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COMPARISON OF CHEMICAL AND NUCLEAR PROPULSION FOR LUNAR AND CISLUNAR TRANSPORTATION SYSTEMS

Reinald G. Finke
Robert C. Oliver

October 1970

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SYMBOLS AND ABBREVIATIONS

EOS	earth-to-orbit shuttle
g_c	32.17405 lbm-ft/lbf-sec^2
I_{sp}	specific impulse, lbf-sec/lbm or "sec"
I_{vac}	vacuum specific impulse, sec
LEO	Low earth orbit
LMSC	Lockheed Missiles and Space Co.
LOR	Lunar-orbit rendezvous
McDD	McDonnell Douglas Astronautics Co.
M.R.	mixture ratio
NERVA	nuclear engine for rocket vehicle application
NR	North American Rockwell Corp.
OOS	orbit-to-orbit shuttle
P_c	chamber pressure, psi
R	stage mass ratio, initial weight to burnout weight
T	thrust, lbf
W_{inert}	inert weight, lbm
W_P or W_{prop}	propellant weight, lbm
W_o	initial weight, lbm
ΔV	velocity increment, fps
ΔV Excess	velocity increment above escape velocity, fps
ϵ	exhaust area/throat area

I. SUMMARY AND CONCLUSIONS

Payload capabilities and operational transportation costs of alternative orbit-to-orbit vehicle concepts are examimed for various lunar and cislunar missions when coupled with a reusable earth-to-orbit shuttle (EOS). Correlative rather than detailed design procedures are used. Chemical propellants considered are F_2/H_2 and O_2/H_2; nuclear propulsion assumes use of the NERVA engine. The performance tradeoff is thus between the low specific impulse (460-480 sec) but light weight (thrust/weight of ~ 50) of advanced chemical rocket engines and the high specific impulse (825 sec) but heavy weight (a 23,000-lb goal for 75,000-lb thrust) of the nuclear engine. Cost tradeoffs utilize available cost estimates and correlations. Emphasis is on operational costs, which, particularly for reusable vehicles, such as an orbit-to-orbit shuttle or OOS, are dominated by EOS delivery cost; a value of $100/lb, a nominal NASA goal, is assumed, but variations are considered. Expendable, partly reusable, and fully reusable systems are considered; conventional staging, however, is not. Payload intrinsic costs are not considered, nor are any existing stages included in the study. Interplanetary missions, the customary justification for nuclear propulsion are not considered.

Results show that high-ΔV missions, e.g., missions to and from geostationary or lunar orbit (the two are about equivalent) from low earth orbit, are reduced in cost by at least partial expendability, e.g., the use of droppable propellant tanks. The nuclear system gains the most and the F_2/H_2 system the least from the use of drop tanks. With drop tanks, NERVA-powered systems are found to involve specific operational costs in the delivery of an arbitrary-sized bulk payload to lunar orbit (with empty-stage return) that are about twice those of chemical systems at 20,000-lb design capacity, about equal at 100,000 to 150,000 lb, and 5 to 12 percent lower at 500,000 lb. Fixed vehicles off-loaded to deliver smaller-than-capacity payloads are always more expensive than are "rubber" vehicles operating at design capacity. The costs per pound of payload of off-loaded nuclear systems rise more rapidly with decreasing payload than do costs of off-loaded chemical systems. These factors suggest that vehicle design capacity should be selected towards the lower end of the payload size/frequency distribution, and that clustered small vehicles be used to deliver large monolithic payloads.

As the ratio of material returned to material delivered increases, benefits from the use of nuclear rather than chemical propulsion increase. The magnitude of those benefits depends on the return ratio, on the vehicle sizes compared, and on the need for and mass of a radiation shield for the payload. The operational costs of advanced F_2/H_2 propulsion are always lower than those of advanced O_2/H_2 propulsion (by about 10 percent for payload delivery to lunar orbit and stage return). For delivery of cargo (unmanned) to lunar or geostationary orbit, expendable vehicles have unit delivery costs comparable to reusable vehicles; for direct delivery one way to higher ΔV's, as to the lunar surface, fully expendable systems are more economical. (Retrieval of high cost/weight components of "expendable" vehicles should be attractive but was not investigated.)

As noted, transportation costs to earth orbit dominate orbit-to-orbit shuttle (OOS) operational costs. Hence, if cargo volume in the EOS is insufficient to utilize the full weight-carrying capability of the EOS when carrying liquid hydrogen, the nuclear system will be at a competitive disadvantage. This point should be kept in mind in EOS design, or external-carry configurations should be considered. In addition, more emphasis should be given to incorporation of OOS tankage drop. High area ratio (~400 to 1) chemical (O_2/H_2 and/or F_2/H_2) rocket engines of moderate thrust (15,000 lb) and pressure ($\geq$ 1000 psi) offer significant reductions in operating costs over current O_2/H_2 engines and should be given consideration for development.

II. INTRODUCTION

A variety of missions and programs can be postulated to take place in the lunar or cislunar space regime, and, in principle at least, each could be carried out by a variety of expendable, partly expendable, or fully reusable vehicles, in stages or in combination. In this paper, in which chemical and solid-core nuclear (NERVA) propulsion systems are compared, only a few of the many possibilities are considered. Specifically, the missions to be considered are all in the high-velocity-increment class, e.g., to lunar or geostationary orbit from low earth orbit (and possibly return), or greater than 25-deg low-earth-orbit plane change. Chemical propellant combinations are restricted to O_2/H_2 and F_2/H_2. The configurations to be considered are restricted to single-stage or stage-and-a-half (tankage drop) classes, but both expendable and reusable modes are permitted. Within these restrictions, however, an attempt is made to treat each class of propulsion system in sufficiently general fashion to find the best (minimum-cost) application regime for each, following which comparisons are made for a specific mission.

The study is transportation-system oriented, i.e., delivery costs are compared and some consideration is given to total transportation program costs, but intrinsic payload costs are not considered.

As a ground rule for the study, the existence of an earth-to-orbit shuttle (EOS) is assumed with specific delivery costs to earth orbit of the order of $100/lb, but the impact of the characteristics of the EOS on the results is noted. Vehicles to be considered are assumed to be designed purely for in-space use, of types which could be available in the late 1970's. The approach utilizes internally consistent techniques to the extent possible, in principle improving the relative validity of the results. Critical parameters are subjected to perturbation analysis to determine their impact.

The results of the study are presented herein in several ways, and should be considered as a whole. In general, the material is presented sequentially in terms increasingly specific to a single mission. A very general performance discussion is provided first, indicating expected regions of superiority and overlap, following which more detailed cost and performance comparisons are made in terms of generalized payload-velocity increment categories, and finally a specific lunar mission is treated. Certain stage sizes are used throughout the work for consistency; the reasons for the selection are not given, however, until the specific mission is considered.

III. PERSPECTIVE

In comparing nuclear and chemical rocket propulsion on a generalized basis (with cost the principal criterion), the problem is basically to explore the interactions of each propulsion system's characteristics with various missions in order to find preferred regimes for each system. The problem is clearly multidimensional, as can be appreciated by considering the performance-determining factors in Table I along with the mission-dependent factors in Table II. (The values in Table I will be justified in later sections of the report.)

TABLE I. COMPARISON OF FACTORS DETERMINING PROPULSION PERFORMANCE

Characteristics	Nuclear (NERVA)*	Chemical (O_2/H_2 or F_2/H_2)
Engine thrust/weight	low (~ 3)	high (50-100)
Propellant bulk density, lb/ft^3	low (~ 4.4)	high (20-40)
Stage propellant mass fraction	low (≤ 0.78)	high (0.90-0.93)
Specific impulse, sec	high (~ 825-900)	low (445-480)
Engine cost/lb thrust, dollars	high (200)	low (20)
Startup and, if reused, cooldown losses	substantial (≤ 50 sec I_{sp} equivalent)	negligible
Shielding for manned missions	substantial (~ 10,000 lb)	none

*The values shown for nuclear systems are for the NERVA 75,000-lb thrust system only, and are not independent of thrust, as is essentially the case with chemical systems. Only the NERVA system was considered here.

TABLE II. MISSION-DEPENDENT FACTORS

- Payload size and divisibility
- Velocity increment(s) required
- Degree of vehicle reusability
- Degree of staging (tankage drop vs. no tankage drop)
- Operating mode (deliver or retrieve payload, etc.)
- Costs of payload delivery to low earth orbit
- Manned vs. unmanned operation

"Slices" in various directions through this multidimensional space will occupy the remainder of this study. Before going into detail, however, it is worthwhile to examine qualitatively certain aspects of these systems to indicate what might be expected in various cases. The intent is to provide insight as to why the various results come out as they do. In approaching the problem the basic rocket equation, $\Delta V = I_{sp}\, g_c\, \ln R$, should be kept in mind: The velocity added ΔV is a product of specific impulse I_{sp}, the units conversion constant g_c, and the logarithm of the initial-to-final-mass ratio R.

To start, note that the NERVA 75,000-lb thrust engine is currently expected to weigh 25,750 lb* (Ref. 1) and cannot readily be scaled downward, whereas chemical rocket engines of the same 75,000-lb thrust would weigh about 1000 to 1500 lb and could be either a single engine or a cluster of smaller engines without much penalty. The large nuclear engine weight means that large propellant weights will be required to achieve a reasonable

*With a goal of 23,000 lb.

stage propellant mass fraction and a reasonable value of R. The low bulk density of the liquid hydrogen fuel means, however, that relatively high weights are associated with each pound of added fuel, so that propellant mass fractions can never reach the values achievable with chemical systems. The much higher specific impulse of the nuclear system is, of course, compensating. In a crude sense, nuclear propulsion achieves performance by high specific impulse, and chemical systems by high mass ratios. Nuclear systems must be fairly large, while chemical systems can be any size desired.

Consider now some limiting conditions, and take first the case where payload is negligible relative to the inert weight, so that R is simply the reciprocal of the inert mass fraction. This case gives the maximum achievable ΔV per stage. If values are taken from Table I for equivalent-sized reusable vehicles,* which values ignore cooldown losses and any need for a radiation shield, one finds that, at the no-payload condition, ΔV for the F_2/H_2 system [480x32.2 $\ln(1/0.07)$ = 41,080 fps] is greater than ΔV for the nuclear system [825x32.2 $\ln(1/0.22)$ = 40,195 fps]. A finite high-ΔV region thus exists in which chemical systems may outperform nuclear systems. Payloads in this region are small, however, so that the point may be of doubtful economic interest; staged systems would probably be preferred.

As a second limiting situation, consider the case where inert weight is negligible relative to payload weight, i.e., where payload weights are large and ΔV's are small. In this case, the question is of payload (W_{PL}) ratios between nuclear and chemical systems at fixed propellant loads W_P and ΔV; R becomes $1 + W_P/W_{PL}$ and $\Delta V/I_{sp}\, g_c$ becomes small, so that

$$R = e^{(\Delta V/I_{sp}\, g_c)}$$

becomes

$$1 + \frac{W_P}{W_{PL}} \cong 1 + \Delta V/I_{sp}\, g_c$$

and

$$\frac{W_{PL}(\text{nuclear})}{W_{PL}(\text{chemical})} \cong \frac{I_{sp}(\text{nuclear})}{I_{sp}(\text{chemical})} \cong 2.$$

Thus, for large payloads at low ΔV's, and at equal propellant weights, nuclear systems will clearly move bigger payloads than will chemical systems.

*I_{sp} = 825 sec for a reusable nuclear system.

In terms of operational modes, one can consider the various systems to be expendable or reusable. For reusable systems, there may be various ratios of material delivered to material returned. If systems are used in an expendable mode, delivering material to, say, lunar orbit from low earth orbit, a ΔV of only about 14,000 fps is required, whereas a ΔV of 28,000 fps is required by systems in the reusable mode. For this mission, the expendable nuclear system approaches (but does not reach) the second limiting condition above, whereas the reusable system approaches the first limiting condition. In short, one can expect that the nuclear system will perform better relative to chemical, at least on a payload per pound of propellant basis, in an expendable than in a reusable mode (payload delivery with empty stage return). In addition, if the reusable system is to be designed to retrieve material from, rather than deliver material to, some higher-energy orbit, high mass ratios will be difficult to achieve for either system on either the outbound or the return leg, and nuclear systems (which, as noted earlier, achieve velocity gain by I_{sp} rather than by mass ratio) may outperform the chemical, if total ΔV is not too close to the limiting stage velocity.

A few points with regard to cost are worth noting here. At the EOS cost goal of \$100/lb to low earth orbit, propellant and vehicle structure change in value relative to earth-based rocket systems. Structure in orbit takes on a value only 2 to 4 times that of consumable propellant, as opposed to perhaps 1000 times the propellant value at the earth's surface. Structure in orbit can thus be more casually "consumed," i.e., expended, if, as might be expected from the above discussion, payload gains make it worthwhile. It thus follows that expendable vehicles, or largely expendable vehicles whose very high unit-cost components are recoverable, may be more attractive than reusable vehicles for routine cargo carrying. The cost of the material thrown away must be established, of course. Finally, at this same \$100/lb, delivery costs totally mask initial costs of propellants. As F_2/H_2 is higher in performance than O_2/H_2, specific operational costs of F_2/H_2 will, under these assumptions, always be lower than those of O_2/H_2. Liquid H_2 for the nuclear stage will cost this same \$100/lb in orbit only if the EOS system is not volume limited, however.

All the above factors are examined further, along with a number of options in modes of use, in the material that follows.

IV. PROCEDURE

In essence, the procedure has been to build up a single design, cost, and performance program using correlative techniques, and then to compare all systems using common procedures under a variety of conditions. For each propellant system, the principal input independent variables in the design program are propellant mass and the number of drop tanks.* The design program synthesizes the vehicle from these inputs, computing the total inert weight from correlations. Payload is then computed to specified ΔV values by a double iterative process made necessary by the need to include gravity losses, which depend on thrust to total weight (including payload) and specific impulse, and, in the nuclear case, startup and cooldown losses which depend on burn time. A triple iteration is involved when empty tanks are to be dropped at some intermediate ΔV point. These drop tanks are sized for the pickup mission, i.e., the mission requiring the least outbound propellant consumption; other payload modes are calculated around this tankage split. Mission costs and total program costs are then built up, it being remembered that outbound payload, propellant, and each new OOS must first be delivered to orbit by the EOS. The items (space vehicle, engines, and tankage) for in-space use are brought to orbit only once during their useful life; this assumption favors the reusable system over the expendable system. (The used nuclear engine probably could not be brought back to earth in any event.) The intrinsic cost of the payload itself is not included, however.

Details of the design and costing procedure are appended. Correlations from previous IDA studies (Refs. 2, 3), modified as necessary, were used to estimate tankage weight, residuals, thrust structure weight, subsystem weights, and so on. To these were added system-specific information for the nuclear and chemically powered systems. Insulation, meteoroid protection, and boil-off correlations were developed from previous detailed studies (Refs. 4-8). A startup and cooldown-loss correlation for the nuclear system was developed from available information (Ref. 8). Gravity-loss correlations were developed for single-burn injection of vehicles from low earth orbit into transfer orbits to geostationary orbit or lunar orbit (see Appendix); multiperigee burn was also considered briefly (Ref. 9).

Once these procedures were mechanized, several suboptimizations were carried out to select reasonable values of certain important parameters. For chemical systems, partly in order to maintain compatibility with existing studies of advanced engines of the RL-10 class but also to minimize R&D costs and to move "out on the learning curve," it was decided to standardize computations on engines of 15,000-lb thrust, clustered as necessary. Predicted specific-impulse data for this class of engines from Pratt & Whitney (Ref. 10) were examined and plotted (Fig. 1). Examination of weight penalties and specific impulse gains with increasing area ratio showed performance still increasing at the maximum area ratio (400) for which predictions were available; a value of 400 was thus selected and a weight of 350 lb per engine of 15,000-lb thrust was assigned. With these engines in the lunar-transfer round-trip missions, a thrust-to-weight-of-propellant T/W_P of 0.3 is shown later to be about optimum with regard to performance.* An engine operating life of 10 hr was assumed, in accordance with current NASA Phase B Shuttle Main Engine goals.

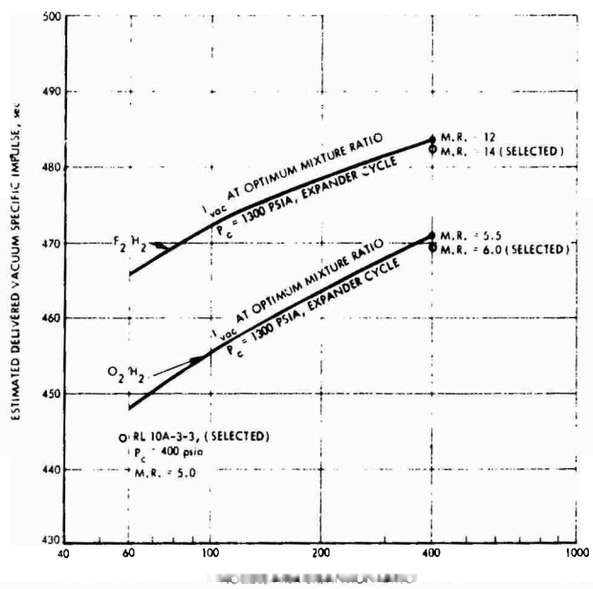

FIGURE 1. ESTIMATED PERFORMANCE OF 15 KLB THRUST CHEMICAL PROPULSION SYSTEMS VERSUS NOZZLE AREA RATIO (REF. 10)

In the nuclear system a specific-impulse value of 825 sec at full thrust was selected for the reusable missions, with 10-hr life. The final throw-away requirement had little impact and was ignored. For expendable nuclear systems, a specific impulse of 900 sec was assumed. As already noted, the engine goals of 75,000-lb thrust and 23,000-lb weight were assumed. The weight figure assigned is the goal, versus a 25,750-lb present value (Ref. 1).

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* Tank staging occurs only once, even with multiple tanks.

* For plane changes in low earth orbit, a thrust/total weight of ~ 0.5 is preferred.

Vehicle life, as opposed to engine life, was given some consideration. If missions are infrequent, if space vehicles are not brought back to earth surface between trips, and if no "space garage" is available, meteoroid protection could impose a significant penalty in terms of vehicle weight and performance, the magnitude depending on the waiting time between trips. A set of assumptions was finally made in which reusable vehicles were given meteoroid protection (0.995 chance of no penetration specified) for a total flight duration equal to the number of 24-day round-trip lunar missions compatible with 10 hr of engine life (and expendable vehicles for a single 3-day lunar transfer). For the chemical systems with a T/W_p near 0.3, the number of missions is 20 to 25, depending on I_{sp}, giving a requirement for 480 to 600 days meteoroid penetration. Nuclear vehicles with fewer missions for the 10-hr engine life have the structure life extended to an integral multiple of the engine life, giving a number of missions nearer to 20 (actual values vary from 15 to 28). This technique assumes either continuous (no waiting time) operation or, more realistically, the availability of a "space garage" for periods between missions.

Boil-off due to heat leak was assumed to be vented before the return trip, which, in the case of lunar missions, was assumed to follow 18 days in lunar orbit. A subtle point in tank sizing was involved in designing the chemical vehicles for use in either geostationary or lunar orbit. In essence, the oxidizer tanks were oversized so as to handle, in the geostationary mission, an equivalent oxidizer mass corresponding to the extra weight of hydrogen needed for boil-off in the lunar missions. In lunar missions the oxidizer tank had extra ullage; in geostationary missions the hydrogen tank had extra ullage.

To show propellant mass fraction results generated by the design portion of the program, Fig. 2 is provided. This shows stage mass fraction as a function of propellant weight for various systems. In the nuclear case, the program has been built around earlier correlations, with weight data specific to the nuclear system included from detailed contractor estimates. The baseline vehicle selected herein for the nuclear system is in fact nearly the same as that of these contractors. The predicted decrease in propellant mass fraction at very large single-tank sizes would be avoided by clustering tanks. Selected values of the parameters used in this study are provided in Table III. A diagram showing the lunar mission is shown as Fig. 3. The geostationary mission involves simple Hohmann transfer and circularization, with optimum plane change at perigee and apogee.

As already suggested, the EOS characteristics are important to the comparison, inasmuch as EOS per-flight costs are presumably constant, so that delivery to orbit of low-density material such as liquid hydrogen may cost more per unit mass than delivery of more dense materials that utilize the full weight-lifting characteristics of the booster. Whether this will be the case depends on payload density and the degree to which it can be broken into sections for delivery. If payload is treated as being subdividable at will and reasonably dense, then a few rough calculations will show that the

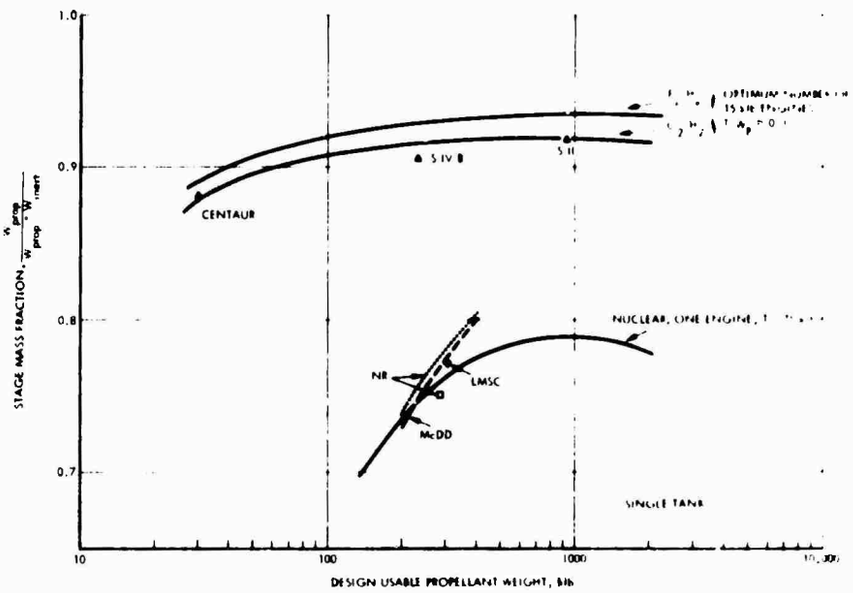

FIGURE 2. COMPARISON OF STAGE WEIGHT CORRELATIONS WITH EXISTING STAGES AND OTHER DESIGNS. SINGLE TANK

TABLE III. SELECTED NOMINAL VALUES OF PERFORMANCE-DETERMINING PARAMETERS[a, b]

Flight

Initial EOS parking orbit	263 nmi x 31.5 deg
Characteristic velocity requirements (each way, from parking orbit)	
Geostationary orbit (including optimum split of plane change)	7,888 fps (perigee) 6,003 fps (apogee)
Translunar injection	10,466 fps
Lunar orbit insertion	3,000 fps
Descent from lunar orbit to surface	6,000 fps
Direct lunar descent	9,000 fps
Translunar (and transearth) coast time	3 days
Lunar orbit stay time	18 days
Velocity pad	0.75% each ΔV
Thrust parallel with velocity vector for g-loss estimates	

EOS Baseline Characteristics

Cargo bay	15x60 ft
Cargo volume	10,000 ft^3
East launch payload (if not volume-limited)	50,000 lb
EOS delivery cost	$100/lb

Engine Data

	M.R.	ϵ	p_c	I_{sp}, sec	Weight, lb	Thrust	R&D, $M	First Unit Cost, $M
Current O_2 H_2	5	57	100	445	300	15,000	50	0.28
Adv. O_2 H_2	6	100	1300	469.5	350	15,000	215	0.28
Adv. F_2 H_2	14	100	1300	482.5	350	15,000	270	0.28
Nuclear (re-usable)[c]	--	100	450	825	23,000	75,000	750	15
Nuclear (expendable)[c]	--	100	450	900	23,000	75,000	750	15

Cost Data, $M

	Typical Stage, First Unit	R&D (Including Engines)
Nuclear (290K)	38.0	1650
Adv. O_2 H_2 (100K)	7.1	485
Adv. O_2 H_2 (290K)	12.8	670
Adv. F_2 H_2 (100K)	6.5	520
Adv. F_2 H_2 (290K)	11.6	680

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[a] See also Appendix.

[b] IDA estimates except as shown.

[c] Ref. 1.

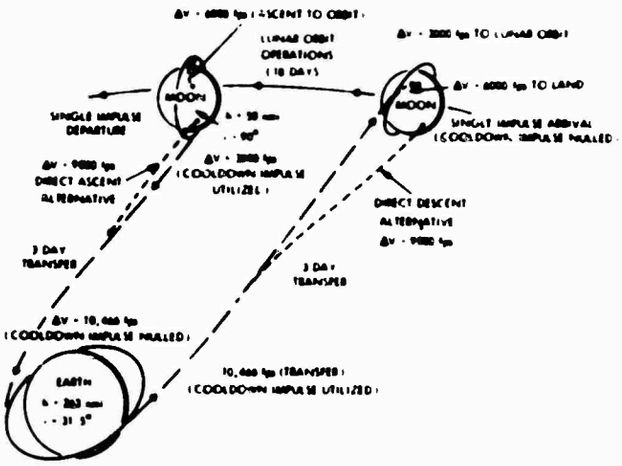


FIGURE 3. SIMPLIFIED LUNAR OPERATION SCHEMATIC

assumed EOS may not be volume-limited, and a $100/lb delivery cost may reasonably be ascribed to both nuclear and chemical systems. Consider that a nuclear stage utilizing 300,000 lb of propellant can typically put about 130,000 lb of payload around the moon, part of which payload must be chemical propellant for the lunar lander, and part of which will be supplies for a moon base, etc. If the average packaging density of payload is 10 lb/ft^3 and of LH_2 4 lb/ft^3, then a volume of 88,000 ft^3 is required for the total of 450,000 lb (allowing 20,000 lb of tankage weight to contain LH_2) delivered. At 50,000 lb per trip,* 450,000 lb requires 9 trips of the EOS, which permits 90,000 ft^3 of material to be carried. If only LH_2 is to be carried, or if the EOS has a higher payload mass capability/ft^3 of cargo bay, then LH_2 must be carried externally to avoid unit delivery cost penalties. These points clearly merit more study for they are critical to the operating cost of the nuclear system. However, for all purposes here except the sensitivity studies, all mass shall be assumed deliverable to parking orbit at equal cost, independent of density.

*East launch.

V. OPTIMUM OPERATING REGIMES

As was mentioned under Perspective, the different characteristics of nuclear and chemical propulsion systems imply different optimum operating regimes. In order to explore the matter in some depth, calculations were carried out to determine average specific operational costs (recurring costs per pound of payload) for the different systems under different circumstances. The operational costs were taken reasonably far down the learning curve (at a common total program delivery weight of 10,000,000 lb to different ΔV values above escape) to minimize cost-quantizing effects arising from the need to purchase an integral number of vehicles. Implicit in a comparison of this type is a scenario in which some agency wishes to purchase the necessary fleet of an already developed class of vehicles, designating them to carry out some defined program, with total costs to be a minimum. R&D costs are thus not included.

The comparison is centered around hypothetical missions similar to the lunar exploration mission in terms of boil-off and gravity losses, but velocity excess is treated parametrically. Because costs are substantially reduced thereby, as shall be shown later, two empty propellant tanks were dropped either after the second burn (reusable vehicles) or after the first burn (expendable vehicles). Multi-perigee burn was not considered in the set of computations reported in this section. One assumed mission was that of delivery of material only, with no payload to be returned; in the other assumed mission, a 10,000-lb payload, presumably a manned capsule, was to be brought back, which in the case of nuclear stages required an additional 10,000-lb radiation shield to be carried round trip. An EOS delivery cost of $50/lb, as well as the baseline cost of $100/lb, was assumed.

The calculations from which the comparisons were developed are illustrated for one example in Fig. 4.* In Fig. 4 are plotted specific operating costs for different vehicles versus vehicle sizes for a 3000-fps velocity excess over escape (approximately the value used in lunar orbit insertion), showing the operating costs as a function of payload delivered and regimes in payload size where different vehicles are preferable due to lowest cost.**

*Note that the curves shown for chemical systems actually represent loci of minima, in which one additional engine is added for each 50,000 lb of propellant. The curvature between numbers of engines is much smaller than with the nuclear engine, and is ignored.

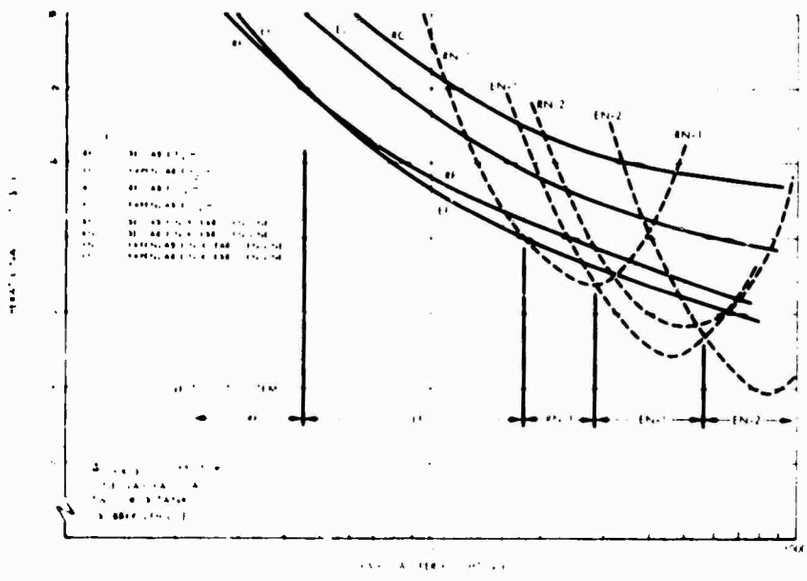
**Figure 4 is perhaps more readily comprehensible if viewed in connection with Fig. 10 (which is for the same velocity excess) and its discussion in the following section. Figure 10 shows payload delivery as a function of propellant loading for expendable and stage return modes shown in Fig. 4. Curves similar to those in Fig. 10 were not, however, necessary in generating Figs. 5-9; curves similar to those in Fig. 4 were.

Similar curves were developed for other velocity-increment values from 0 to 9000 fps above escape, and minimum-cost regimes determined at each ΔV, in the same fashion as noted in the arrows paralleling the abscissa in Fig. 4. Results are shown in Figs. 5-7. Note that, as expected, the nuclear system is preferred only for large payloads because of its large fixed weight. Note also, that at high-ΔV values, expendable systems are preferred for the delivery mode. (Presumably guidance and other high-unit-cost items would later be recovered, if possible.) At the ΔV corresponding to injection into lunar orbit, with $100/lb EOS cost, the reusable nuclear systems show lower operational costs than O_2/H_2 in the region above 110,000 lb flight (Fig. 5). At $50/lb (Fig. 6), the map is similar, but the reusable systems are competitive to higher ΔV values.* With fluorine (Fig. 7), at $100/lb the reusable nuclear system is cheaper in a narrower region: at lunar-orbit ΔV requirement, the nuclear systems become cheaper only above 175,000 lb per trip. If velocity excess values of 4000 fps are required (plane changes in lunar orbit of the order of 1000 fps), the reusable nuclear region almost disappears against O_2/H_2 at $100/lb, and does disappear against F_2/H_2 at $100/lb. The $50/lb F_2/H_2 case is not shown as it was not fully investigated. It was evident, however, that the reusable fluorine area grew at the expense of the reusable nuclear system.

Marked on the abscissa of Fig. 5 are, respectively, the ΔV's for escape, 30-deg low-earth-orbit (LEO) plane change, geostationary orbit, lunar orbit, 10-deg LEO plane change, direct descent to the lunar surface, and 50-deg LEO plane change. Comments on "toss-catch" and direct descent are postponed until the next section.

As can be seen by examination of Fig. 4, the cost differences remain small for substantial increases in payload near the boundaries. A measure of the large effect of a small cost uncertainty on the location of the boundaries is also shown on Fig. 5, indicating the more limited areas where each system is 5 percent less expensive than its nearest competitor. Included is the boundary at which nuclear systems became 10 percent less expensive than O_2/H_2 systems, approximately following a 200,000-lb payload level.

For manned operations, expendable but staged operation is obvious possible if two or more stages are used. As noted in the Introduction, such configurations have not been considered; manned operations are here assumed to be carried out with reusable (or partly expendable) systems. To compare manned systems, the same ground rules as in Figs. 5-7 were used, but cost figures were computed for the case in which a 10,000-lb payload was brought back each trip from the assigned ΔV; that is,



[illegible] AS A FUNCTION OF PAYLOAD DELIVERED [illegible] $100/LB EOS DELIVERY COST.

* The three nuclear engine option was not considered in developing Figs. 4-6.

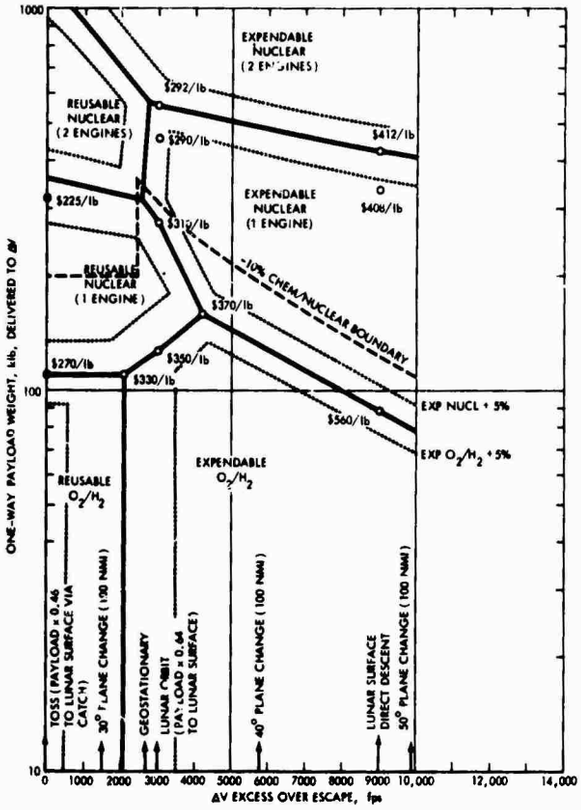


FIGURE 5. OPTIMUM OPERATING REGIMES, O_2/H_2 VS. NUCLEAR. EOS COSTS = $100/LB

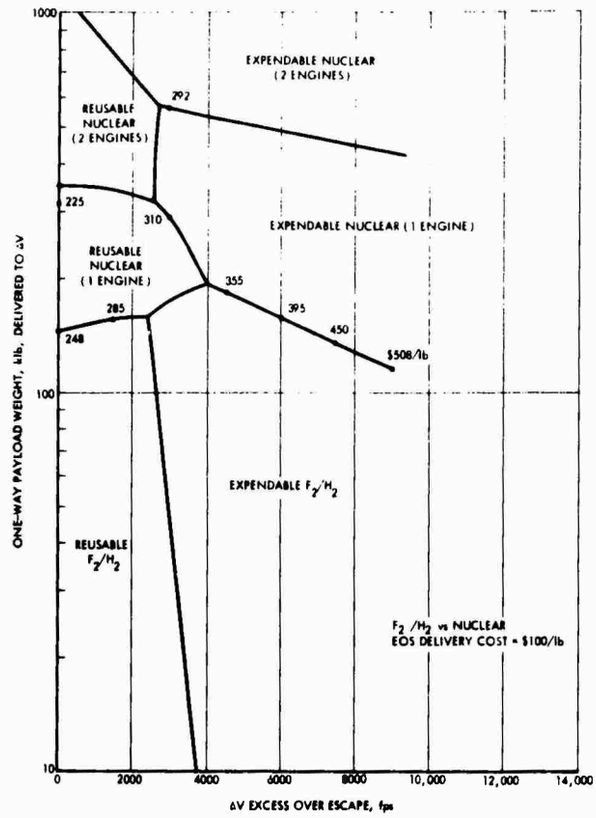


FIGURE 7. OPTIMUM OPERATING REGIMES, F_2/H_2 VS. NUCLEAR. EOS COSTS = $100/LB

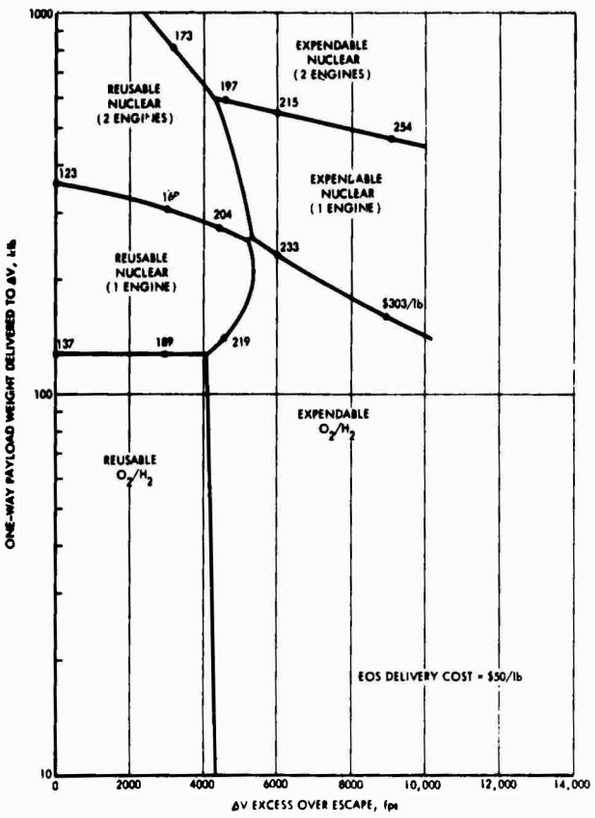


FIGURE 6. OPTIMUM OPERATING REGIMES, O_2/H_2 VS. NUCLEAR. EOS COSTS = $50/LB

outbound payload capabilities of rubber vehicles (at various propellant loadings) were computed on the assumption that the vehicle itself, plus 10,000 lb, was brought back each trip. The same total outbound delivery program was assigned, implying that the smaller vehicles made more trips and brought back greater quantities of return payload, for which they were given no extra credit. Results are shown for O_2/H_2 versus nuclear propulsion in Figs. 8 and 9. At $100/lb EOS cost, the nuclear system is cheaper for payloads above about 120,000 lb. At $50/lb, the nuclear system was attractive only for payloads above about 150,000 lb. F_2/H_2 was again not investigated in sufficient detail to map. At $100/lb, it appeared that F_2/H_2 was less expensive to lunar-orbit ΔV up to about 300,000 lb of payload per trip, against two-engine nuclear vehicles. At $50/lb, F_2/H_2 was cheaper at all payloads and ΔV's, except at velocity values below about 1000 fps above escape. Note, however, that in these comparisons the nuclear system was substantially penalized relative to the chemical systems in that a 10,000-lb radiation shield was carried round trip, the shield having been assumed necessary for personnel protection. Had this not been the case, or had larger return payloads been assumed, the nuclear systems would have increased in attractiveness. The latter point will be explored further in the next section along with criteria for specific vehicle selection.

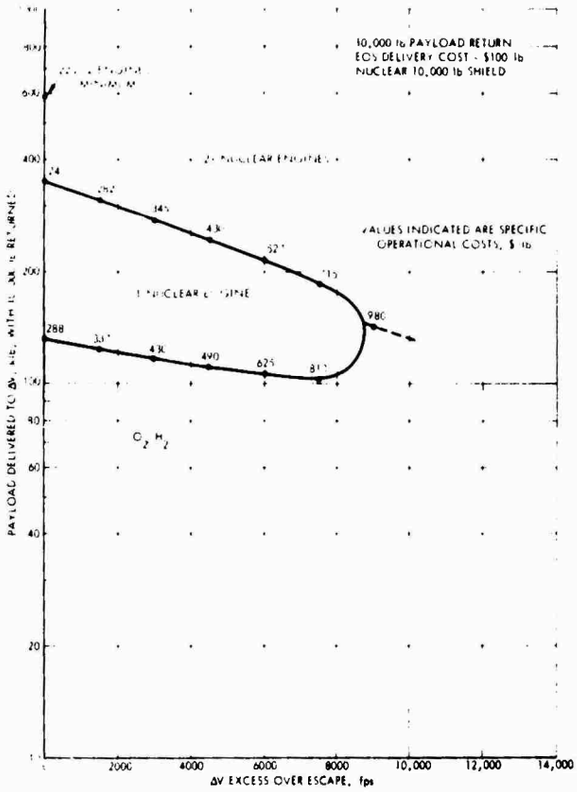


FIGURE 8. OPTIMUM OPERATING REGIMES, O_2/H_2 VS. NUCLEAR, 10,000 LB RETURNED EACH FLIGHT. EOS DELIVERY COSTS = $100/LB.

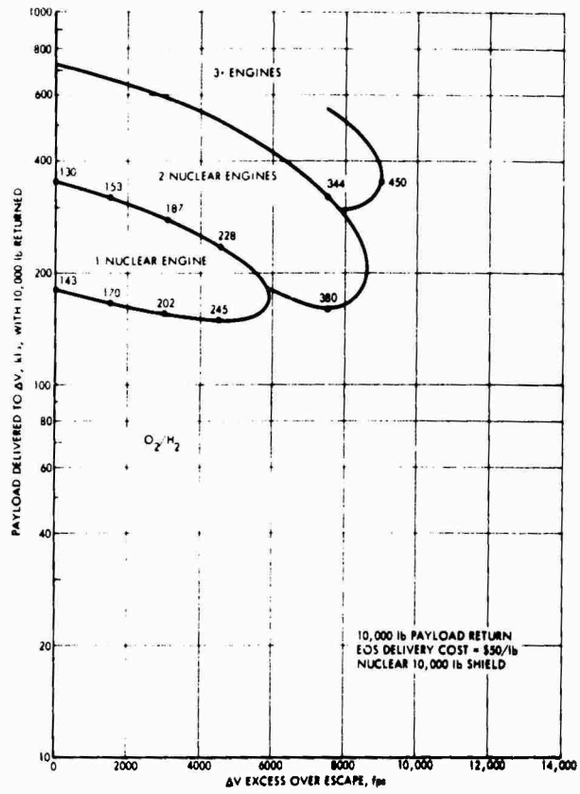


FIGURE 9. OPTIMUM OPERATING REGIMES, O_2/H_2 VS. NUCLEAR, 10,000 LB RETURNED EACH FLIGHT. EOS COSTS = $50 LB.

VI. LUNAR AND GEOSTATIONARY MISSION STUDIES

A. ALTERNATIVES AND COMPARISONS

In previous sections, payloads and velocity requirements have been treated parametrically in order to locate minimum operational cost regimes for the different systems. In this section, the lunar mission will be emphasized. It will be shown that lunar-orbit payload capabilities of specific vehicles tend to be close to those required for geostationary-orbit payloads. Statements previously made about the advantages of the nuclear system in certain use modes and the advantages of partial expendability (propellant drop tanks) in specific applications will be justified, and specific payload ratios and vehicle sizes will be compared.

Figure 10 presents payload performance results as a function of propellant design weight for "rubber" vehicles. The plot shows vehicles delivering material to lunar orbit in expendable, stage-return-only (payload-delivery), and payload-round-trip modes. The expendable and stage-return-only vehicles are assumed to be unmanned and without need of a radiation shield. Furthermore, for expendable vehicles meteoroid protection is necessary for only three days, and the nuclear system is assumed to deliver a 900-sec specific impulse at full thrust due to its short permitted life. The plot shows the large gain in performance of the expendable nuclear system over the expendable chemical systems (arrow A) and of both over the reusable systems, the substantial advantage of nuclear over chemical propulsion in the round-trip mode with large payloads (arrow B), and the lesser advantage of nuclear over chemical propulsion, even with large payloads, in the stage-return mode (arrow C).

Further perspective is given to the various systems in the lunar exploration mission by Fig. 11. In this figure, operational costs of three specific vehicles are computed for missions in which the ratio of payload to and from lunar orbit is varied. The abscissa (return payload)/(return + outbound payload) is plotted from 0 to 1. In fact, however, if a chemical shuttle from lunar orbit to the lunar surface (lunar lander) is used, the ratio cannot exceed about 0.55. At this ratio the outbound payload is composed only of propellant for the lunar lander and the return payload must be picked up from the lunar surface. The ratio can, of course, rise to unity if

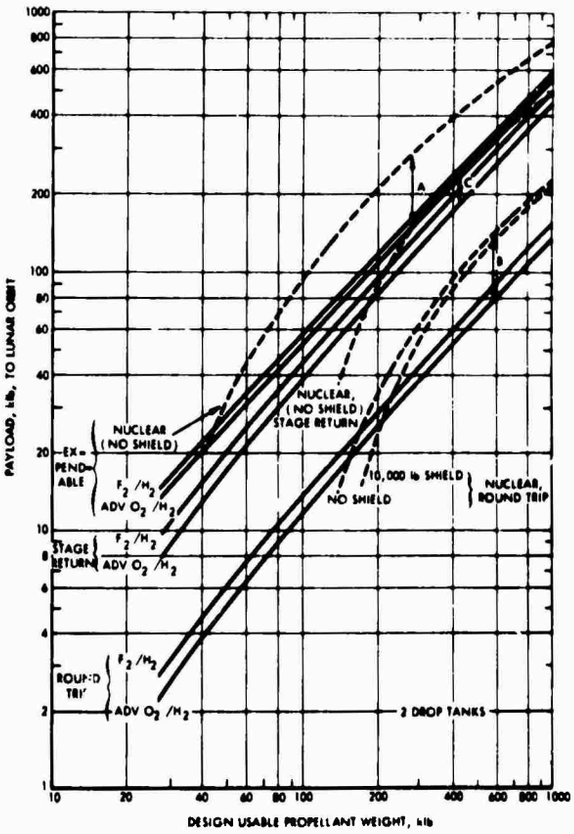


FIGURE 10. ONE-WAY AND ROUND-TRIP PAYLOAD TO LUNAR ORBIT FOR VEHICLES WITH DIFFERENT PROPELLANT CAPACITY. TWO DROP TANKS.

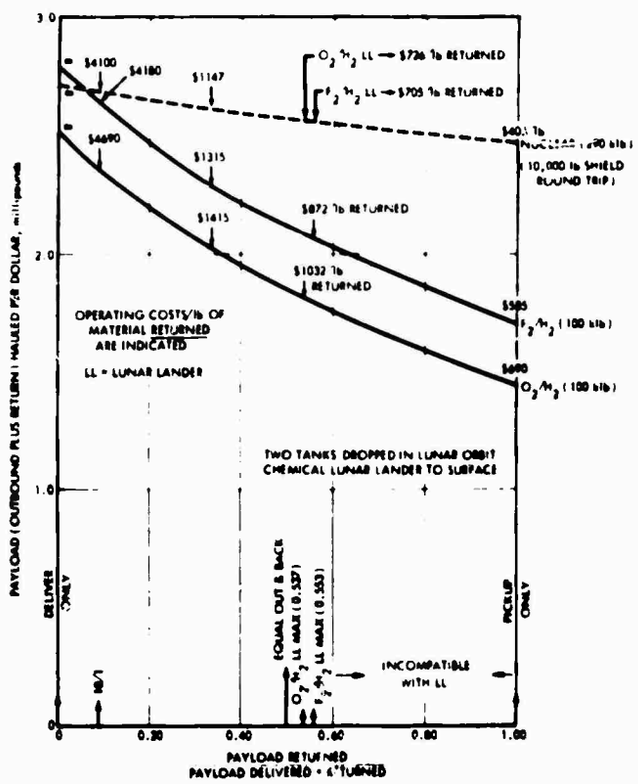


FIGURE 11. PAYLOAD RETURN RATIO EFFECTS IN LUNAR ORBIT MISSION. TWO DROP TANKS. EOS COSTS = $100/LB.

the mission is simply to retrieve some item already in lunar orbit. Note that, as the percentage of material returned increases, the larger nuclear system increasingly outperforms the smaller chemical system.* Note also the cost of picking up material from the moon by this technique, high in comparison with the value of most materials,** but several orders of magnitude below APOLLO costs. (Moon-mining enthusiasts, of course, have suggested far cheaper ways, e.g., the use of electromagnetic accelerators on the moon with atmospheric braking at the earth.)

In actual lunar exploration, at least in the early stages, the amount of material delivered to orbit will undoubtedly substantially exceed that returned; in Ref. 6 it is suggested that a typical operational ratio will be 10/1 in terms of material carried to lunar orbit to that returned (1/11 of the sum returned*). The 10/1 point is noted in Fig. 11 for this reason; it is clear from Fig. 11 that this ratio puts chemical and nuclear vehicles in a competitive region. This 10/1 ratio was used to make a series of further comparisons in the lunar mission. Before proceeding with these, however, it is worth noting Fig. 12, in which the ratios of 10/1 geostationary to 10/1 lunar-orbit payload capabilities of the various vehicles are plotted. Note that, except at extreme propellant loadings for the single-engine nuclear vehicles, the geostationary payload is very similar to the lunar-orbit payload. This is as might be expected, since total characteristic velocity requirements are nearly identical, as shown in Table 3. Thus further comparisons of lunar-orbit capabilities can with some care be interpreted as applying also to geostationary missions. The variation with propellant loading in nuclear vehicles is due to differences in gravity losses between geostationary and lunar transfers as thrust/weight changes (Appendix, Fig. A-1). In the chemical systems, the thrust/propellant weight is kept constant at 0.3 so that gravitational losses are independent of size.

Plots of the payload delivery capability to lunar orbit of chemical and nuclear vehicles with two drop tanks in the 10/1 mission are shown in Fig. 13.

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*Larger propellant loadings lead to lower operational costs, as can be seen from Fig. 4. The advantage of the nuclear system relative to the chemical system would be less if comparison had been made at the same propellant loading. Use of Fig. 10 and Fig. 4 shows that at 290 klb of propellant the ordinate at an abscissa of zero is about 2.8 for O_2/H_2 and 3.1 for F_2/H_2.

**Compare gold, for example, at $480/lb.

*Or 1/10 of the outbound.

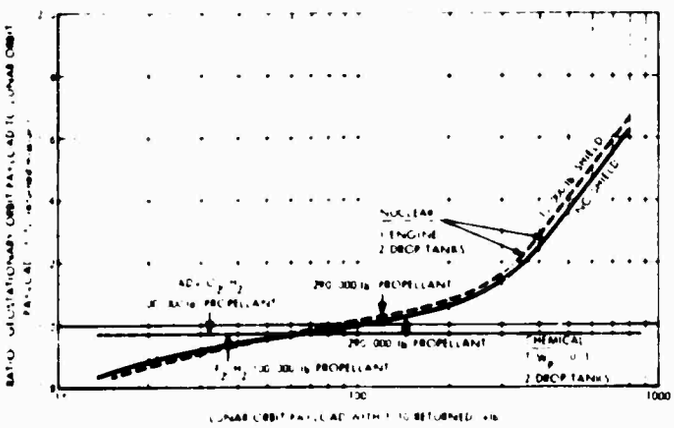

FIGURE 12 VARIATION OF GEOSTATIONARY ORBIT PAYLOAD TO LUNAR ORBIT PAYLOAD WITH VEHICLE SIZE

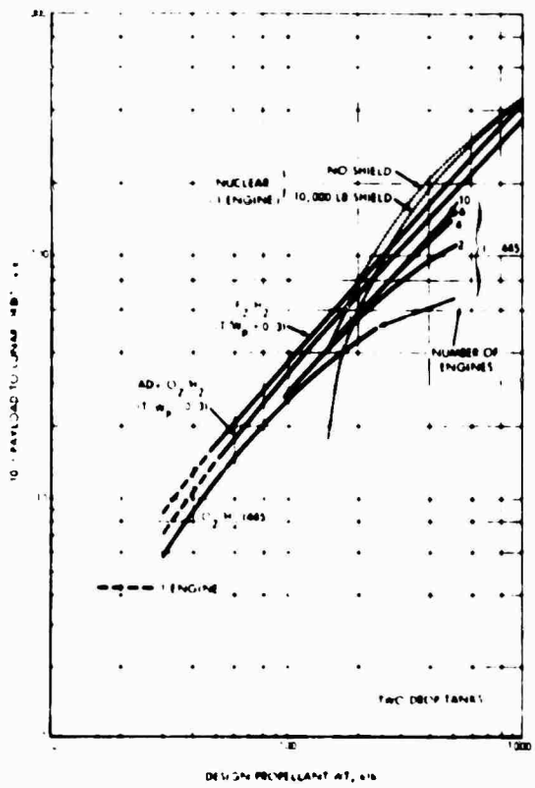

FIGURE 13 LUNAR ORBIT 10:1 PAYLOAD CAPABILITIES OF ALTERNATIVE VEHICLES AS A FUNCTION OF DESIGN PROPELLANT WEIGHT TWO DROP TANKS

Data plotted in Fig. 13 for the current O_2/H_2 propellant system indicate that a th st to propellant weight of about 0.3 is near optimum on a pound-payload/pound-propellant basis for chemical vehicles in this mission. The nuclear vehicle in direct ascent peaks out on the same basis at about 0.2. It can also be seen that, at 100,000-lb propellant, the use of high-expansion-ratio advanced O_2/H_2 engines (469.5 sec) yields about a 20 percent gain in payload over current-technology O_2/H_2 (445 sec).

Figure 14 shows specific operational costs for vehicles with no drop tanks and with two drop tanks in the lunar-orbit mission. Nuclear vehicles with both one and two engines are included in Fig. 14b, as are approximate savings with multiperigee burn. Multiperigee burn extends the optimum propellant loading to very high values but does slow down the mission in an operational sense. The ΔV and cooldown losses for this mode were estimated from Ref. 10. Multiperigee burn was not further considered herein. The values shown in Fig. 14 are all for "rubber" vehicles.

Figure 15 shows payload and cost advantages through use of separable propellant tanks. The gains are clearly greatest with the lowest density propellant system (nuclear), and least with the densest (F_2/H_2). The gains also increase on a fractional basis as the stages without drop tanks become marginally able to perform the mission. Figure 15 can, if desired, be used in conjunction with Fig. 13 to calculate payload performance of vehicles without drop tanks. Note that the gains in payload are of the order of 50 percent for the nuclear systems versus only 15 to 30 percent for the chemical systems. The cost gains are not directly equivalent, however, as the cost of the tanks discarded also varies with the specific volume of the propellant system. Tankage production costs here were estimated at \$150-\$300 per pound, depending on size

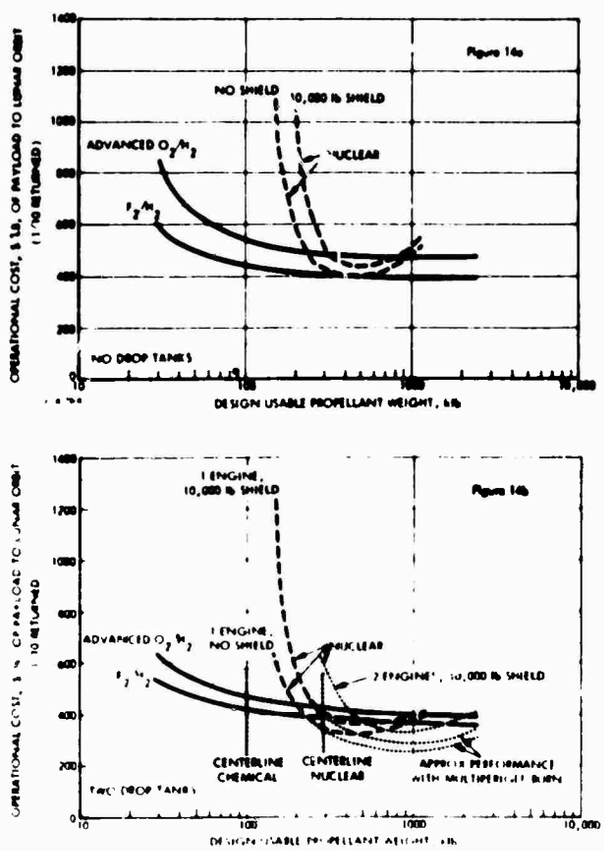

FIGURE 14 SPECIFIC OPERATIONS COSTS FOR DELIVERY OF PAYLOAD (10/1) INTO LUNAR ORBIT EOS DELIVERY COSTS \$100/LB.

and number produced, whereas some approximate arithmetic shows that the nuclear system (290,000 lb of propellant at $100/lb EOS costs) would break even on a cost basis (with and without drop tanks) at about $800/lb of tank. It should be noted from Fig. 14a that, if tankage separation is not permitted for operational reasons, the nuclear system will be less competitive with the chemical systems.

Another form of comparison relates to the performance of specific vehicles in delivering less than maximum payloads, a situation which would almost certainly be encountered in operational use. This case is considered in Fig. 16. Note that the nuclear vehicles have a steeper increase in cost with reduced payload than do the chemical vehicles of the same design propellant weight. In addition, it is shown in Fig. 16 that large vehicles off-loaded for small payloads are much more expensive to operate than small vehicles operated near their design point. Modularization (a number of small vehicles, in clusters, if necessary, for large unit payloads) should be attractive on this basis.

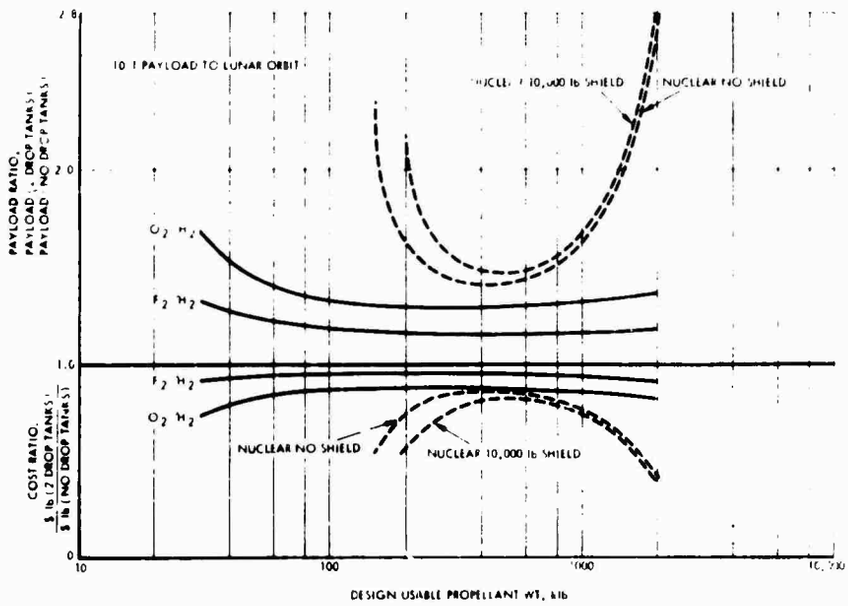

FIGURE 15. PAYLOAD AND COST RATIOS OF VEHICLES WITH TWO DROP TANKS TO VEHICLES WITH NO DROP TANKS. 10/1 PAYLOAD TO LUNAR ORBIT.

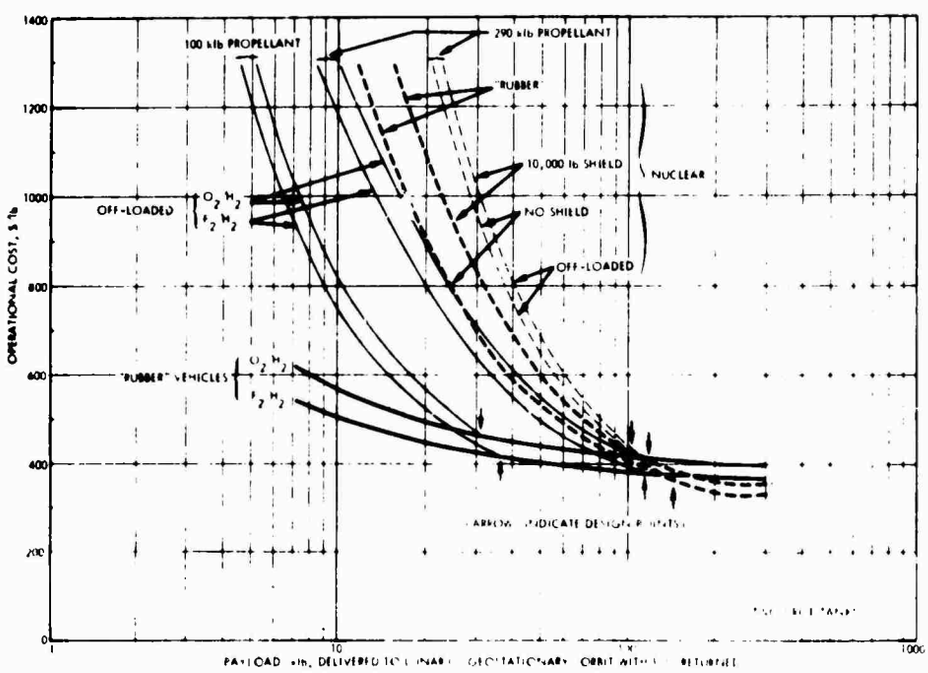

FIGURE 16. COMPARISON OF OFF-LOADED VEHICLES TO "RUBBER" VEHICLES IN TERMS OF OPERATING COSTS. 10:1 PAYLOAD DELIVERED TO LUNAR (~GEOSTATIONARY) ORBIT. VEHICLES WITH 2 DROP TANKS

Lunar exploration is, of course, not necessarily carried out by the use of lunar-orbit rendezvous. Lunar-orbit rendezvous, coupled with a lunar lander eliminates or makes expensive the possibility of arbitrary mission times, so that from the standpoint of a pure operational convenience, direct descent would appear to be preferred. The capabilities of advanced vehicles in direct descent and lunar-orbit rendezvous have been computed and are shown in Fig. 17. The nuclear system was granted the capability of direct descent here for academic comparison, although no present configuration is being considered that would be suitable for this mode. Lunar-orbit rendezvous is obviously attractive in terms of payload capability (or cost/lb) to the lunar surface if round trips are required. Only F_2/H_2 is shown in direct-descent, no-parts-drop mode, as F_2/H_2 is the only combination capable of starting from low earth orbit, carrying out a direct descent, and returning to low earth orbit without dropping any parts. This might be an interesting emergency vehicle, but would be expensive to use per pound delivered or returned.

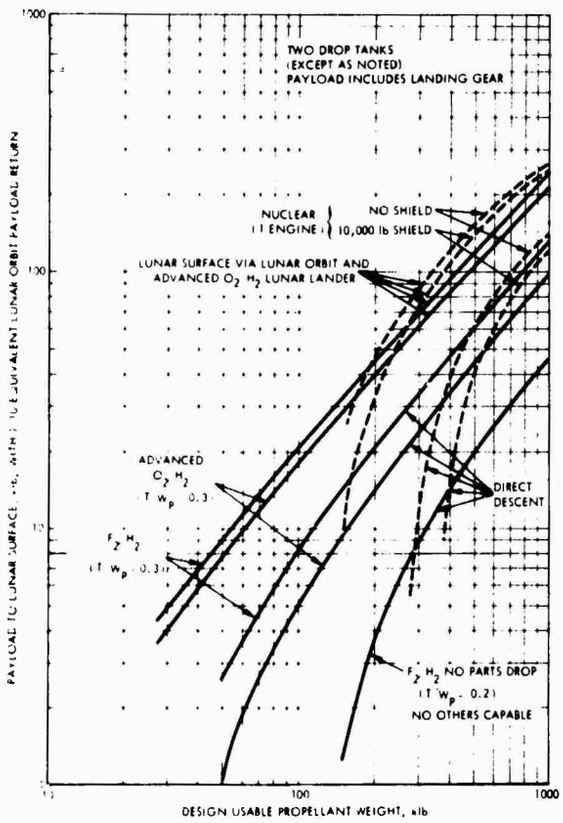

FIGURE 17. LUNAR-SURFACE PAYLOAD CAPABILITIES OF VEHICLES WITH RETURN OF 1/10 OF THE EQUIVALENT LUNAR-ORBIT PAYLOAD. TWO DROP TANKS. PAYLOAD INCLUDES LANDING GEAR.

Data shown earlier on Fig. 5 are also pertinent here. In Fig. 5 the "toss-catch" mode, in which the lunar lander and the OOS rendezvous at the earth-moon null point, as well as conventional lunar-orbit technique and direct descent, is indicated. Dividing the indicated delivery-cost-per-pound figures at these ΔV's by the indicated ratio (Fig. 5) gives the cost per pound to the lunar surface via toss-catch and lunar-orbit rendezvous to be compared with the value for the direct-descent mode. The respective specific delivery costs of about \$580/lb, \$550/lb, and \$560/lb for the three different modes, for lunar-surface payloads of 80,000-90,000 lb, do not indicate a marked superiority of any of these three ways of delivering one-way payload to the lunar surface.

B. TRADEOFFS AND SENSITIVITIES IN THE LUNAR EXPLORATION MISSION

Previous calculations have, for the most part, considered "rubber" vehicles and made generalized comparisons. Eventually, however, some specific vehicles will need to be selected and compared. This focusing is particularly necessary if sensitivities are to be explored. The basis for this exploration is provided by the following material. The rationale for the vehicle choices will be given, recognizing that subjective factors enter. The R&D costs for the selected vehicles will be compared and differences in operational costs used to ascertain a crossover payload at which program costs would be equal. The crossover payload is a mathematical artifice in a sense, for the same value may result from the ratio of a large difference in R&D costs to a large difference in operational costs, or a small difference in R&D costs to a small difference in operational costs; it is nevertheless a sensitive measure and well suited to demonstrating analysis of perturbations around a baseline.

The technique employed to select and compare centerline vehicles is illustrated in Figs. 18-21. Total program costs for various program levels were estimated for different sized O_2/H_2 and shielded nuclear vehicles, and a reasonable size was selected for each (Figs. 18, 19) based on minimizing the total program cost for a reasonable 10-yr payload-delivery program. (Payload is assumed to be divisible arbitrarily into packages just large enough to fill the capacity of each different vehicle; real programs generally produce payloads sized to the mission rather than the transporation system.) The true program is unknown, of course, so the object is to select a size giving near-minimum costs over a variety of traffic levels. This criterion resulted in selection of a 100,000-lb-propellant chemical vehicle in Fig. 18 and a 290,000-lb-propellant nuclear vehicle in Fig. 19. (The latter was admittedly influenced by the fact that this value is also close to that being considered for the projected nuclear stage.) The pie charts in Fig. 20 show the relative contribution of different cost components to the total program cost for the two vehicles at each of the four traffic levels. The circles

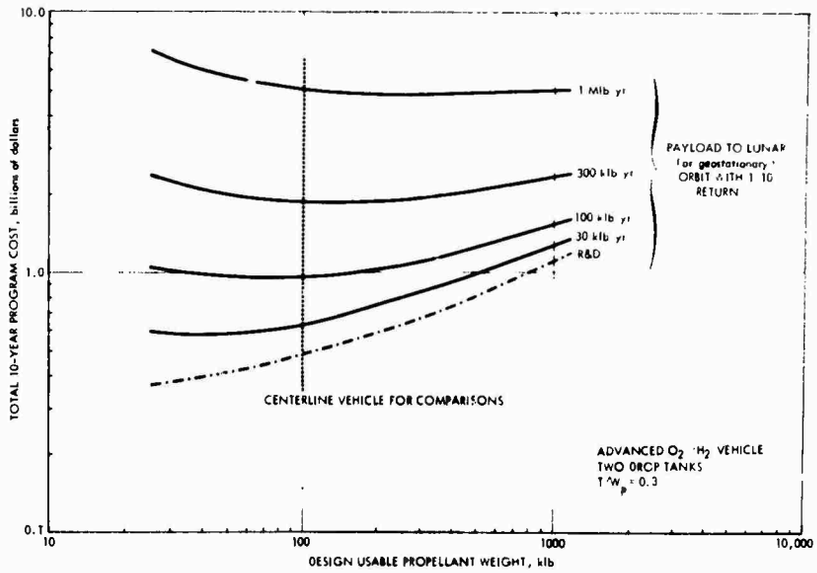

FIGURE 18. TOTAL PROGRAM COSTS (R&D PLUS 10 YEARS OF OPERATIONS AT FOUR DIFFERENT RATES) AS FUNCTIONS OF DESIGN PROPELLANT CAPACITY FOR ADVANCED O_2/H_2 VEHICLE WITH OPTIMUM NUMBER OF ENGINES AND 2 DROP TANKS.

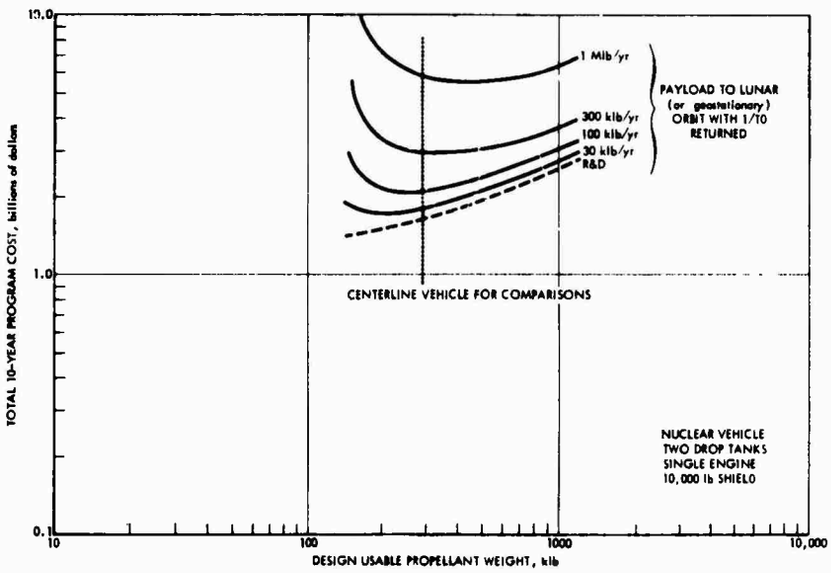

FIGURE 19. TOTAL PROGRAM COSTS (R&D PLUS 10 YEARS OF OPERATIONS AT FOUR DIFFERENT RATES) AS FUNCTIONS OF DESIGN PROPELLANT CAPACITY OF NUCLEAR VEHICLE WITH SINGLE ENGINE, 10,000-LB SHIELD AND 2 DROP TANKS.

for the O_2/H_2 vehicle are normalized in area to show the relative cost to the nuclear system at each traffic level. The principal cost contributors, together more than 90 percent of the total, are R&D and the EOS delivery, the former predominating at lower traffic levels and the latter at higher traffic levels. The total program costs for each traffic level are plotted in Fig. 21 to show the crossover concept for these specific vehicles and include similar current O_2/H_2, F_2/H_2, and unshielded nuclear vehicles.* Note that a significant payload to lunar orbit, approximately 1.5 Mlb, is required on this basis to justify any propulsion development beyond the present RL-10 technology. The 1/10th-return capability from lunar orbit for all these small chemical vehicles may, however, be somewhat marginal (i.e., 2500 lb for the RL-10, 3300 lb for advanced O_2/H_2, and 3600 lb for F_2/H_2), and particularly so for the RL-10. A comparison at equal return capabilities, if larger than these values, would have increased the program costs for the lower performance vehicles and lowered the crossovers.

*Neither the nuclear nor the chemical system has been charged with costs for the lunar lander needed for lunar-surface operation. The chemical system could probably share much of its technology and hardware with the lander, however, so that the comparison favors the nuclear system by not penalizing it for the requisite separate lander development program.

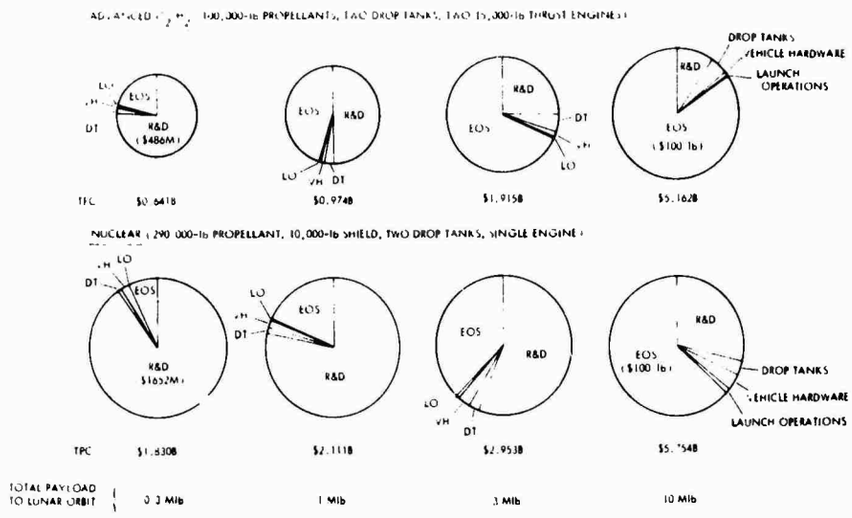

FIGURE 20. COMPARATIVE SIZES OF COST COMPONENTS OF CHEMICAL AND NUCLEAR SYSTEMS FOR FOUR DELIVERY PROGRAMS. 1/10 RETURN MISSION.

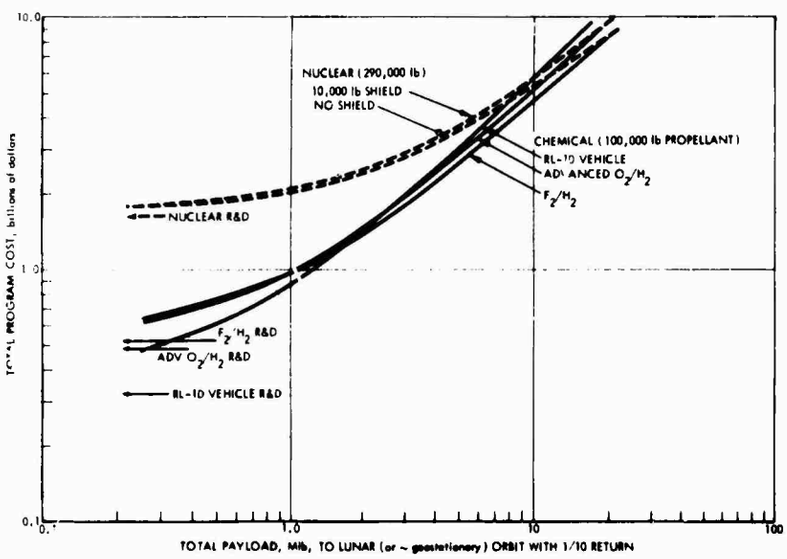

FIGURE 21. TOTAL PROGRAM COSTS FOR CUMULATIVE PAYLOAD WEIGHT DELIVERED BY CENTERLINE VEHICLES TO LUNAR ORBIT.

Figure 22 shows how program cost differences, i.e., the excess cost of the nuclear system over the advanced O_2/H_2 system, decrease with increasing total payload delivered, at rates that vary with EOS costs. The intercepts with the abscissa show crossover payloads to increase as EOS costs decrease; this is perhaps self-evident in that fixed differences in R&D cost are being divided by proportionately smaller absolute differences in operational cost. Of interest, however, is the near constancy of total program costs indicated at the crossover. It appears that $9 or $10 billion in lunar transportation costs must be expended to recover the additional investment needed for nuclear propulsion development if a 10/1 outbound/return payload ratio is

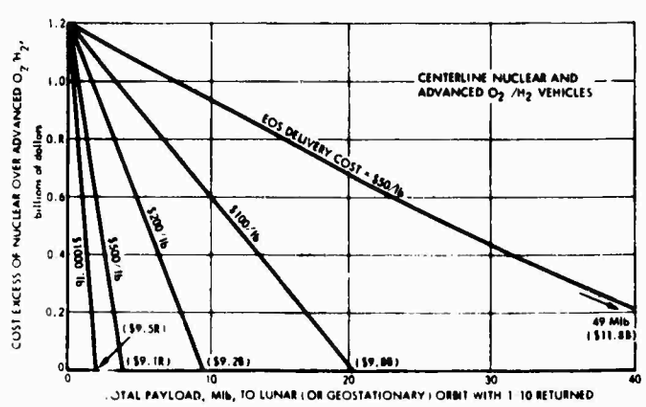

FIGURE 22. INFLUENCE OF EOS DELIVERY COST ON DIFFERENCE IN TOTAL PROGRAM COST BETWEEN CENTERLINE NUCLEAR AND ADVANCED O_2/H_2 VEHICLES.

assumed and if this is the only application of the nuclear system. This near constancy is also not too surprising, at least after the fact; EOS costs dominate program costs, and nuclear systems have a certain percentage advantage over O_2/H_2 in propellant consumption per pound of payload. The difference in R&D costs divided by the percentage saving dominates the total program cost at crossover. Or, more precisely, by considering Fig. 14b, note a 13 percent difference between nuclear specific operations cost ($410/lb) and advanced O_2/H_2 ($165/lb). A $1.2 billion difference in R&D cost divided by a 13 percent saving implies a $9.2 billion total program cost at crossover.*

Figure 23 depicts most of the results of the sensitivity analysis, showing the changes in crossover from the nominal 20.3 Mlb due to variations in the controlling parameters. A few additional single-point evaluations of sensitivities are noted below.

The ranges of variation discussed are chosen to be larger than expected deviations from the baseline values. As mentioned earlier, the reciprocal nature of the crossover concept makes it a sensitive parameter; at infinite crossover, operating costs are equal.

The curves in Fig. 23 show the marked sensitivity of the nuclear system to average load factor (in line with observations on off-loading in Fig. 16), engine weight, development cost, and specific impulse, and to relative costs of delivering LH_2 versus O_2/H_2. Nuclear engine life does not have much impact unless it gets below 4 hours or so. The smaller effects of improvements in nuclear engine costs, thrust, and other nuclear stage parameters can be noted from the plots.

With regard to sensitivities involving the O_2/H_2 system, the crossover is moderately sensitive to O_2/H_2 structural weight, to outbound-to-return ratio

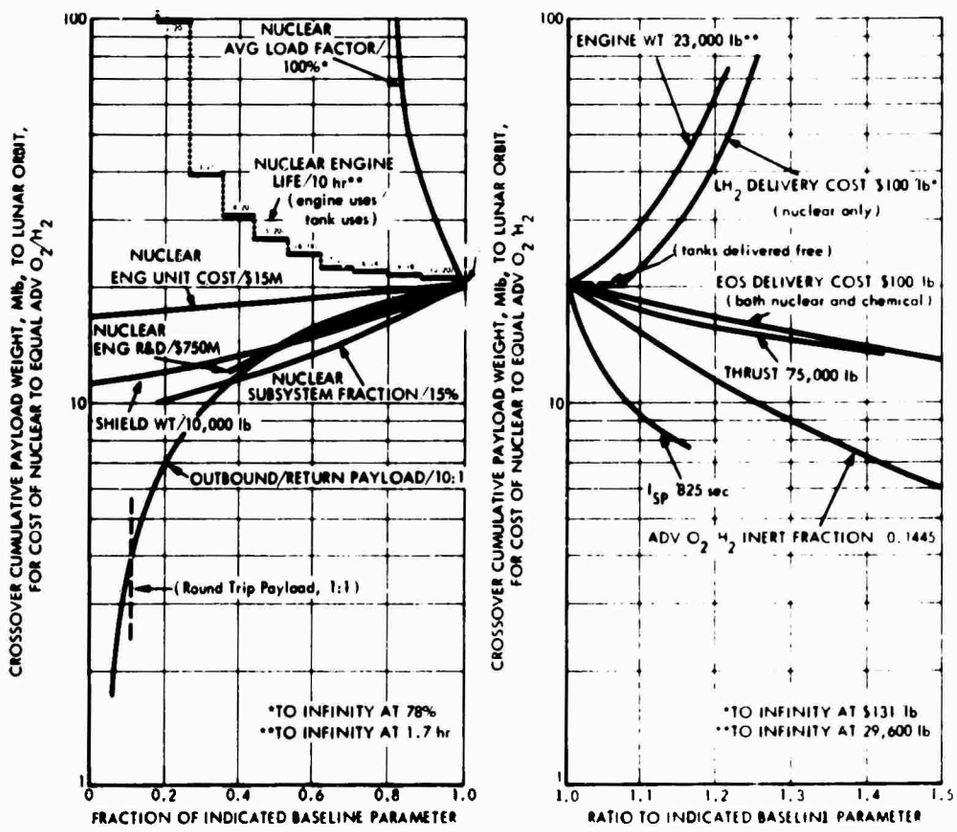

FIGURE 23. INFLUENCE OF VARIATION OF BASELINE ASSUMPTIONS ON CROSSOVER CUMULATIVE PAYLOAD (10/1). CENTERLINE VEHICLES

*For this return ratio and these two size levels.

and, not plotted, to specific impulse. A 10-second decrease in O_2/H_2 specific impulse decreases the crossover to 14.5 Mlb. conversely, substitution of F_2/H_2, as can be appreciated from Fig. 21, raises the crossover beyond 100 Mlb. An increase in astrionics-unit weight in both systems affects the lighter chemical system more than the heavier nuclear system. this effect is somewhat different from the illustrated structural weight change because of the higher specific cost of electronics and for, say, a 1500-lb increase, brings the crossover to 11.3 Mlb. The effect of an increase in O_2/H_2 R&D costs is the same as that from a decrease in nuclear engine R&D costs by the same amount. A $10 million increase in first unit cost of the O_2/H_2 system (above $7.1 million) lowers the crossover to 16.5 Mlb.

The effects described are not fully linear either individually or in combination. However, a rough estimate of the combined effects of a number of changes can be made by multiplying sensitivities (i.e., changed crossover/nominal crossover) together, and applying the product as a correction factor to the nominal crossover. This product in most cases will predict a lower crossover value than the one obtained from a recalculation of the modified system with the changes introduced simultaneously. To illustrate, assume the following large changes all acting in the same direction (against the chemical system):

	Change	Crossover Ratio
(1)	nuclear shield weight reduced to zero	0.555 (=11.3/20.3)
(2)	difference in R&D cost reduced by $500M	0.631
(3)	chemical first unit cost increased by $10M	0.813
(4)	chemical specific impulse reduced by 10 sec	0.715
(5)	payload return ratio doubled (to 20%, for both)	0.675
(6)	EOS delivery cost to $120/lb (for both)	0.811
(7)	astrionics weight increased by 1500 lb (both)	0.557

The product of these sensitivity values (0.062) would predict a crossover at 1.3 Mlb versus 2.4 Mlb computed with all effects incorporated simultaneously. Actually, if this magnitude of effects were expected, the vehicles should be reoptimized.

APPENDIX

BASELINE ASSUMPTIONS AND GROUND RULES

Effects

Average I_{sp} nuclear for full thrust burn time t ($t \geq 100$) sec

odd burns: $\bar{I} = F_1/F_2$; even burns: $\bar{I} = (F_1 - 356C_1)/F_2$

where: $F_1 = 3.195\times10^6 + 75{,}000t + 460S_1 + 356C_1$

$S_1 = 860 + 0.75t - 296t^{-0.7} - 7\times10^{-5}t^2$ (shutdown integrated flow)

$C_1 = 32.8t^{0.734}$ (cooldown integrated flow, lb)

$F_2 = 4557 + (75{,}000/825)t + S_1 + C_1$

356 sec is effective cooldown specific impulse, allowing for low thrust losses

H_2 boil-off = 0.025 lb/day/ft^2 of tank area + 1 lb/sec of burn time (nuclear only)

$$\text{Meteoroid protection and insulation}\left\{(\text{lb/ft}^2)\right. = 0.21\left(\frac{\text{Total flight life}}{45\text{ days}}\right)^{0.25} \times \left(\frac{\text{Propellant vol}}{68{,}000\text{ ft}^3}\right)^{0.167}$$

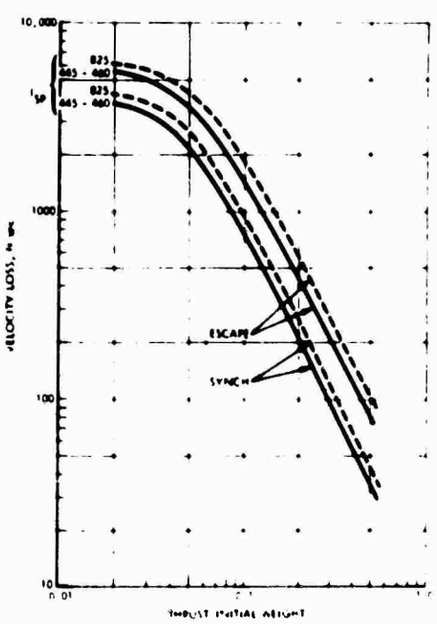

FIGURE A-1 GRAVITY LOSSES FOR LOW-THRUST TAKEOFF FROM 100-NMI-ALTITUDE EARTH ORBIT FOR SYNCHRONOUS-ALTITUDE APOGEE AND ESCAPE

Gravity losses as a function of initial thrust weight ratio and I_{sp} for acceleration from a 100 nmi orbit to escape or synchronous-orbit transfer are shown in Fig. A-4. The velocity loss in injection into a transfer ellipse to synchronous altitude from 263 nmi is reproduced by

$$\Delta V_{loss} = (I_{sp}/445)^{0.31} (W_o/T)^2 [11.2 - 1.13 (W_o/T)^{0.55}] \left(\frac{g_{263}}{g_{100}}\right)$$

The accompanying velocity gain at synchronous altitude due to raising the altitude at which transfer velocity is attained is reproduced by

$$\Delta V_{gain} = (I_{sp}/445)^{0.27} (W_o/T)^2 [2.18 - 0.00072 (W_o/T)^2]$$

The velocity loss in attaining escape velocity from 263 nmi orbit is reproduced by

$$\Delta V_{loss} = (I_{sp}/445)^{0.25} (W_o/T)^2 [76.1 - 50 (W_o/T)^{0.1}] \left(\frac{g_{263}}{g_{100}}\right)$$

A "pad" of 0.75 ϵ is also allowed for each velocity increment required.

Weights

Contingency	= 3% of dry weight
Ullage volume	= 3% of propellant volume
Residuals	= 1% of usable propellants
Subsystems (fraction of dry weight)	= 3% (chem)
	= 15% (nuc)
	= 1.5% (drop tanks)

Weight Relations

$$W_{inert} = W_{tank} + W_{plumbing} + W_{subsystems} + W_{residuals} + W_{main\ propulsion} + W_{thrust\ structure} + W_{guidance\ control} + W_{contingency}$$

Where

$$W_{tank} = S\,(2.32 + MP + 48\,S/10{,}000\ ft^3) \times \left(\frac{\text{propellant density}}{20.24\ lb/ft^3}\right)^{1/2}$$

MP = meteoroid protection (includes insulation, see above)

$$S = 4.5\pi \left(\frac{\text{tank volume}}{\pi}\right)^{2/3}$$

$W_{plumbing}$ = 0.1 lb/ft^3 of tank volume

$W_{main\ propulsion}$* = 300 lb (current O_2/H_2)

350 lb (advanced O_2/H_2 and F_2/H_2)

23,000 lb (nuclear)

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* Per 15 klbf thrust chemical engine, or 75 klbf nuclear engine.

$W_{thrust\ structure}$ $0.0025\ T_{vac}$

$W_{guidance/control}$ $300\ lb + 0.1\ (W_{inert} - W_{residuals})$

Costs

Assumptions

Life - 10 hr for engines

- 10 hr or about 20 flights (see text) for tanks

Earth-to-orbit shuttle delivery cost $100/lb

Learning curve slope - 90%

Correlations

First Unit Costs (1969 dollars):

Chemical engine - $756\ (T_{vac}(lb))^{0.61}\ (p_c/1000\ psi)^{0.2}$

Nuclear engine = 15×10^6

Airframe = $3.078 \times 10^3\ (W_{AF}(lb))^{0.639}$

where W_{AF} $W_{inert} - W_{mp} - W_{gc} - W_{res}$

Subsystems - $2.4 \times 10^4\ (W_{gc}(lb))^{0.725}$

R&D Costs (1969 dollars):

Chemical engine $50 \times 10^6 + 4.8 \times 10^6 (T_{vac}(lb))^{0.32}\ (P_c/1000\ psi)^{0.2} \times \alpha$

$$\text{where } \alpha = \begin{cases} 0 & \text{current } O_2/H_2 \\ 1.5 & \text{advanced } O_2/H_2 \\ 2.0 & F_2/H_2 \end{cases}$$

Nuclear engine 750×10^6

Airframe $0.759 \times 10^6\ (W_{AF}(lb))^{0.578}$

Subsystems 25 x first unit cost

Checkout costs per flight 1% vehicle hardware cost

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