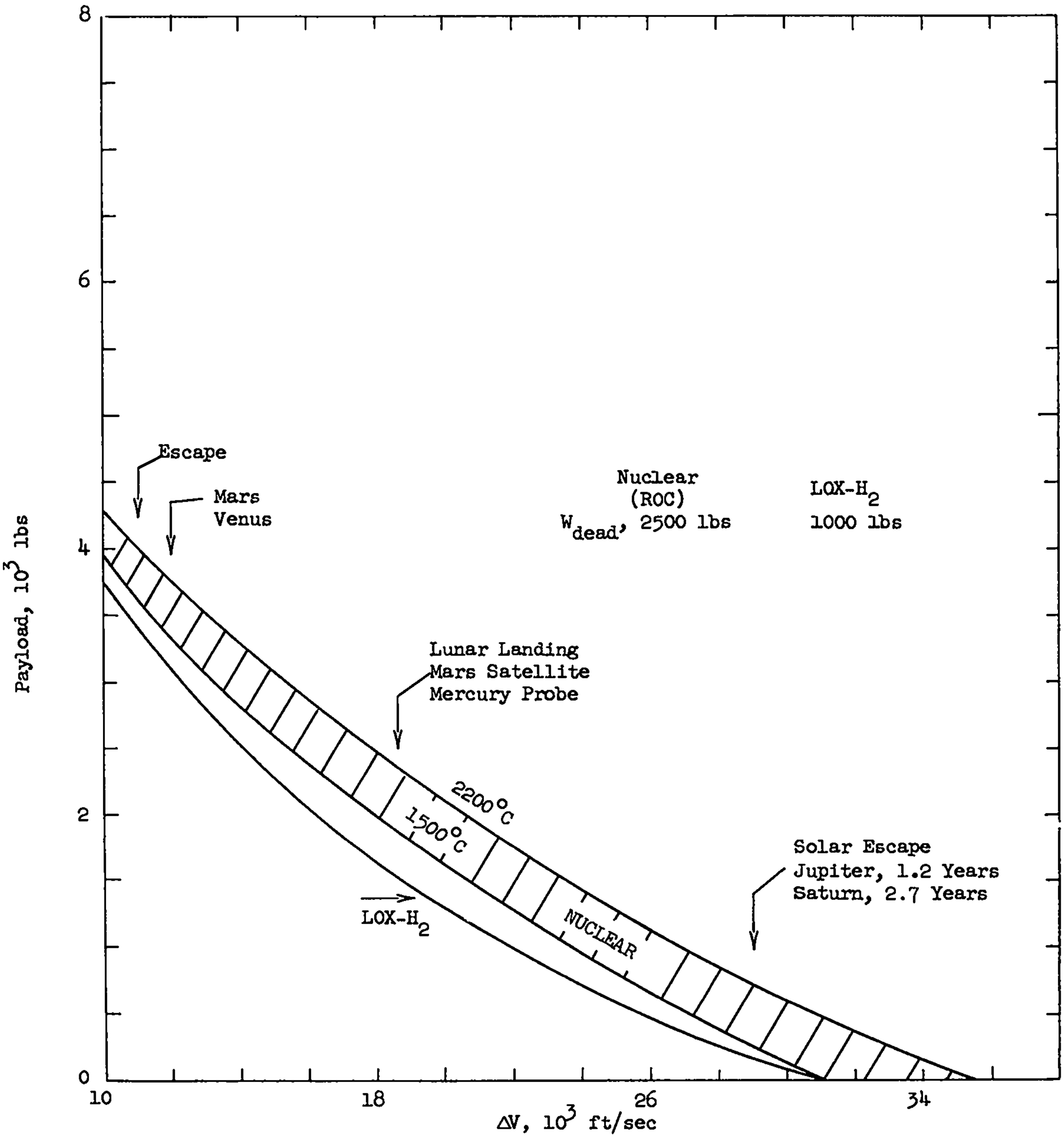

Figure 4 Payloads for 10,000 pound orbital start vehicles

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interest in extensive orbital operations. While very low powers may be of interest in manned interplanetary flight, there the emphasis will be on high performance (I_{sp}) at the expense of reactor weight (and power), and thus a different type of reactor will be desired.

Analogous results (Figure 5) are obtained for the 20,000 pound case. Because the engine weight need not increase much to double the power, the payload advantage over chemical propulsion is clearer than in the 10,000 pound case, but otherwise the same comments apply. Furthermore, the Saturn C1 represents an interim nonoptimized configuration, and unless some new booster of this payload capability is developed, the 20,000 pound orbital stage is not of permanent interest. The 50,000 pound vehicle is of greater interest. Here a lightweight reactor gives considerably better performance than either chemical propulsion or a KIWI-type nuclear engine. Results are presented in Figure 6 and Table I. The latter includes a number of missions as characterized by their velocity increment (from orbit) requirements. Except for the difficult missions ($\Delta V \gtrsim v_e$ = 25,800 ft/sec), the payloads for this size vehicle are not very sensitive to reactor weight or specific impulse and insensitive to power or thrust. Quantitatively, a pound of engine weight saved is a pound of payload earned, and the 6000 pound maximum difference between the KIWI and ROC engines may be considered important, for example, when it doubles the payload, which occurs approximately at $\Delta V = v_e$. The change in payload with specific impulse is ~25 pounds/second, which corresponds to 5 pounds/$^{\circ}$C in the exit gas temperature range to 2500°C

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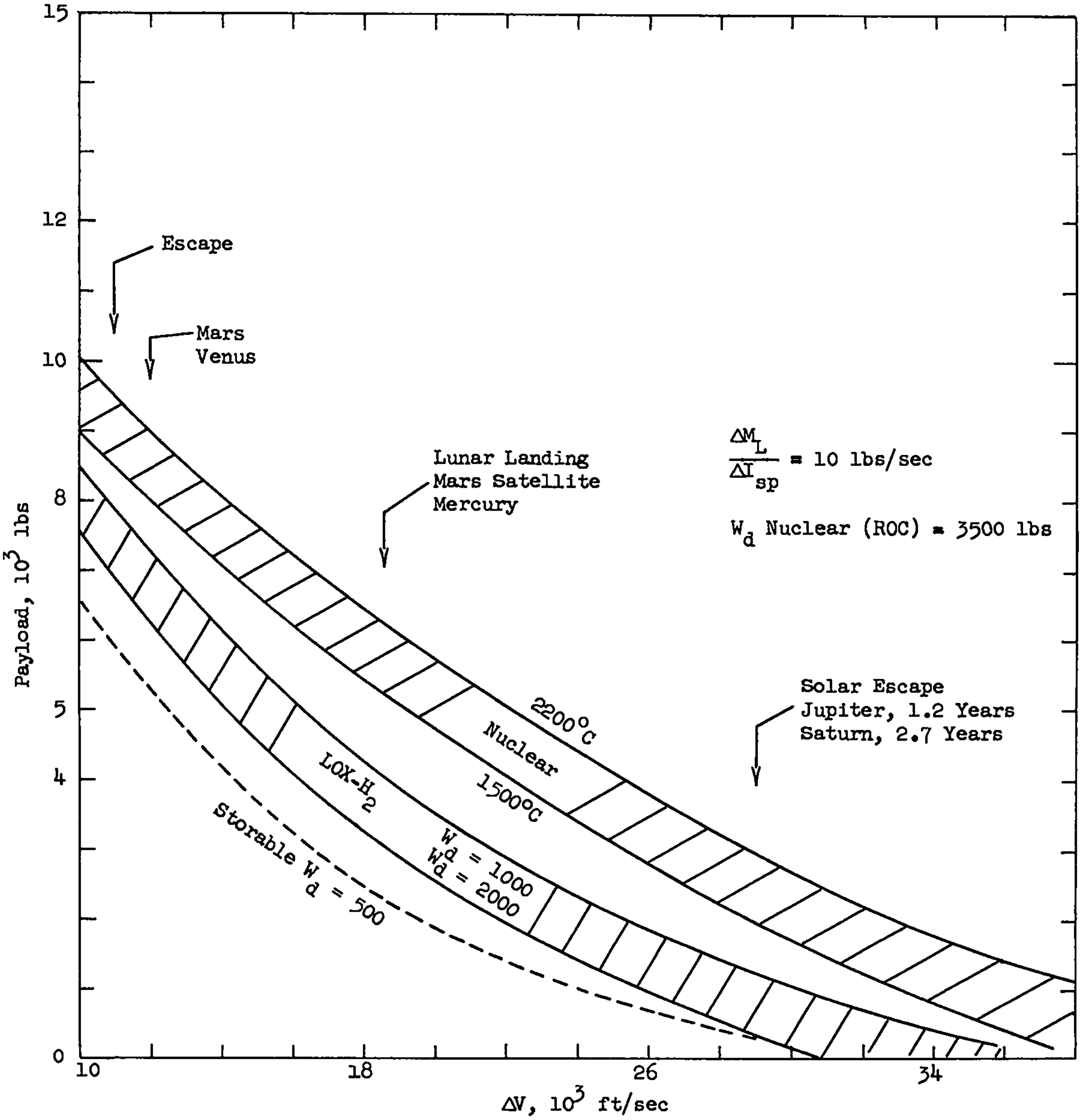

Figure 5 Payloads for 20,000 pound orbital start vehicles

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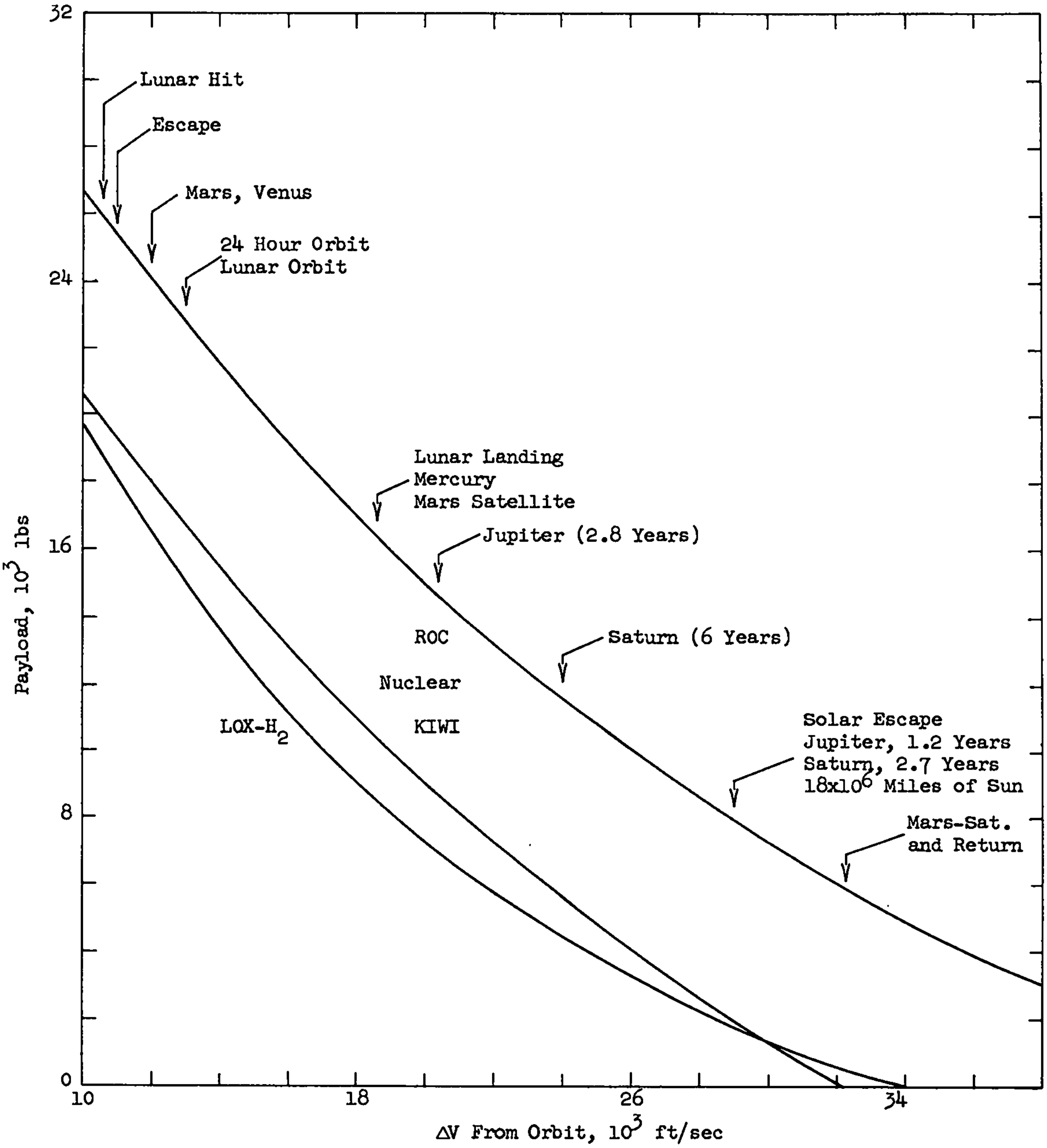

Figure 6 Payloads for 50,000 pound orbital start vehicles

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Table I

Payloads for Orbital Start Stages

Mission	ΔV, ft/sec from orbit	Payloads in 10^3 lbs: LOX-H_2	KIWI, 1000 Mw	ROC, 400 Mw
Lunar hit or pass	10,500	19	20	26
Escape	11,000	18.1	19.3	25.3
Low lunar orbit; 24 hour orbit	13,000	15.0	16.7	22.7
Soft lunar landing	18,600	8.4	10.5	16.5
Probes:				
Venus (min. energy)	11,500	17.3	18.6	24.6
Mars (min. energy)	12,000	16.5	18.0	24.0
Mercury (min. energy); Mars satellite	18,600	8.4	10.5	16.5
Jupiter, 2.8 yrs.	20,500	6.7	8.7	14.7
Saturn, 6 yrs.	24,000	4.4	5.7	11.7
Solar escape; Jupiter, 1.2 yrs.; Saturn, 2.7 yrs.; Solar probe, 18×10^6 mi.	29,000	1.8 ~3 (2 stage)*	1.9 ~3 (2 stage)*	7.9
Mars satellite and return	32,200	< 0 ~1 (2 stage)*	< 0 ~1 (2 stage)*	6.0
Assumptions:				
I_{sp}, sec		416	800	800
Thrust, lbs		40,000	50,000	20,000
Engine weight, lbs		1,000	8,000	2,000
Tanks, insulation, and structure, lbs		1,500	5,000	5,000
Misc. dead wt. (guidance etc.), lbs		1,500	1,500	1,500
Total dead wt., lbs		4,000	14,500	8,500

*Upper stage LOX-H_2 (dead wt. = 2000 lbs)

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(no dissociation). Thus for most of the missions, the effect of exit gas temperature is relatively unimportant (~1000 pounds/200°C). Unless one can get to H_2 dissociation temperatures $\geq$ 3000°C, increasing temperature is a hard way of significantly increasing the payload of this size vehicle, especially if one must increase the engine weight to raise the exit gas temperature. The added payload becomes important only for marginal missions for which other solutions (staging or dropping tankage) should be considered. We are not saying that, in general, higher impulse is not very desirable, but that in this case, its effect is small. For example, consider an extreme case where one achieved 3300°C gas at 100 psi, which would correspond to about 1100 seconds specific impulse. This would increase the payload for most missions about 5000 pounds less the extra reactor weight which might even completely cancel the gain. It is difficult to believe the extra development effort would be justified <u>for this application</u>.

For most unmanned probes, the payload weight is not crucial, especially in the weight range of ~10,000 pounds. Thus for mission velocities up to ~20,000 ft/sec, the KIWI type reactor, already under development, would be adequate as would LOX-H_2 propulsion. The advantage of the lightweight reactor appears for the difficult missions such as fast probes and returnable vehicles. Here the weight is necessary for the requisite guidance and long range communication. Solar probes would require thermal insulation as well. While such missions could serve as a justification for lightweight reactors, similar results could be accomplished with a KIWI powered third stage plus a Centaur fourth stage on the Saturn C2.

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Orbital Ferries

We shall use this term to distinguish between reusable vehicles that carry a payload one way and return themselves to their starting point and those that carry a payload both ways, which we shall call <u>maneuverable satellites</u>. For the first case, consider a vehicle, the ferry, which starts in a low earth orbit and receives payload and fuel which have been placed in orbit. The ferry carries the payload to its destination (e.g., 24 hour earth orbit or lunar orbit, each of which requires ~12,500 ft/sec one way) and uses the remaining fuel to return itself to low earth orbit. We assume that the ferry contains an engine, guidance, and tankage sufficient for holding the fuel for the return trip only. The mass which is placed in low earth orbit (M_o) will contain a useful load (M_u), propellant (M_p), and tankage (M_t) for that propellant. We define

m_f = mass of the ferry

M_o = initial mass in low orbit, excluding the ferry

M_u = "useful" mass in payload in earth orbit

M_p = propellant mass in payload in earth orbit

M_t = fM_p, payload tankage proportional to propellant

M_{Bo} = burnout mass at end of one-way trip

R = mass ratio for one-way trip.

We have

$$\begin{aligned} M_o &= M_u + M_p + M_t \\ &= M_u + (1 + f)M_p \end{aligned} \tag{4}$$

[redacted]

For the return trip, the ferry plus propellant must weigh Rm_f. For the outward trip, the relation

$$M_f + M_o = RM_{Bo}$$

leads to

$$M_o + M_f = R(M_u + fM_p + Rm_f). \tag{5}$$

Eliminating M_p and rearranging gives

$$M_u = \left[\frac{1 - (R - 1)f}{R}\right] M_o - \frac{(R^2 - 1)(1 + f)}{R} m_f. \tag{6}$$

We can assume the ferry mass to be fixed and compute the "useful" load as a function of the mass placed in orbit. For masses up to 100,000 pounds in orbit, a thrust of ~25,000 pounds (~500 Mw) is sufficient for orbital transfer. We will examine the transfer from low earth orbit to 24 hour or lunar orbit using a lightweight reactor, a graphite reactor, and chemical propulsion for orbital masses up to 100,000 pounds and more advanced reactors and LOX-H_2 propulsion for larger loads ($\leq$ 400,000 pounds). Assumptions are given in Table II and results in Figures 7 and 8. We see that nuclear ferries become superior to chemical ones in the orbital mass range of 30,000 to 50,000 pounds and that lightweight reactors offer significant improvement over KIWIs in this range. The velocity requirement for this mission is small, which tends to minimize the advantage of the higher I_{sp} of nuclear propulsion;

[redacted]

Table II

Ferries Up To 100,000 Pound Orbital Mass

Propulsion	ROC	KIWI	LOX-H_2
Engine wt., lbs	2,000	6,000	300
Tankage, lbs	500	1,000	200
Misc. dead wt., lbs	1,000	1,000	500
Ferry mass, lbs	3,500	8,000	1,000
Thrust	25,000	50,000	30,000
Return propellant, lbs	2,000	4,500	1,500
I_{sp}, sec	860	860	420
R	1.57	1.57	2.52
f	.1	.1	.03
M_u	$.6\ M_o-3600$	$.6\ M_o-8200$	$.38\ M_o-2200$

Ferries Up To 400,000 Pound Orbital Mass

	Advanced Technology		
Propulsion	ROC	Advanced	LOX-H_2
Ferry mass	6,000	10,000	2,000
Power, Mw	1,500	2,000	--
Thrust, lbs	80,000	80,000	80,000
I_{sp}	860	1,100	420
R	1.57	1.42	2.52
f	.1	.1	.03
M_u	$.6\ M_o-6400$	$.67\ M_o-7000$	$.38\ M_o-4400$

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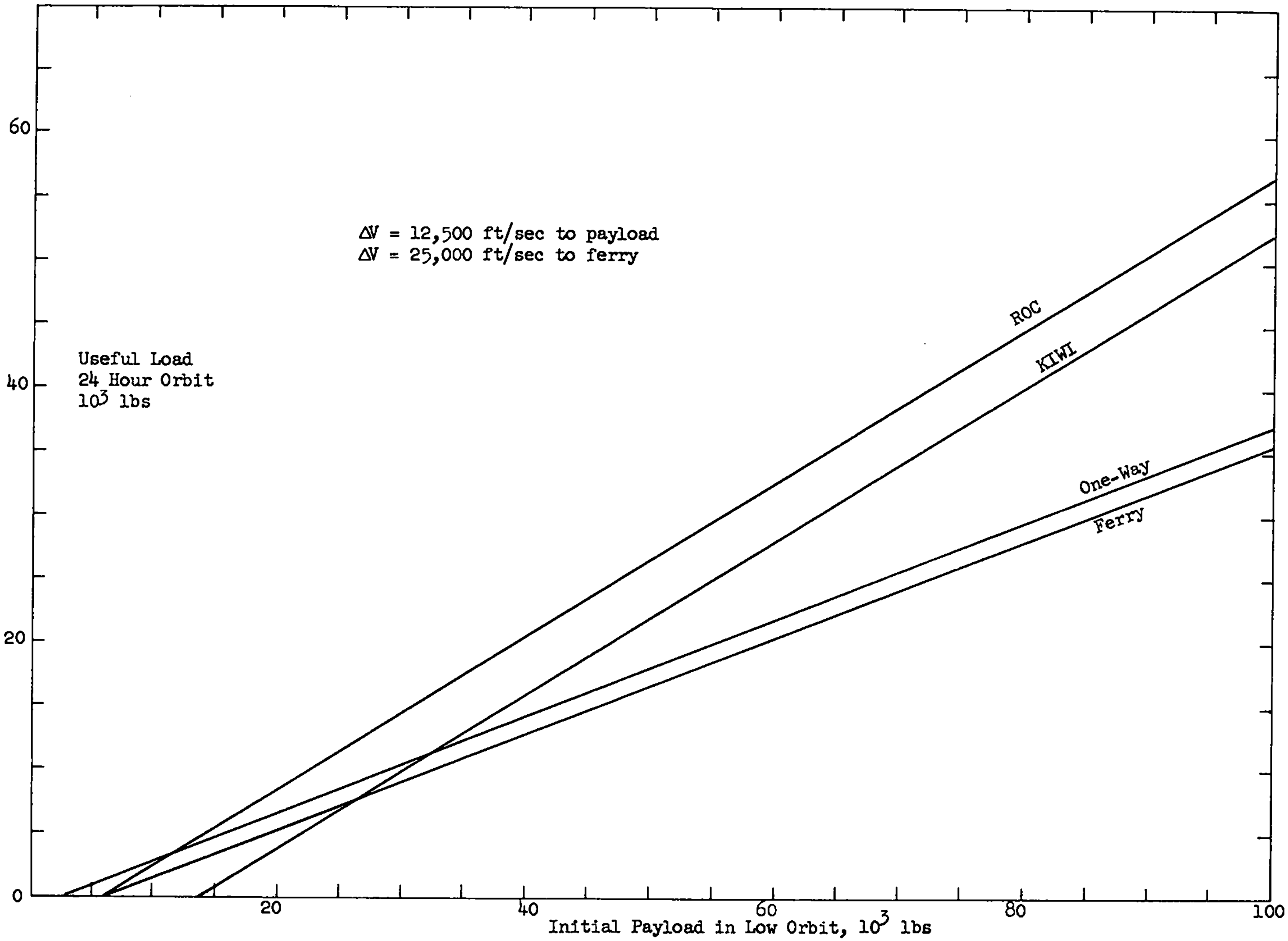

Figure 7 Useful loads for interorbital transfer ferries

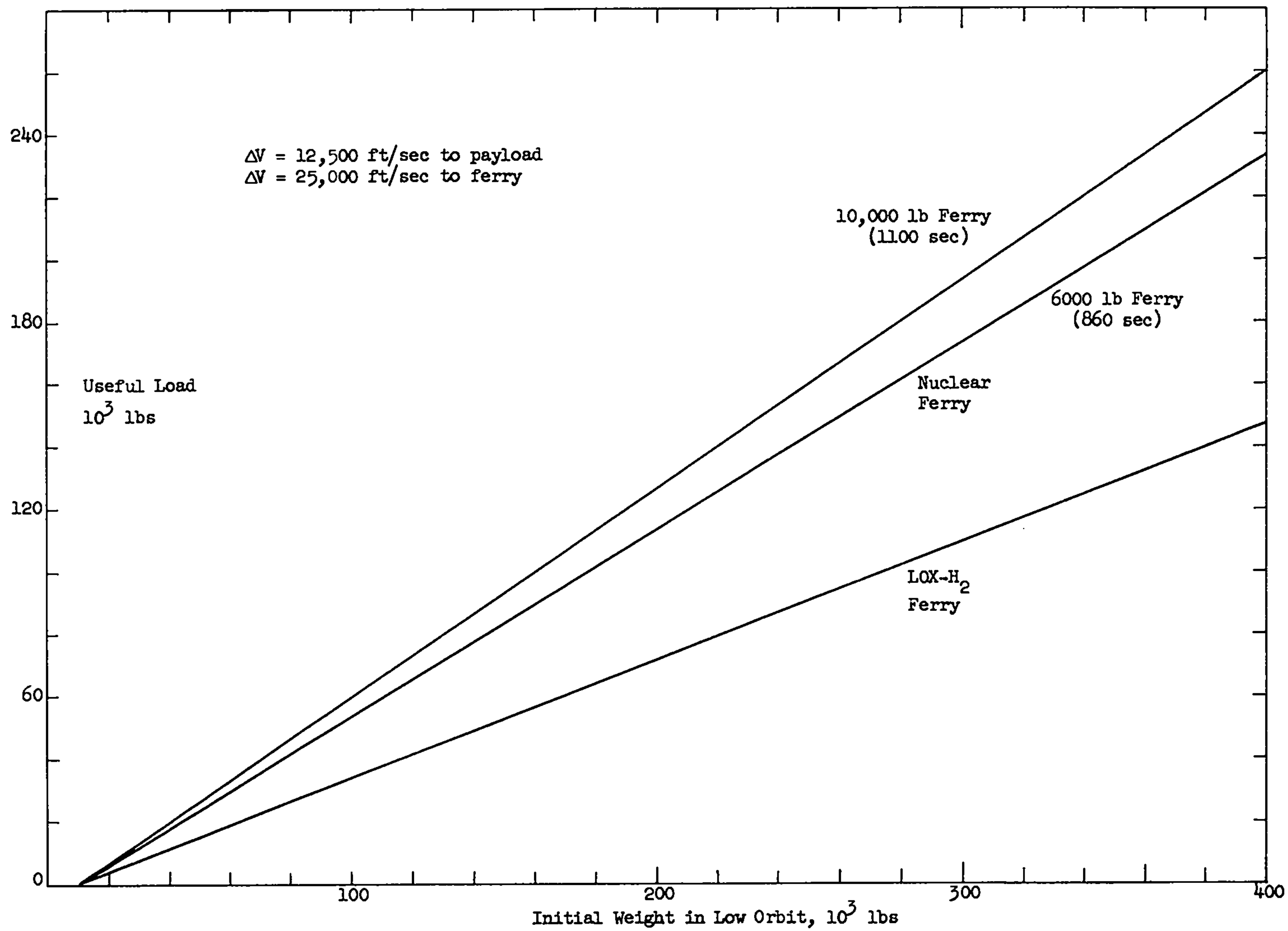

Figure 8 Useful loads for interorbital transfer ferries

and as can be seen from Figure 8, even large increases in I_{sp} give only moderate performance increases. Above 100,000 pounds, the reactor weight becomes less significant, and the useful load for nuclear ferries approaches 160% of that with a chemical ferry. For this mission, reusing the chemical engine is not costly in terms of useful payload as compared to simply leaving it at the terminus.

The situation is somewhat changed by greater mission difficulty as shown by the results for ferries from low earth orbit to the lunar surface and back (ΔV = 20,000 ft/sec each way). We have made similar assumptions for the low orbital masses ($\leq$ 100,000 pounds) and for the larger ones assumed the ferry weight was proportional to the orbital mass. This was to allow for the somewhat higher thrusts required for the lunar landing phase. These values were taken to be 4% for the nuclear ferry and 1% for the LOX-H_2 ferry to match their weights at the 100,000 pound payload. In the low orbital mass range ($\leq$ 100,000 pounds, Figure 9), results are more sensitive to the ferry weight. The KIWI and LOX-H_2 performances are equal at 50,000 pounds, while the lightweight reactor is about twice as good. However, the absolute values of the useful loads are rather small (5000 to 15,000 pounds), becoming of interest for lunar supply for orbital weights of 100,000 or more. For higher powers (Figure 10), the graphite reactors approach and reach the power densities available with fast reactors in the smaller sizes. Then the choice between the two depends on other factors than weight, e.g., maximum temperature, uranium requirements, reusability, shielding requirements, etc.

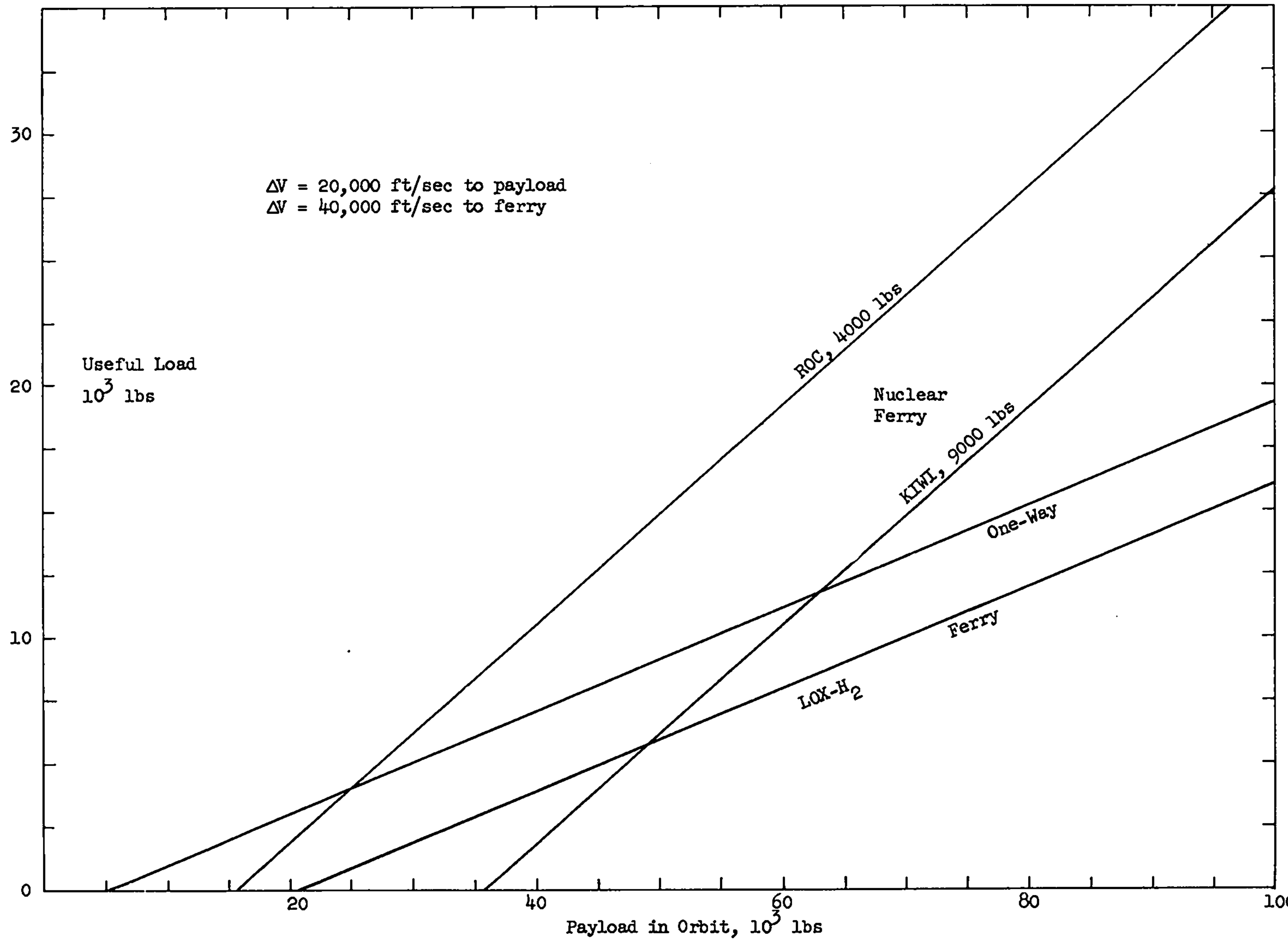

Figure 9 Useful loads for lunar transfer

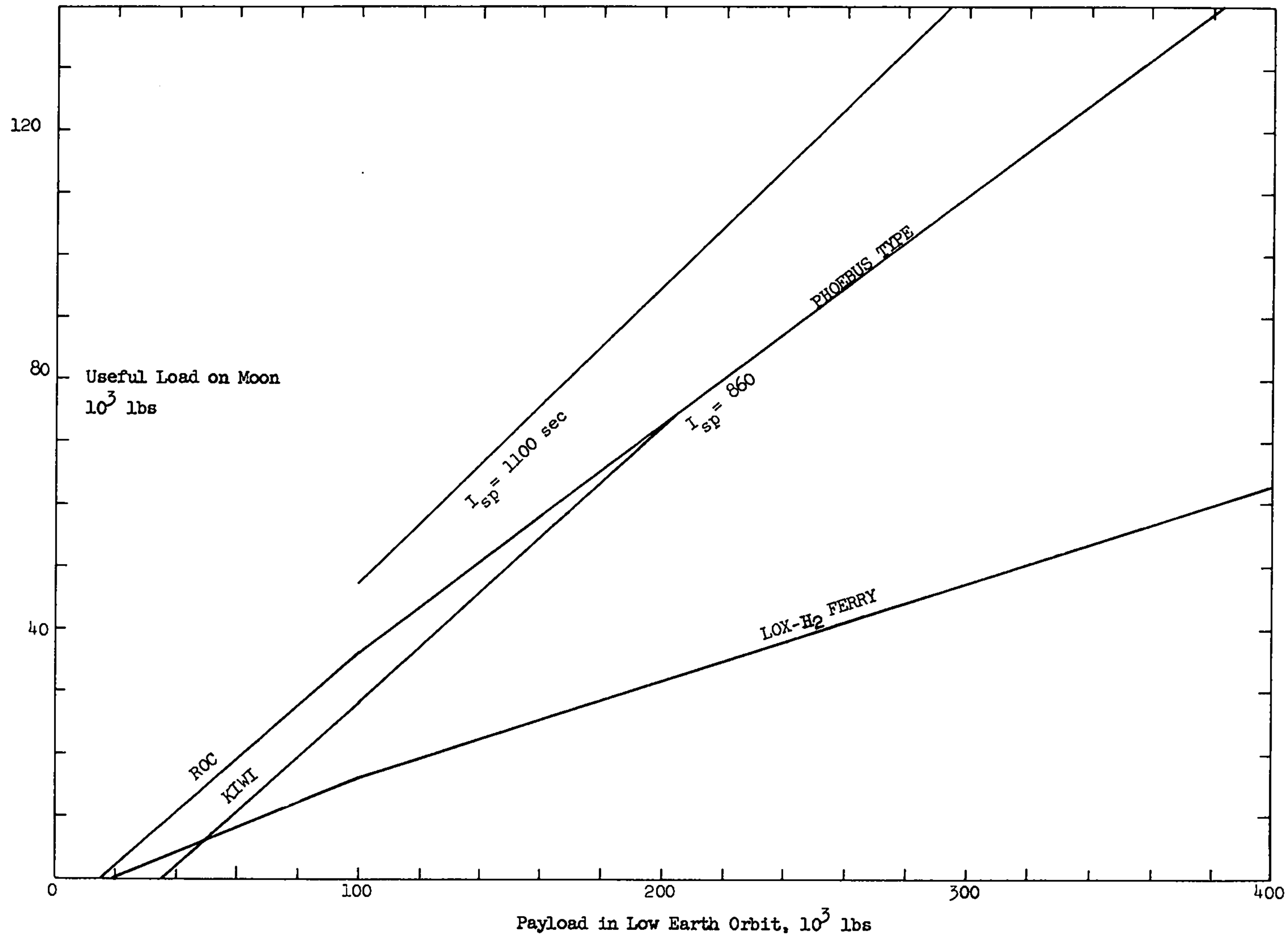

Figure 10 Useful loads for lunar transfer

With larger vehicles and higher mission velocities, the reactor weight becomes less important compared to specific impulse, which can give significant performance increases (~30% in useful load). In the large weight range, chemical propulsion gives only half the useful load possible with nuclear propulsion. Cislunar operations do not represent very difficult missions even for chemical propulsion, and so one can gain factors of only two to four by using nuclear rockets.

Maneuverable Satellites

Next we shall examine maneuverable satellites, which are very similar to the orbital start probes, but with different emphasis, particularly on refuel requirements and payload weight. Thus we choose to fix upon two mission velocity requirements and vary the vehicle mass continuously. The vehicles are assumed to start and return to low earth orbit, carrying the payload mass for the entire trip. We have selected 25,000 ft/sec and 40,000 ft/sec, which could represent numerous transfers among various orbits but also correspond to two interesting cases. The first (25,000 ft/sec) is the approximate requirement for a round trip to a 24 hour orbit, to a lunar orbit, or to escape. The second (40,000 ft/sec) is sufficient for a round trip to the lunar surface or even for a low energy interplanetary reconnaissance round trip.

We are including manned vehicles and thus are considering heavier and more advanced vehicles than in the orbital probe section. For example, for lunar operations, a fair portion of the payload might be required to have a shielding function. Useful material (aluminum sheet and water) could be carried out and ore or soil from the moon used for shielding on the return trip. Thus essentially the full payload mass will be carried both ways. Where the return payload is small, we have the case treated under orbital ferries. As before, we will assume a fixed engine weight (M_e, including miscellaneous items, guidance, etc.) for vehicles up to 100,000 pounds and a linear relation for larger vehicles where the constant is ϵ ($\epsilon = M_e/M_o$). A simple analysis gives the payload

$$M_u = M_o\left[\frac{(1 + f)}{R} - f\right] - M_e \qquad M_o \leq 100{,}000 \text{ pounds}$$

$$= M_o\left[\frac{1 + f}{R} - f - \epsilon\right] \qquad M_o > 100{,}000 \text{ pounds.} \qquad (7)$$

The refuel requirement, including tankage to contain it, is

$$M_{rf} = \left(1 - \frac{1}{R}\right)(1 + f)M_o, \qquad (8)$$

where f is the same value of M_t/M_p as assumed for the orbital vehicle and includes rendezvous and fuel transfer equipment. The tankage could be left in the low earth orbit for use in a space station or might be part of an earth-to-orbit shuttle. In any event, we charge this tankage to the refueling operation. Later we shall examine the gains possible

[redacted]

by tank staging and use of the refuel tankage. Parameter values are listed in Table III with the specific forms of Eqs. (7) and (8).

Results are presented in Figures 11 and 12, giving payload vs. initial weight in orbit. These demonstrate a considerable advantage for nuclear propulsion over LOX-H_2 but do not show the important effects of refuel requirements. To illustrate this, let us consider the refuel requirements for a 25,000 pound payload satellite for the 25,000 ft/sec mission. The results (Table IV) show the nuclear stage requires only 20% to 40% as much support weight (fuel + tankage) as the chemical system. Alternatively, one might ask what size satellite could be refueled by the Saturn C2 (50,000 pound orbital payload). The answer (Table V) again shows advantages for the nuclear system, particularly considering a minimum payload of considerable size (> 10,000 pounds) would be required for manned satellites which pass through the radiation belts.

The more difficult mission (40,000 ft/sec) is a marginal one for single stage LOX-H_2 or 860 sec nuclear rockets, where the payloads represent 1% and 10% of the gross weight, respectively, which are less than or equal to the tankage weights. Since one must carry the refuel propellant to orbit in tanks, one might stage the tanks during the trip and replace them for the next trip. For tank staging in n equal steps

$$M_u = M_o\left[\frac{(1+f)}{R^{1/n}} - f\right]^n - \epsilon M_o. \qquad (9)$$

[redacted]

[redacted]

Table III

Maneuverable Satellites -- Assumptions and Results

	$LOX\text{-}H_2$	Nuclear	Nuclear
I_{sp}, sec	420	860	1100
M_e (minimum), lbs	1000	3000 (ROC) 7000 (KIWI)	8000
ϵ	.01	.03	.04
f	.03	.10	.10
ΔV = 25,000 ft/sec			
R	6.38	2.47	2.03
M_u	$.132\ M_o - 1000$ $.122\ M_o$	$.345\ M_o - 3000$ $.315\ M_o$	$.442\ M_o - 8000$ $.402\ M_o - 4000$
M_{rf} (refuel)	$.867\ M_o$	$.655\ M_o$	$.542\ M_o$
ΔV = 40,000 ft/sec			
R	19.4	4.24	3.1
M_u	$.023\ M_o - 1000$ $.013\ M_o$	$.160\ M_o - 3000$ $.130\ M_o$	$.225\ M_o - 8000$ $.215\ M_o - 4000$
M_{rf} (refuel)	$.975\ M_o$	$.84\ M_o$	$.745\ M_o$

[redacted]

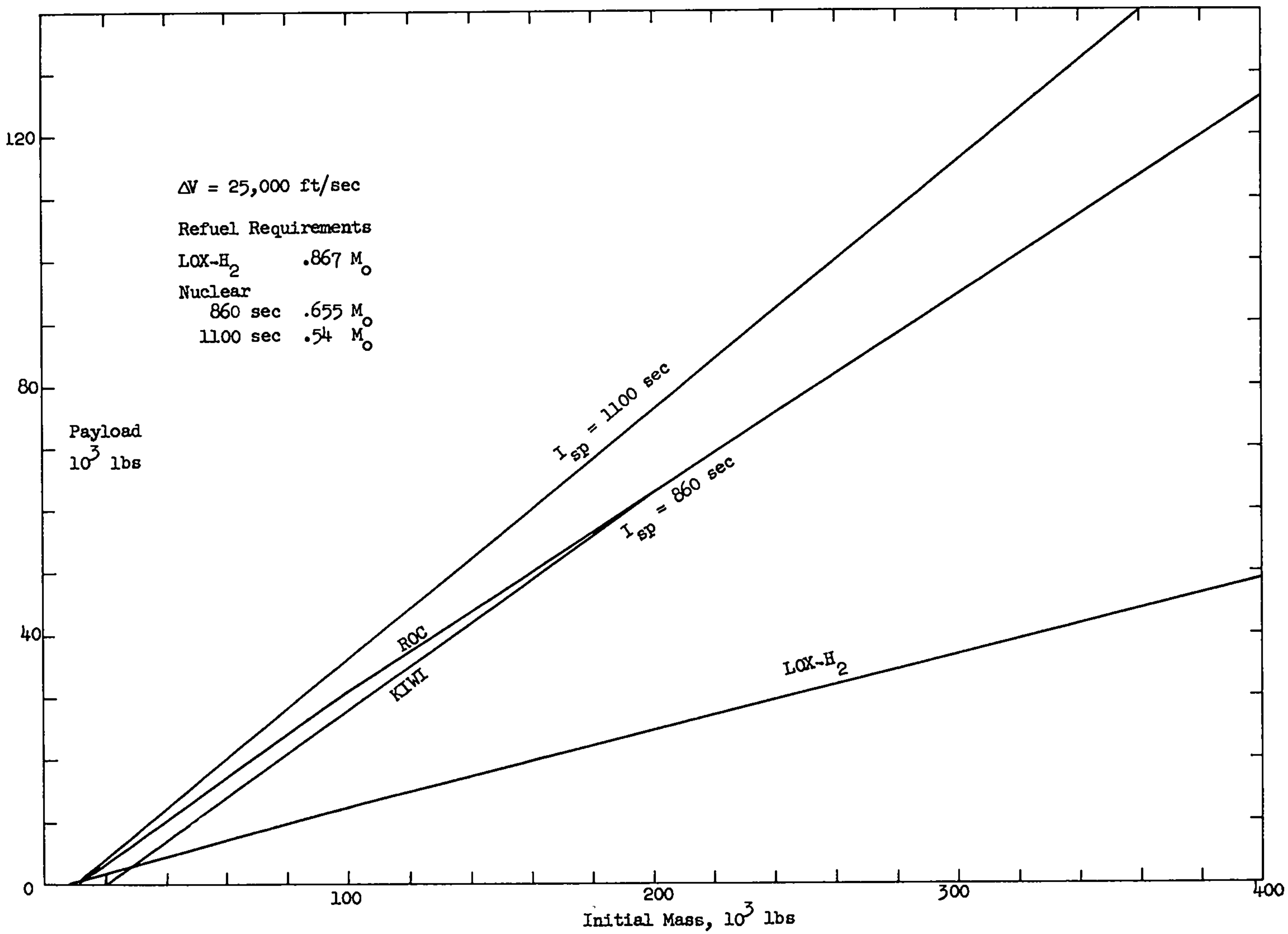

Figure 11 Maneuverable satellites

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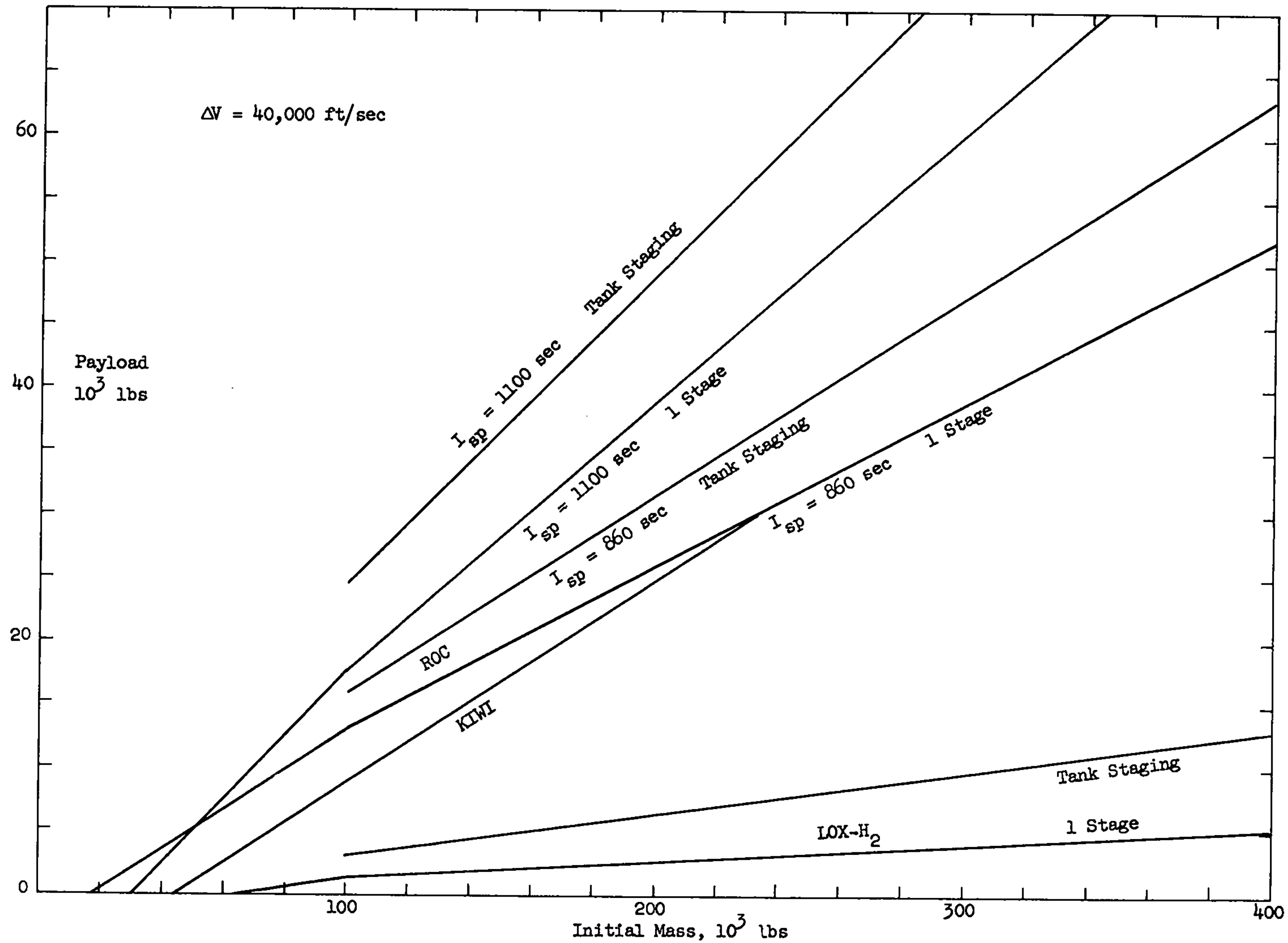

Figure 12 Maneuverable satellites

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Table IV

Refuel Requirements, 25,000 Pound Payload

ΔV = 25,000 ft/sec	LOX-H_2	Nuclear (KIWI)	Nuclear (1100 sec)
M_o, lbs	205,000	92,000	72,000
M_{rf}, lbs	178,000	60,000	39,000

Table V

Satellite Supplied by Saturn C2

	LOX-H_2	Nuclear (KIWI)	ROC (860 sec)	Nuclear (1100 sec)
Satellite gross wt. (M_o)	57	76	76	93
Payload (M_u)	7	20	23	33

(ΔV) satellite = 25,000 ft/sec

[redacted]

This has a limit for $n \to \infty$, continuous tank staging.

$$M_n = M_o \left[\frac{1}{R^{1+f}} - \epsilon \right] . \tag{10}$$

The results (Table VI) show that payload increases of 30 to 200% can be obtained by tank staging, that both the LOX-H_2 and nuclear systems are substantially improved, and that most of the effect is obtained with one staging. The chemical system is closer to its limiting velocity and thus gains a larger fraction of its very small payload, while the nuclear system gains more on an absolute basis because of its heavier tankage. Typical examples are shown in Table VII, including the effect upon refuel requirements.

Table VI

Effect of Tank Staging, ΔV = 40,000 ft/sec

	Payload Fraction	
	LOX-H_2	Nuclear, 860 sec
No staging	.013	.130
$n = 2$	.032	.158
$n = 4$	.035	.168
$n = \infty$	.037	.174

Finally, in Figure 13, we show the payload fraction vs. velocity requirement with and without tank staging.

[redacted]

Table VII

Tank Staging -- Examples

$\Delta V = 40{,}000$ ft/sec $M_o = 100{,}000$ Pounds

Tanks Staged at $\Delta V = 20{,}000$ ft/sec

Masses in 10^3 Pounds

	LOX-H_2	Nuclear, 860 sec	Nuclear, 1100 sec
M_u	3.2	15.8	24.5
M_p	93	73.8	65.0
M_e	1.0	3.0	4.0
M_t	2.8	7.4	6.5
M_t dropped	2.32	5.15	4.3
Fraction dropped	.83 (~5/6)	.70 (~2/3)	.66 (~2/3)

25,000 Pound Payload

40,000 ft/sec

		LOX-H_2	Nuclear (KIWI) 860 sec	Nuclear 1100 sec
One stage	M_o, 10^3 lbs	1,920,000	200,000	134,000
	M_{rf}, 10^3 lbs	1,870,000	168,000	100,000
Single tank staging	M_o, 10^3 lbs	780,000	158,000	102,000
	M_{rf}, 10^3 lbs	750,000	127,000	73,000

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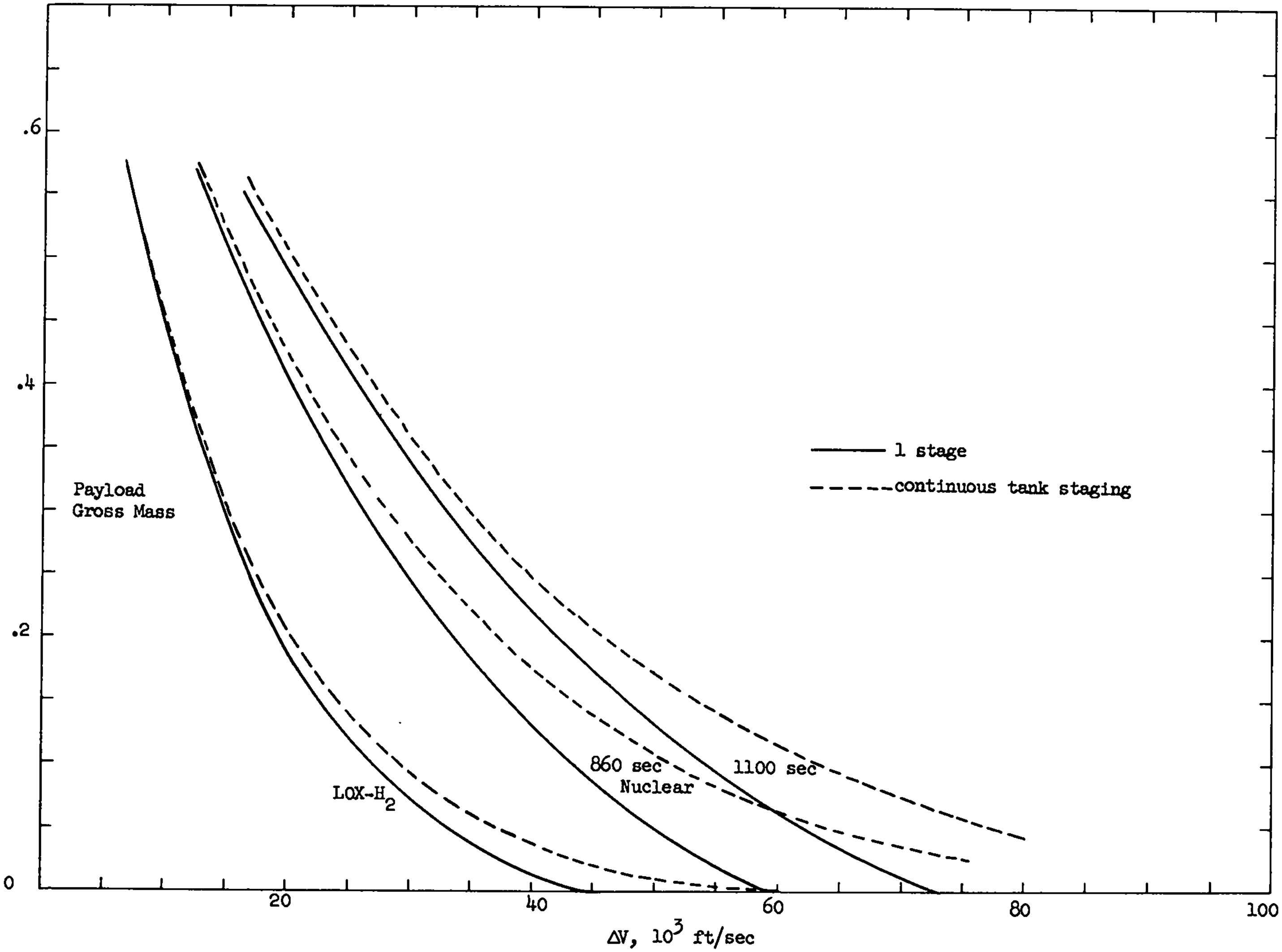

Figure 13 Payload fraction vs. velocity requirement

There is a possible application for low power reactors in large interplanetary expeditions even where the gross vehicle weight in orbit is 10^6 pounds or more. While a large, high thrust engine is desirable for the earth escape phase to reduce gravitational losses and minimize time in the radiation belts, once this is accomplished, much smaller thrusts are sufficient.

One might then want to drop the large engine and use a smaller, shielded engine for the rest of the journey. Such a case was considered by Ehricke* in which a 170 Mw second stage engine was used for orbital reconnaissance of Mars and Venus. For this application, which is probably at least ten years away, the small nuclear heat exchanger may face serious competition from electrical propulsion (ion, plasma, etc.) which has high I_{sp} but very low thrust (~10 to 100 pounds). Other uses, such as for small rescue vessels, may appear with the further development of space activities.

Small Suborbital Stages

Now let us examine the possibility of nuclear second stages on ICBM class boosters. Some of the smaller nuclear stages, while competitive on a payload basis, would be primarily for development and testing purposes. The limiting stage weight would probably be determined by propellant volume considerations due to the 10 foot diameter of present

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*Convair Astronautics Report AZM-072 (March, 1959).

boosters. This diameter allows only 3400 pounds H_2/ft of tank length which can lead to excessive L/D ratios for the vehicle. Larger diameter upper stages are possible but might lead to aerodynamically unstable vehicles. Typical examples are presented in Table VIII (together with similar all-chemical rockets) for upper stages of 15,000 to 60,000 pounds. All the ROC powered stages yield larger payloads than equivalent chemical stages, which is also true of the larger ($\geq$ 50,000 pound) KIWI powered stages. In these examples, we have chosen lower performance (I_{sp}) in order to have smaller (ROC) engine weight in the small stages where reactor weight is more significant. From the equations for exchange ratios:*

$$dM_L \simeq \frac{M_o}{eI_{sp}} dI_{sp} - dM_e$$

$$\simeq \frac{M_o}{2000} dI_{sp} - dM_e, \qquad (11)$$

we can see that 1000 pounds of engine weight is more valuable than 100 sec of impulse for stage weights up to 20,000 pounds. The lower performance (I_{sp}) engines would probably be more readily attainable even with lower engine weights. The choice is not a crucial one and would affect the payloads only by ~ ± 10%. On the other hand, there is a significant difference (~5000 pounds) between the lightweight engine and KIWI powered stages, particularly where payloads are small (small stages or difficult

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*R. Cooper, "Mission Studies for Nuclear Heat Exchanger Rockets," Los Alamos Scientific Laboratory Report LAMS-2512 (December, 1960).

Table VIII

Nuclear Stages Upon ICBM Boosters

	Upper stage weights, lbs					Payloads	
	Gross	Total Inert	Engine + Misc.	Propellant	Tank	Low Orbit	24 Hr. Orbit
Atlas-Agena B	15,000	1,000				5,300	~700
Atlas-ROC, 200 Mw	15,000	3,000	2,000	4,500	1,000	7,500	
I_{sp} = 700 sec, 1500°C		4,000	2,000	9,000	2,000		2,000
Atlas-Centaur	30,000	3,000				8,700	1,300
Atlas-ROC, 600 Mw	30,000	6,000	3,500	12,000	2,500	12,000	
I_{sp} = 775, 2000°C		7,000	3,500	19,000	3,500		4,000
Atlas-KIWI	(W engine ~6000 lbs greater than ROC)					6,000	1,400*
Titan A, 226,000 lbs	50,000	3,000				4,500	~1,000*
Titan A-ROC, 1200 Mw	50,000	8,000	4,000	26,000	4,000	16,000	
I_{sp} = 800, 2200°C		9,500	4,000	35,300	5,500		5,200
Titan A-KIWI	(W engine ~5000 lbs greater than ROC)					11,000	2,500*
Titan B, 319,000 lbs	60,000	3,000				7,500	~1,700*
Titan B-Centaur, 2 stg.	60,000	5,000				11,500	1,500
Titan B-ROC, 1500 Mw	60,000	9,500	4,500	29,500	5,000	21,000	
I_{sp} = 800, 2200°C		11,000	4,500	41,200	6,500		7,800
Titan B-KIWI	(W engine ~5000 lbs greater than ROC)					16,000	2,800

*Three stages, two-stage payload negative.

[redacted]

missions). One case, Titan B-ROC, offers performance equal to that of the Saturn C1 with one-third the gross weight and one-fifth the manufactured weight.

Stages of this power level are also of interest as third stages on the Saturn C2. This case is being studied in detail for the KIWI engine by the NASA Rift Study Contractors and the NASA Marshall group and thus will not be discussed here except to indicate an increase in payload of ~5000 pounds for small ($\leq$ 100,000 pound) stages by the use of a fast reactor.

Shielding of Radiation

There are many problems associated with radiation, natural and reactor produced, but we shall touch upon only one or two. Radiation heating of the hydrogen propellant has long been considered a problem, but recent estimates indicate that relatively little shielding would be required on this account. Results are dependent upon tank shape and placement relative to the reactor, pump cavitation characteristics, and reactor details. However, for simple estimates, the reactor may be considered as a small volume source with a shadow shield (taken to be about same diameter as the reactor) placed between the reactor and tank. This shield will also serve to protect the payload, of particular importance when the propellant is nearly exhausted and if the vehicle is manned. If one hopes to use the vehicles for manned operations (such as landing vehicles or rendezvous craft) where the crew would occasionally want to

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leave their (presumably shielded) quarters, then additional shielding might be necessary even long after shutdown of the reactor. To illustrate this, we shall use an example of an earth orbit to lunar landing nuclear stage. There will be two propulsive phases, earth escape and lunar landing, which can be assumed to be impulsive. They will occur about two days apart, which is the transit time for low energy trips. Assume the gross weight of the ship to be 150,000 pounds of which 80,000 pounds are propellant. The escape phase will consume 50,000 pounds and the landing phase 30,000 pounds. Data from the KIWI A test indicated doses of ~400 r/hr. at 20 feet from the reactor one hour after shutdown, due primarily to fission product activity in the core. This dose rate can be given in terms of time, distance, and total energy release as

$$\text{D.R.} \sim 4t^{-1.2}\, r^{-2}\, E \text{ r/hr.}, \tag{12}$$

where t is time from shutdown in hours, r is distance in feet, and E is energy release in Mw sec. One Mw sec is the energy requirement for ~0.07 pounds of H_2 propellant, which can also be used as a base for computing the dose rate. The dose rate history (for a point 20 feet from the reactor) is given in Figure 14. During the outward passage, the dose rate decays from ~7000 r/hr. one hour after escape to ~70 r/hr. just prior to landing. The landing phase again raises the radiation field to ~4000 r/hr. (~1 r/sec) at one hour after landing. This creates a problem in descending from the cabin to the lunar surface, which extends for quite a time as can be seen from Table IX.

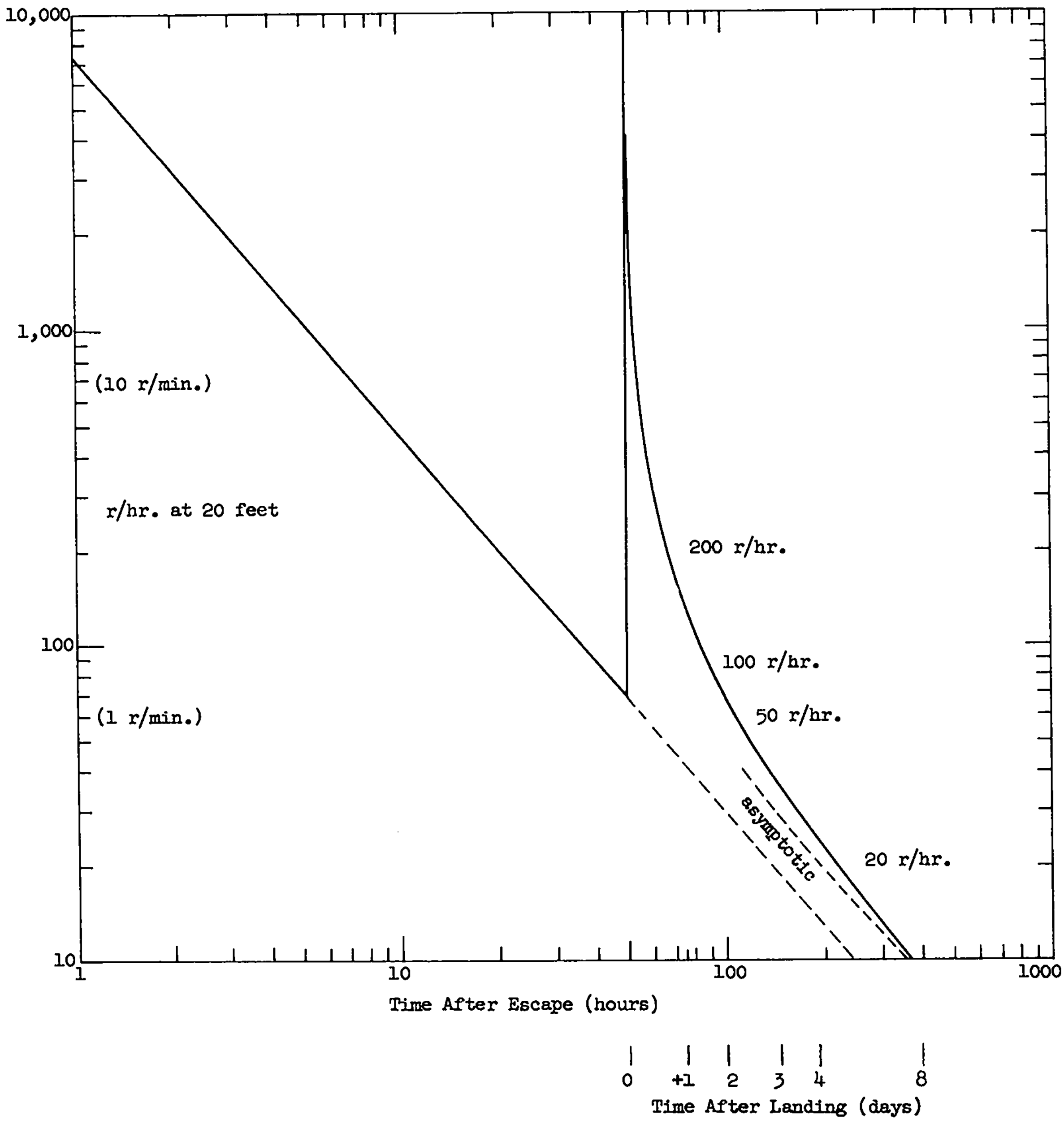

Figure 14 Dose rate history of a lunar flight (20 feet from reactor)

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Table IX

Dose Rate 20 Feet From Reactor

Time	Dose Rate, r/hr.	r/min.
1 hour	4000	70
10 hours	300	5
1 day	130	2
2 days	70	1
4 days	23	0.4

Thus one might want to shield the sides of the reactor as well, and the physical size of the reactor can become significant. Fast reactors, besides being lighter than graphite moderated reactors, are much more dense and, therefore, are considerably smaller as shown by Table X. Also given are representative shield weights for 10 inch shadow shields at the reactor end (attenuation ~5000) and for a 5 inch circumferential shield (attenuation ~70), assuming reactor length equals reactor diameter. The 10 inch shadow shield has been estimated to be sufficient (though perhaps not entirely necessary) for protection of passengers. The peripheral shield (or an angular segment of it) might be necessary if the reactor had to be approached if only to 20 feet, e.g., in disembarking or rendezvous operations. Approaching this close to reactors operating at any but very low powers (1 Kw gives 10 r/hr.) would be impossible without very heavy shields. Table X indicates that if thick shields are necessary, reactor size can be crucial for small size vehicles. For orbital operations

[redacted]

where accelerations and stresses are low, the vehicle can be shaped to minimize shield requirements. Fast reactors may use heavy element reflectors (such as Ni) which can also act as shields with greater effectiveness per unit weight because of the smaller radius possible.

Table X

Reactor Sizes and Representative Shield Weights

	Sizes (inches)		Weights (pounds)		
				Lead Shields 11 gm/cc	
Reactor	Core Diameter	Reflector Diameter	Reactor	10 Inch Shadow	5 Inch Circumferential
200 Mw UC	10 inches	18 inches	300	1250	2,700
1000 Mw UC-ZrC	18 inches	26 inches	1500	2600	5,200
1000 Mw Graphite (KIWI)	35 inches	50 inches	7000	9600	17,600

Discussion and Summary

A wide variety of missions for low power nuclear rocket engines have been examined. In most cases, the KIWI engine powered stages are competitive (on a payload fraction basis) with LOX-H_2 stages for gross weights of 50,000 pounds and show distinct advantages (e.g., doubling the payload) above 100,000 pound stage weights. The development of small, lightweight engines (e.g., the fast reactor) could extend the region of applicability of nuclear propulsion to much smaller stages (10,000 pounds) and greatly increase the advantage of nuclear over chemical propulsion for stages in

[redacted]

the 30,000 to 100,000 pound class. It has been shown that for small stages, engine weight is relatively more important than specific impulse, and thus low power engines with specific impulses of ~700 seconds or less can perform significant missions. They could be of use in attacking those many operational problems which are independent of the specific impulse, e.g., radiation, ground handling, flight control, etc., using much cheaper stages and boosters. The author personally believes that a small ($\leq$ 1000 pound), low power (~200 Mw), pure UC core reactor engine, designed to give exit gas at 1200 to 1500°C (I_{sp} = 630 to 700 seconds) could be developed quite rapidly with a determined effort. A tungsten or graphite support plate could be used to relieve the UC of structural duties. A Be reflected UC core reactor with 30% void volume would weigh only about 200 pounds. The low power and temperature would relax the requirements on components such as pumps and nozzles to where standard components (e.g., the LR-115 H_2 pump) could be used. Also, repeated runs with a single core should be practical, speeding the testing phase. Changes in core size and fuel element material (e.g., to UC-ZrC solid solutions) would be natural developments to achieve a higher performance and higher power engine. The desirability of such a device and program is much less clear than its technical feasibility. The engine weight advantage of fast reactors over graphite moderated reactors becomes less important for power levels over 2000 Mw (thrusts of 100,000 pounds) or stage weights much over 100,000 pounds. However, other factors such as shielding or a particularly desirable vehicle combination (e.g., Titan B-ROC) for a special

[redacted]

purpose etc. could significantly affect the desirability of developing a new reactor type. Because development costs are so high, even to obtain a single device, one might prefer using an existing reactor type in a non-optimum configuration. This leads to economic and over-all planning questions which are beyond the scope of this report.

For the larger orbital vehicles, nuclear propulsion in any form is distinctly superior to chemical propulsion, particularly for difficult missions. The cases involving repeated refueling in orbit also show a great advantage for nuclear propulsion. Such operations should become quite common and important when space activities are extensive and should also prove useful in the early periods of manned exploration, as refueling is no doubt simpler than assembly in orbit.

[redacted]

APPENDIX

THE EFFECT OF THRUST/WEIGHT RATIO UPON ORBITAL-START VEHICLES

To illustrate this effect, we shall use the results of Brueckner[*] for orbital take-off with thrust parallel to velocity.[**] In order to use his results directly, we will assume the exhaust velocity to be equal to the initial orbital velocity (v_o = 25,200 ft/sec) which corresponds to I_{sp} = 782 sec or ~2000°C exit gas. The results are a function of the final energy, and two cases have been computed. They correspond to escape with zero final kinetic energy (ΔV = 10,400 ft/sec) and with final energy $1/2\ mv_o^2$ (ΔV = 18,400 ft/sec). We shall consider a 50,000 pound vehicle with dry weights equal to those assumed previously for such probes (i.e., 14,500 pounds for KIWI and 8,500 pounds for ROC powered stages). We shall neglect the variation of reactor weight with power, which would be only ~1000 pounds over the entire power range from 0 to 1000 Mw, but will indicate its effect later. The results (Table A1 and Figure A1) show that significant losses occur when T/W_o drops below 0.2.

[*] "Topics on Thrust and Orbit Optimization" by G. Bell et al., unpublished work.

[**] This is close to the optimum case with variable thrust direction.

[redacted]

Table A1

Payload vs. Reactor Power for a 50,000 Pound Stage

Power	T/W	Mass Ratio	M_{BO}	KIWI M_d = 14,500 lbs	ROC M_d = 8500 lbs	Equivalent vel. loss ft/sec
∞	∞	1.514	33	18.5	24.5	0
1000	1	1.515	33	18.5	24.5	20
500	.5	1.52	32.9	18.4	24.4	100
300	.3	1.535	32.5	18.0	24.0	350
200	.2	1.55	32.2	17.7	23.7	600
150	.15	1.57	31.8	17.3	23.3	900
100	.1	1.64	30.4	15.9	21.9	2,000
50	.05	1.82	27.4	12.9	18.9	4,600
10	.01	2.03	24.6	10.1	16.1	7,300
0	~0	2.72	18.4	3.9	9.9	13,800
LOX-H_2	1	2.18	23	19		

ΔV = 18,400 ft/sec, Mercury Probe, Fast Martian Probe

Power	T/W	Mass Ratio	M_{BO}	KIWI M_d = 14,500 lbs	ROC M_d = 8500 lbs	Equivalent vel. loss ft/sec
∞		2.08	24.1			0
1000	1	2.10	23.8	9.3	15.3	250
500	.5	2.14	23.4	8.9	14.9	700
300	.3	2.20	22.7	8.2	14.2	1,300
200	.2	2.28	21.9	7.4	13.4	2,300
150	.15	2.38	21.0	6.5	12.5	3,400
100	.1	2.61	19.1	4.6	10.6	5,700
50	.05	3.08	16.2	1.7	7.7	9,800
10	.01	3.78	13.2	< 0	4.7	15,000
LOX-H_2	1	3.90	12.8	8.8		

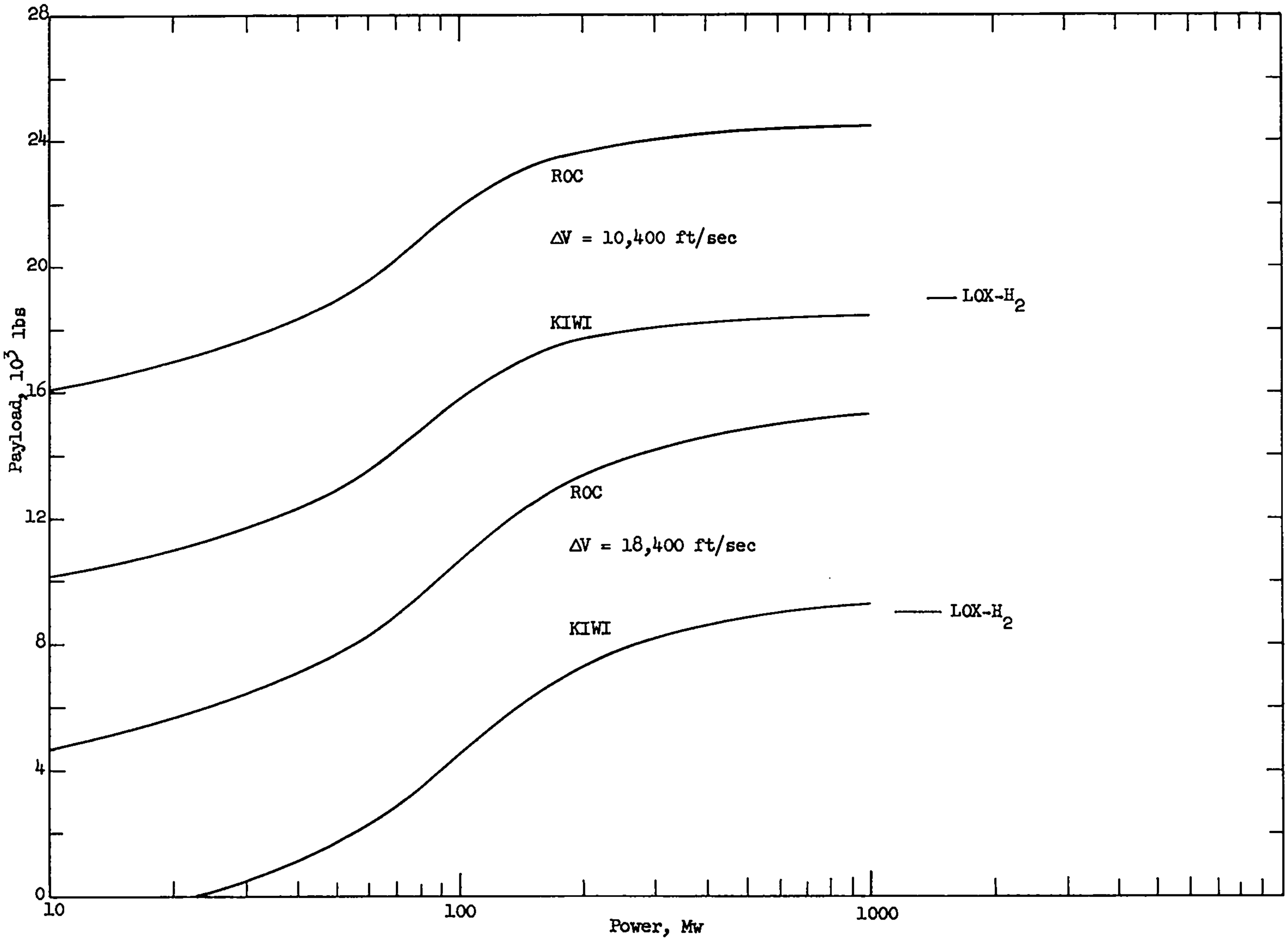

Figure A1 Payload vs. reactor power

In fact, if we allowed engine weight to change 1 pound/Mw, the payload would show a maximum (at ~T/W = .3 for escape and .5 for ΔV = 18,400 ft/sec). Thus values of T/W in the neighborhood of 1/4 are adequate for most missions. Equivalent velocity losses are included in Table A1 and apply to any size vehicle. For vehicles making additional maneuvers after the earth escape phase (e.g., a Martian round trip), the subsequent losses will be much smaller because the T/W_o has increased (W has decreased from propellant consumption) and the gravitational effects (e.g., at high orbit, the moon or Mars) are usually smaller. For example, if one leaves earth orbit with 0.2 g_e acceleration, one would land on the moon with a final acceleration of 0.4 g(earth) or ~2.5 g(moon) which is quite adequate. Brueckner examined a lowering of the reactor pressure (and power) at higher temperatures (3000^o to 4500^oK) where the effect of hydrogen dissociation is significant and found similar payload maxima at even lower T/W ratios.