AD 744979

PAPER P-796

ON OPTIMIZING THE DESIGN OF SPACE-BASED TRANSPORTATION SYSTEMS

R. C. Oliver
R. G. Finke

October 1971

D D C
RECEIVED
APR 13 1972
B

NATIONAL TECHNICAL
INFORMATION SERVICE

INSTITUTE FOR DEFENSE ANALYSES
SCIENCE AND TECHNOLOGY DIVISION

DISTRIBUTION STATEMENT A
Approved for public release;
Distribution Unlimited

IDA Log No. HQ 71-12919
Copy 57 of 75 copies

The work reported in this document was conducted under contract DAHC15 67 C 0011 for the Department of Defense. The publication of this IDA Paper does not indicate endorsement by the Department of Defense, nor should the contents be construed as reflecting the official position of that agency.

Approved for public release; distribution unlimited.

ACCESSION for
CFSTI WHITE SECTION ☑
DDC BUFF SECTION ☐
UNANNOUNCED ☐
JUSTIFICATION .....
BY .....
DISTRIBUTION/AVAILABILITY CODES
DIST. AVAIL. and/or SPECIAL
A

PAPER P-796

ON OPTIMIZING THE DESIGN OF SPACE-BASED TRANSPORTATION SYSTEMS

R. C. Oliver

R. G. Finke

October 1971

INSTITUTE FOR DEFENSE ANALYSES
SCIENCE AND TECHNOLOGY DIVISION
400 Army-Navy Drive, Arlington, Virginia 22202

Contract DAHC15 67 C 0011
Task T-58

UNCLASSIFIED
Security Classification

DOCUMENT CONTROL DATA - R & D
(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) INSTITUTE FOR DEFENSE ANALYSES 400 Army-Navy Drive Arlington, Virginia 22202	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b. GROUP --

3. REPORT TITLE
On Optimizing the Design of Space-Based Transportation Systems

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)
Paper P-796 - October 1971

5. AUTHOR(S) (First name, middle initial, last name)
R.C. Oliver and R.G. Fiske

6. REPORT DATE October 1971	7a. TOTAL NO. OF PAGES 55	7b. NO. OF REFS 12
8a. CONTRACT OR GRANT NO. DAHC15 67 C 0011 b. PROJECT NO. Task T-58 c d	9a. ORIGINATOR'S REPORT NUMBER(S) P-796 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) None	

10. DISTRIBUTION STATEMENT
Approved for public release; distribution unlimited.

11. SUPPLEMENTARY NOTES None	12. SPONSORING MILITARY ACTIVITY Advanced Research Projects Agency Arlington, Virginia 22209

13. ABSTRACT
The factors involved in selecting propulsion stages to give minimum costs for future space-based transportation systems are outlined. A parametric procedure is described for selecting such minimum cost vehicles and applied to generalized and specific (lunar exploration and geostationary transfer) missions. The results indicate the different regimes wherein reusable and expendable vehicles (or mixes thereof) in single- or two-stage versions should give minimum costs. The results also show that for preferred configurations the costs of space transportation are essentially proportional to earth-to-orbit costs.

DD FORM 1 NOV 65 1473

UNCLASSIFIED
Security Classification

UNCLASSIFIED
Security Classification

14 KEY WORDS	LINK A ROLE	LINK A WT	LINK B ROLE	LINK B WT	LINK C ROLE	LINK C WT
chemical rocket propulsion						
space transportation						
geostationary vehicles						
lunar exploration						
expendability versus reusability						
cost minimization						

UNCLASSIFIED
Security Classification

FOREWORD

The authors wish to thank Messrs. Robert S. Swanson and James H. Henry of IDA and Don A. Hart of NASA for their many helpful criticisms and suggestions in reviewing this document.

A preliminary report on this work (with the same title as this paper) was given at the XXIInd International Astronautical Federation Congress in September 1971 in Brussels, Belgium.

ABSTRACT

The factors involved in selecting propulsion stages to give minimum costs for future space-based transportation systems are outlined. A parametric procedure is described for selecting such minimum cost vehicles and applied to generalized and specific (lunar exploration and geostationary transfer) missions. The results suggest the different regimes wherein reusable and expendable vehicles (or mixes thereof) in single- or two-stage versions should give minimum costs. The results also show that for preferred configurations the costs of space transportation are essentially proportional to earth-to-orbit costs.

NOMENCLATURE

ΔV	velocity increments, fps
EOS	earth-to-orbit shuttle
fps	feet per second
klb	kilopound
OCPM	operating cost per mission
OOS	orbit-to-orbit shuttle
TPC	total transportation program costs
E1	expendable first stage (or single stage)
E2	expendable second stage
R1	reusable first stage (or single stage)
R2	reusable second stage
R1 + 2DT	reusable stage with two expendable drop tanks
E1R2′	expendable first stage plus reusable second stage, with the second stage used only on the return trip
E1R2 E1E2 R1R2 R1E2	As per code above, using equal propellant weights in each stage (except as noted in the text). This definition generally gives minimum total development costs within the assumptions made.

CONTENTS

I.	SUMMARY AND CONCLUSIONS	1
II.	INTRODUCTION	4
III.	DISCUSSION	8
	A. Space Transportation Cost Factors and Selection of Optimum Configurations	8
	B. Reusability Versus Expendability--Some Qualitative Arguments	9
IV.	COMPUTATION PROCEDURES AND PERFORMANCE PARAMETERS	11
	A. General	11
	B. Performance Parameters	13
V.	MISSION STUDY PROCEDURE AND RESULTS	18
	A. Mission Study Procedure--Baseline Design Conditions	18
	B. Cargo-Carrying Vehicles	19
	C. Lunar and Geostationary Mission Studies	34
VI.	CONCLUDING COMMENTS	49
	References	51
	Appendix--Additional Assumptions and Correlations	52

I. SUMMARY AND CONCLUSIONS

A space-vehicle design procedure involving empirical weight and cost relationships has been developed which permits investigation of a wide range of the variables that will be involved in the optimization of future space transportation systems. This procedure has been utilized to suggest preferred configurations and sizes for three classes of potential space-based* vehicles: the first, a class of generalized cargo-carrying vehicles, delivering payload one way to arbitrary velocities; the second, a class encompassing vehicles that might be considered for a lunar manned exploration mission with payload return each flight; and the third, a class involving vehicles for one-way geostationary missions. The study is arbitrarily limited to configurations that involve no more than two stages and two propulsion modules. The analyses emphasize arbitrary, fairly large payload delivery programs, and cost, weight, and performance factors which imply some improvements over the current state of the art.

The results for the generalized cargo-carrying vehicles suggest the regimes that exist wherein various staging and stage reuse conditions should give minimum costs. In general, higher earth orbital shuttle (EOS) costs (i.e., cost of delivery to earth orbit) and higher ΔV requirements make partially or totally expendable vehicles increasingly attractive compared to fully reusable vehicles. In essence, the costs of delivering the propellant (necessary to bring the stage back)

*Space-basing here implies availability of orbital facilities for propellant storage and vehicle servicing, and hence constant specific delivery costs (i.e., per pound) to orbit rather than constant per-flight costs to earth orbit for all payloads up to some vehicle capacity.

become greater than the stage is worth. Single-stage reusable vehicles are attractive in the lower velocity regime, approximately up to escape velocity or about 10,000 ft/sec above orbital. Above this velocity, two-stage reusable vehicles, partially expendable or mixed reusable-expendable vehicles, and expendable vehicles would, in turn, be the respective preferred configuration. At extreme velocities (beyond about 30,000 ft/sec above orbital), two-stage expendable vehicles should be utilized for minimum cost. Reusable vehicles will be preferred to higher velocities than indicated if the hardware costs exceed those predicted herein. Operating costs of reusable systems are almost directly proportional to EOS delivery costs, and thus, in contrast with expendable systems, are largely independent of uncertainties in hardware cost correlations; the operating costs of reusable systems, however, are sensitive to weight uncertainties, particularly for difficult missions. The greater development costs of reusable systems relative to expendable systems will lead to a preference for expendable systems (where acceptable) in programs involving only a small number of flights.

The lunar exploration study considered the problem of vehicle size optimization in the presence of an unknown total program. It was found that, assuming 10,000 lb is to be returned each trip from the lunar surface, and assuming a nominal NASA goal of $100/lb as an EOS cost, a combination of an earth-orbit-to-lunar-orbit transfer vehicle and a lunar lander delivering about 60,000 lb (58,000 lb was selected) to the surface would be appropriate on a minimum total transportation program cost basis for a manned lunar exploration program. A two-stage reusable vehicle (coupled with a lunar lander) involves the lowest costs for this mission, but a single-stage reusable (with free-return rendezvous) or a reusable stage with drop tanks (lunar-orbit rendezvous) is only modestly (less than 15%) more expensive, and might have higher reliability. It was noted that for manned lunar exploration, the free-return rendezvous option involves slightly lower costs than the more conventional lunar-orbit rendezvous, and may be of interest from an operational standpoint.

For the geostationary mission, if part of a large program, it was again concluded that there is generally a cost advantage, at $100/lb EOS cost, to the use of two-stage reusable vehicles over other configurations. As is the case in the other missions, however, other configurations have other advantages. The single-stage expendable vehicle, for example, although of the order of 10% higher in operating costs, would be considerably smaller and simpler than any reusable configuration.

Substantial cost penalties are involved in the use of vehicles oversized for a particular payload desired. This observation suggests that vehicles should be sized toward the small end of the payload spectrum, and clusters of vehicles used when needed to accelerate large payloads.

The study also considered the advantages of advanced propulsion for volume-unconstrained space-based systems on a purely economic basis, i.e., without considering possible advantages to the EOS system that result from the greater average propellant density of advanced systems over present systems. On this limited basis, while advanced propulsion systems are more attractive for reusable vehicles than for expendable vehicles, the small operating cost savings make the rather high development costs difficult to justify.

The study also demonstrates that for preferred configurations the operating costs of space transportation systems are almost proportional to earth-to-orbit costs. Achievement of low cost to earth orbit is thus clearly the key to low-cost space transportation. If $100/lb to earth orbit can be achieved, extensive lunar exploration, for example, becomes economically feasible.

II. INTRODUCTION

The coming generation of vehicles for space transportation will, according to plan, include some form of earth-to-orbit shuttle (EOS), capable of delivering payloads to low earth orbit at low cost and, very likely, some additional space stage, perhaps an orbit-to-orbit shuttle (OOS), for carrying out higher energy missions. The EOS has, as its objective, the capability to deliver payload to low earth orbit at low cost, with $100/lb a nominal goal (based on 1969 figures of $5,000,000 per launch carrying 50,000 lb to orbit). However, due to questions of total and peak annual funding, the EOS design at the time of writing (October 1971) is in a state of flux, so that any additional staging designed to supplement it is necessarily even less firmly fixed.

The EOS and OOS, in various forms, have been the objects of numerous studies (as examples, Refs. 1-9) with the greatest emphasis on the EOS. Most OOS studies have been in two broad categories: the first for vehicles in the 50,000-lb-gross-weight class associated with specific cargo bay limitations of hypothetical EOS vehicles; and the second for much larger vehicles (in the 300,000-lb class) often utilizing nuclear propulsion. To some degree these OOS studies have been "point" designs, or at least of limited scope in a parametric sense.

Because of the above considerations, and in view of the highly tentative nature of future space stages, it was decided to approach the problem of space transportation systems design in a purely parametric way recognizing the importance of costs and illustrating with examples certain aspects of such systems that may impact on system

selection. In carrying out the work, in order to remove any constraints associated with cargo-bay and weight-carrying capabilities of the still undefined EOS vehicles, it was assumed that the space transportation system would be "space-based," which implies orbital facilities for propellant storage, vehicle repair and assembly, and space "garages" for long-term meteoroid protection.* On this basing assumption, more than one EOS flight might be required for a single flight of the space transportation system. Furthermore, on any flight of the EOS, an excess weight-carrying capacity could normally be used to carry propellant or other material to orbit for storage. Thus a reasonable parameter to be used here for earth-to-orbit costs (which has considerable significance in the total cost picture) is a specific cost (in dollars/lb) rather than a quantized per-flight total cost, as might be more reasonable for a ground-based system.

It should, perhaps, be pointed out that the first space stages to be used in connection with an EOS will likely be ground-based rather than space-based; i.e., they will, if reusable, be returned to earth after each use rather than being refueled in orbit. Examination of space-based stages is worthwhile, however, since the considerations involved do impact on shuttle design; e.g., if a two-stage space vehicle is found to be preferable for future space missions, the EOS cargo bay should have a greater length to diameter ratio than would be the case if the preferred space vehicle were to involve a single stage. Compatibility with both types of basing is desired. Gregory (Ref. 9) makes the point that space staging considerations should enter into EOS design. The design of the space stage and the EOS is clearly an iterative process.

The material that follows reports on several aspects of the effort. Section III is a discussion of the general problem, particularly

---

*"EOS costs" herein thus include any in-orbit assembly costs and in-orbit transfer losses. Possible differences in assembly costs between reusable and expendable systems are not considered.

with regard to factors affecting cost. Section IV outlines the computational procedures and discusses certain parameters that are critical to stage performance. Section V presents the results of the several mission studies that have been chosen as being representative of future missions of interest. Some concluding comments are offered in Section VI.

The mission studies are intended to illustrate applications of the techniques developed and, while preliminary in nature, to be of interest in their own right. Three missions were studied:

1. An unmanned cargo-carrying mission of arbitrary ΔV requirement with no payload return requirement. Here the object is to show the effects of various parameters on cost, and to indicate regimes of EOS cost-ΔV requirement where different configurations are most attractive on a cost basis.
2. A lunar exploration mission where payload is to be returned each trip; the problem is to select perferred configurations and optimum vehicle size.
3. A geostationary mission, without payload return requirement, in which the costs and certain other aspects of various configurations are considered.

A number of technological questions are explored along with the various mission studies.

As the mission study is purely parametric and intended to be internally consistent, no review of previous studies or of the applicability of existing stages (CENTAUR, BURNER II, and AGENA) is attempted. Some studies on existing stages have been done elsewhere (Ref. 8). In minimized R&D costs systems, however, the use of an existing O_2/H_2 engine (the RL10-A-3-3*) is considered, as it appears to be in a thrust regime of interest. The study considers only O_2/H_2 and F_2/H_2 as propellants; nuclear propulsion (not considered here) was the subject of a previous paper (Ref. 3). Vehicles with a single stage, with one stage

*Hereinafter referred to as "current" O_2/H_2.

and propellant drop tanks, and with two stages are considered; however, in the two-stage studies, no more than two propulsion modules (i.e., no parallel clustering) are considered.

While this study considers some effects of changes in costs, it will give emphasis to the nominal 1969 NASA goal of $100/lb to orbit. This may be unrealistic for some of the interim vehicles now being considered, but this figure (100 1969 dollars) may not be a bad one for use in considering space-based vehicles that will not be in use for a decade or more. Because of the uncertainties involved and to be consistent with a previous study (Ref. 3), no attempt was made to revise a cost model expressed in 1969 dollars. Discounting and scheduling were not considered.

No consideration is given to the cost of payloads in the examples reported; if payload costs could be defined, it would be a relatively easy matter to include such costs in the computer program developed. Tradeoff between expendable and reusable systems would be affected by payload costs, but mission-dependent factors would need to be considered carefully.

III. DISCUSSION

A. SPACE TRANSPORTATION COST FACTORS AND SELECTION OF OPTIMUM CONFIGURATIONS

Two broad classes of cost are of concern here: total transportation program costs and operating costs. Total transportation program costs include all necessary additional R&D costs as well as total operating costs from program initiation through some specified number of flights. Operating costs on the other hand are computed assuming all R&D costs are sunk, and the only expenditures necessary are those to buy the required number of designated, previously developed vehicles and to carry out the desired number of flights. Total transportation program costs are of primary interest to the original developer; operating costs are of primary interest to a secondary user.

There are many variables affecting cost and these interact differently for different missions and programs. In general, if payload, mission, and number of missions are specified, one can select a configuration, propulsion system, etc., which will minimize the total cost associated with <u>that</u> number of missions for <u>that</u> payload. This configuration may still not be the preferred system, for this minimum in total program cost might well be achieved by doing a minimum of R&D, and the resultant system may have significantly higher operating costs than one with higher R&D costs and may not meet other criteria which are less easy to quantify, such as reliability or growth potential.

R&D costs, in general, are determined by the state of technology available and/or desired for the components, by the complexity of the mission(s), and by the size of vehicle required. The size of vehicle required is, in turn, dependent on the divisibility of the <u>total</u> payload that must be transported. Supplies and propellants for a moon

colony would appear to be subdividable in delivery to orbit whereas a large telescope, for example, might not be.

Operating costs are affected by the earth-to-orbit costs, and by checkout, refurbishment, and other costs. Operating costs are also affected by the specific impulse of the propellants, by the staging used, by what parts are reused, and by what parts are expended, etc. Operating costs per unit of payload usually decrease with vehicle size, and R&D costs increase with vehicle size, so that an optimum size exists giving minimum total transportation program costs with arbitrarily subdividable payloads.

The mission itself affects costs, with costs increasing as a function of total energy requirement as well as trip time, the latter factor entering into insulation (boil-off) and meteoroid protection requirements. Payload return requirements obviously affect costs, as twice the velocity requirement is involved in a round-trip mission as in a one-way mission. Manned systems, besides having obviously greater redundancy and reliability requirements, thus involve substantial penalties which must be compensated for by other factors.

Costs, either total program or operating, can be studied as a function of any or all of the many variables such as ΔV requirement, payload size, total program size, state of technology, configuration, expendability, and so forth. All of these will be touched on in the material that follows. However, there is one aspect of the problem that seems to be of particular interest: the question of reusability versus expendability; this point will be discussed in the next section and given special emphasis in the results.

B. REUSABILITY VERSUS EXPENDABILITY--SOME QUALITATIVE ARGUMENTS

It may be helpful to understand the results to be presented in discussing qualitatively the way reusability affects costs.

Superficially, reuse of expensive space-based stages would generally appear to be attractive. Only propellants and payload would need to be brought to earth orbit in each new mission by the EOS (once

the stage itself is in orbit) and the cost of the stage would be written off over a large number of missions. In contrast, if an expendable stage is used, the stage (or its components) in addition to the payload and all propellants must be carried to orbit before each mission, and the full cost of the space-based stage must be charged to each flight. The problem becomes more complicated on examination, however. First (and dominating the comparison), in order to reuse a stage, enough additional propellant (over and above that required for the expendable stage) must be carried to earth orbit to bring the empty reusable stage back and to accelerate the return-leg propellant to the same velocity as the payload itself. Propellant in orbit is expensive, its effective cost being at least as great ($100 or more per pound) as the EOS delivery costs (the propellant cost itself is nearly negligible and boil-off and transfer losses in low earth orbit are ignored here). Clearly, as mission ΔV's are increased, at some point it will be found that it cost more to haul the extra propellant to orbit than will be saved by recovery of the hardware. Furthermore, the reusable stage, which is larger for a given payload and has a longer life requirement, will cost more for development than will the expendable stage, adding a penalty in terms of total transportation program costs. In addition, only a small number of reusable stages will be required for a large number of missions, so that the manufacture of such stages is at a point much nearer to first unit cost on the "learning curve" than is the case with expendable stages after the same number of flights. There are other factors that penalize the reusable stage also, such as a requirement for greater component life, more meteoroid and insulation protection, etc.

The reader is referred to the article by Gregory (Ref. 9) for a further illustration of the relative merits (i.e., operating costs, in his case) of reusability and at least partial expendability.

IV. COMPUTATION PROCEDURES AND PERFORMANCE PARAMETERS

A. GENERAL

This section describes the procedure by which the computations are made, and some general performance parameters. Section V describes the study procedure for the missions examined, including the baseline conditions assumed, and gives the results.

The problem here is to determine how much vehicles of different configurations, all performing the same mission, would weigh and cost, and how these relative costs would be affected by changes in the controlling parameters. In doing so, the basic rocket design equation, $\Delta V = \Sigma I_i g_c \ln R_i$, is utilized, wherein ΔV is the total mission velocity including losses, I_i is the specific impulse in stage i, g_c is the units conversion constant, and R_i is the ratio of vehicle weight, including payload, at the beginning and end of stage i burn. The inert weight associated with a given type of stage (as shown later in Fig. 2) varies in a complex way with propellant weight and volume. The problem requires iterative solution, for the velocity losses depend on thrust-to-weight ratio, which here varies stepwise as the number of engines change, and the vehicle inert weights vary nonlinearly with propellant mass and volume. Other, more subtle effects, such as boil-off losses, must be included in some cases, as in moon exploration missions. Where more than one stage is involved, the problem is further influenced by the staging ratio assumed.

The iterative solution process utilizes open-form weight relations. An initial guess is made assuming a propellant weight. From this, and an input thrust-to-weight requirement, total inert weights are determined. Given these data, and the input payload, specific impulse and ΔV requirement, a new propellant weight is calculated from

the rocket equation (i.e., the amount of propellant needed to give the required R), and the process repeated until a stable solution is found. Once the vehicle is sized, the vehicle is costed using cost correlations to give the total transportation program and operating costs for the number of flights considered. Learning curves (90%) are applied to purchased hardware in determining operating costs. Suboptimization is necessary to select thrust-to-weight; lower thrust-to-weight involves lower engine weights and costs but higher ΔV losses. The various weight and cost relationships used in the iterative design and costing program are appended. These relationships have been built up over the past 6 years for use primarily in reusable launch vehicle studies (Refs. 1,2).

Velocity (or gravity) loss correlations were generated in an earlier paper from detailed kinematic calculations (Ref. 3) for geostationary and lunar orbit missions. As the data at escape velocity agreed well with prior published data (Ref. 10), the latter data were used to extend loss estimate correlations to velocities beyond escape.

Guidance weights and costs are included in all stages except for expendable lower stages in two-stage systems.

One important point is to be noted with regard to vehicle cost optimization. In some cases it is found that two-stage vehicles have lower operating costs than do single-stage vehicles. In order to minimize vehicle R&D costs, and to take advantage of learning, it is usually attractive to make the two stages as nearly identical as possible, differing perhaps only in the number of engines. In this work, staging velocities were sought which yielded equal (within 1%) propellant-mass stages; if this criterion was met, the total vehicle R&D cost was set equal to that of the more complex of the two stages involved. If lower operating or total transportation program costs were achieved at some other staging velocity (as does happen at very high ΔV requirements), each stage was charged with its full R&D program.

In using the computer program developed, one must specify the payload desired, the propulsion type utilized, the staging being used, whether a stage is reusable or expendable, and the EOS costs and other variables that may be involved. As output, the calculations give vehicle weight, total 10-year transportation program costs for arbitrary traffic levels that may be selected, and average operating costs per pound for the number of missions specified.

B. PERFORMANCE PARAMETERS

Certain performance specifications are of particular importance to this analysis. The first of these, propellant specific impulse, is shown in Fig. 1. This figure shows the estimated specific impulses of oxygen/hydrogen and fluorine/hydrogen propulsion systems at the 15,000-lb-thrust level as a function of area ratio; the data for the curves are from Ref. 11. Note that by going to very high area ratios, high-performance oxygen/hydrogen and fluorine/hydrogen propulsion systems are possible. Calculations made in the course of this work suggested that area ratios as high as 400:1, in spite of the weight penalty,* are attractive on a cost basis. Three selected points are shown as being representative of current O_2/H_2 (57:1), advanced O_2/H_2 (400:1), and F_2/H_2 (400:1). Standardized 15,000-lb-thrust engines were assumed to be used, singly or in multiples, to keep engine development costs to a minimum.

Propellant mass fraction is the other prime variable in stage performance calculations. Propellant mass fraction is a measure of stage efficiency; it is defined as $M_p/(M_p + M_i)$ where M_i is the mass of inert accompanying the mass M_p of the propellant. The mass fraction results from the computer-generated designs are given in Fig. 2. (These are nominal or baseline design values, the influence of deviations from I_{sp} (Fig. 1) and mass fraction values will be shown later for one class of problem.) The mass fraction curves as a function of

*Space basing largely removes any geometric constraints.

stage weight are shown for expendable and reusable single-stage systems utilizing either fluorine/hydrogen, advanced oxygen/hydrogen or conventional oxygen/hydrogen propulsion systems. A thrust-to-weight ratio of approximately 0.3, found to be near optimum in preliminary studies, is used as a guide to select the nearest integral number of 15,000-lb-thrust engines. The curves have been smoothed through the actual minor step changes resulting from changes in numbers of engines.

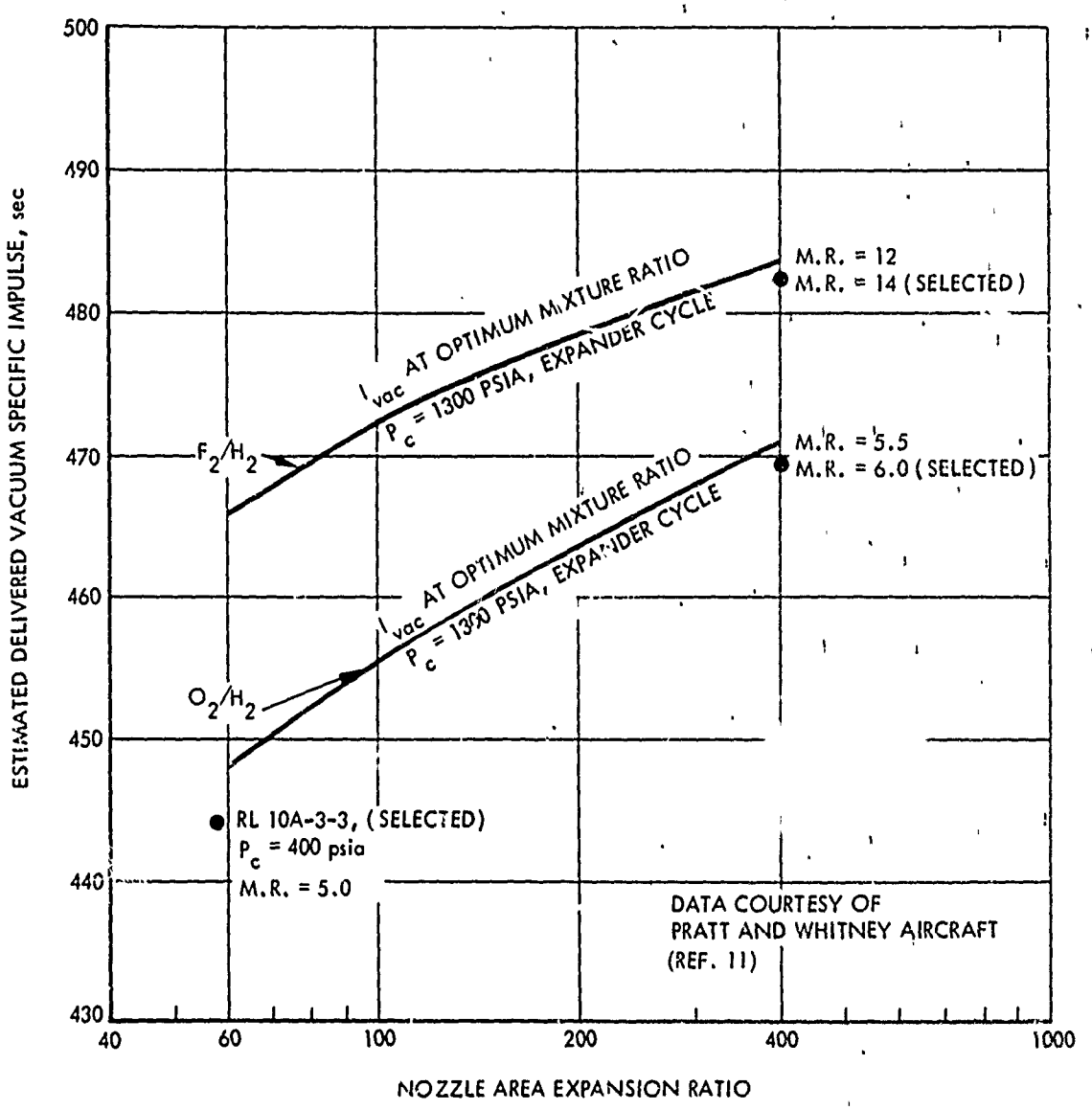

FIGURE 1. Estimated Performance of 15-klb Thrust Propulsion Systems Versus Nozzle Area Ratio

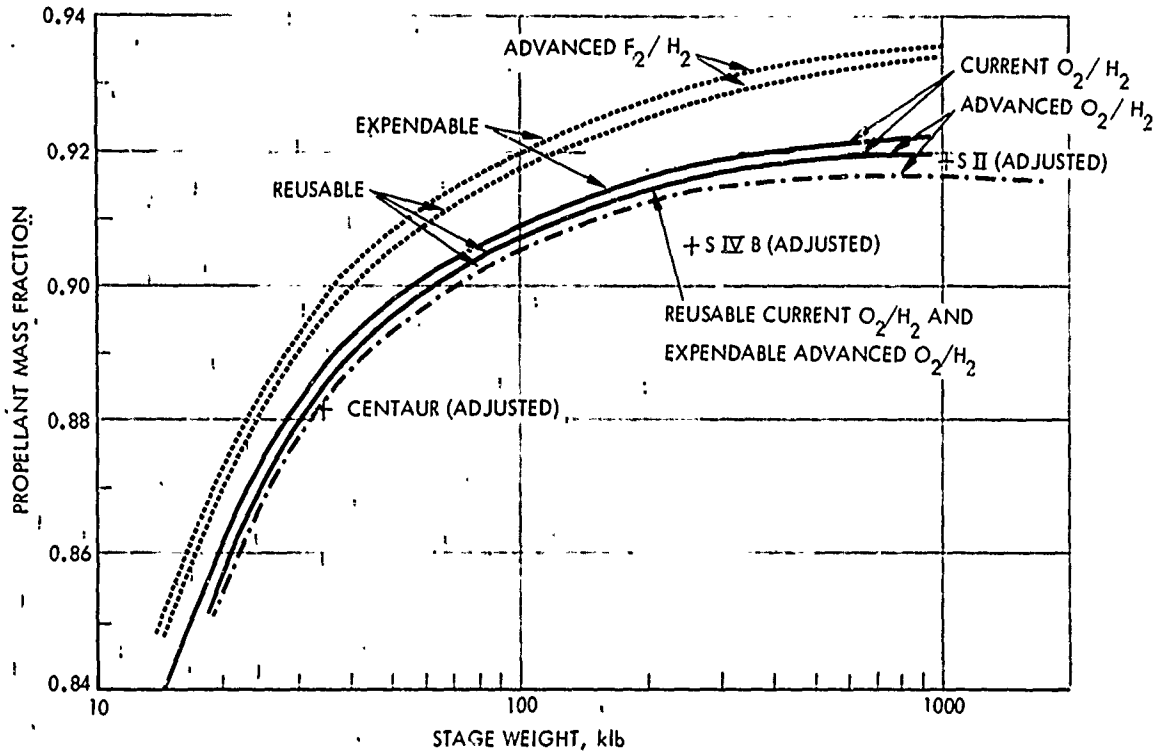

FIGURE 2. Propellant Mass Fraction as a Function of Propulsion Type and Stage Weight

The fluorine stages have a higher mass fraction than the oxygen stages because of greater average propellant density. The reusable stages have a slightly lower mass fraction than the expendable stages because of the requirement for additional insulation and meteoroid protection. Three points are given for "existing" stages; these stages however have been "adjusted" in propulsion weight to a uniform thrust-to-weight of 0.3 to be consistent with the basis used in the curves; they have not been adjusted in terms of structural load requirements, which would be lower (resulting in higher mass fraction) for stages that do not have to pass through the atmosphere.

Table 1 provides a summary of selected values used in the study. The orbital propellant facility is assumed to be in a harmonic orbit at 263-nmi altitude and 31.5-deg inclination. The EOS delivery costs

are parameterized from \$50 to \$1000/lb. The different 15,000-lb-thrust engines for the orbit-to-orbit shuttle are assumed to deliver either 445 sec, 469.5 or 482.5 sec, as plotted earlier (Fig. 1). Some sample cost estimates are shown for the engines and for the two types of stages of interest.* A nominal research and development (R&D) cost of \$50 million is assumed for the current O_2/H_2 engine to give it reuse capability; the new O_2/H_2 engine is estimated to required \$215 million and the F_2/H_2 engine \$270 million for development. The first unit cost of each engine is assumed to be \$280,000. The expendable and reusable single-stage vehicles whose costs are listed are capable of delivering 10,000 lb to 13,450 ft/sec impulsive velocity equivalent (one way to lunar orbit). The reusable vehicle is about twice as large and costs about 10% more to develop and about 20% more to buy. A learning curve slope of 90% is assumed throughout.

*As is discussed in the Appendix (q.v.), hardware (tankage) and engine cost figures used herein are lower than current experience might suggest. Improved tankage fabrication techniques are now known which make the projected costs for tankage appear readily achievable. Engine cost reductions to the levels shown may be more difficult to achieve, but a doubling of engine costs would affect vehicle costs by only about 10%. Some effects of doubling of all hardware costs are shown in Section V.

TABLE 1. SELECTED VALUES

ORBITAL PROPELLANT FACILITY ORBIT

263 nm x 31.5°

EARTH-TO-ORBIT SHUTTLE (EOS) DELIVERY COSTS

$50 TO $1000 PER LB

STANDARD ENGINES:

THRUST: 15,000 LBF

LIFE (MAXIMUM) 10 HR

TYPE	PERFORMANCE	WT, LB	MIXTURE RATIO	FIRST UNIT COST, $M	R&D COST, $M
CURRENT O_2/H_2	445 sec	300	5:1	0.28	50
ADVANCED O_2/H_2	469.5 sec	350	6:1	0.28	215
ADVANCED F_2/H_2	482.5 sec	350	14:1	0.28	270

STAGE THRUST-TO-WEIGHT ≈ 0.3 (NORMALLY)

TYPICAL VEHICLE DATA (10,000 LB PAYLOAD, 13,450 FT/SEC, ADVANCED O_2/H_2)

VEHICLE	GROSS WT, LB	R&D COST, $M	FIRST UNIT COST, $M
EXPENDABLE SINGLE STAGE	22,221	330	2.91
REUSABLE SINGLE STAGE	45,106	365	3.53

LEARNING CURVE SLOPE: 90%

V. MISSION STUDY PROCEDURE AND RESULTS

A. MISSION STUDY PROCEDURE--BASELINE DESIGN CONDITIONS

The procedure utilized in the following is one of selecting certain baseline conditions that seem reasonable (or at least plausible) and then, by use of the computer program, studying the effects of changes along the various dimensions. These baseline conditions selected for the two principal cases are summarized in Table 2. The geostationary mission is a special case of the cargo-carrying vehicles, as no payload is returned.

TABLE 2. BASELINE DESIGN CONDITIONS

Common

- Advanced O_2/H_2 propulsion
- \$100/lb EOS costs
- 10-hr first-stage engine life, if reused (typically 20-25 flights, with 50-flight maximum, varying with ΔV and thrust-to-weight ratio)
- 20-flight life for reusable second stages (thrust-to-weight quantized near 0.3)

Unmanned Cargo Vehicles

- 10,000-lb payload, no payload returned
- 100 flights in total program
- Single engine burn (no boil-off losses)

Manned Lunar Exploration System

- 10,000 lb returned each trip from lunar surface
- Variable outbound payload; payloads fully divisible
- Separate vehicles for lunar lander and orbit-to-orbit transfer
- Lunar lander uses same propulsion technology as orbital transfer vehicle
- Boil-off losses included for the transfer vehicle
- 20 reuses for lunar lander

B. CARGO-CARRYING VEHICLES

The first mission studied was the cargo-carrying mission with no payload return requirement, considering ΔV requirement to be an arbitrary parameter and considering the effects of configuration choice and EOS costs, etc. Each ΔV assumes a new vehicle design, i.e., these calculations are for "rubber" vehicles.

Figure 3 provides an overview of the operating cost results of this portion of the study; additional plots will follow to show further details of the relationships. In Fig. 3 the operating costs are shown for various configurations in those specific regions where the particular configuration gives minimum operating cost. Almost certinaly, the most important point to be noted from Fig. 3 is that, over a very wide range, the operating costs of the preferred configurations are almost proportional to EOS costs. Thus, reductions in cost to low earth orbit will proportionately decrease transportation costs for higher energy space missions. This point is discussed furthere in Section VI.

Up to approximately escape velocity, the single-stage reusable vehicle (R1) provides the lowest costs. Depending on EOS costs, a two-stage reusable vehicle (R1R2) or a single-stage expendable vehicle (E1) provides somewhat reduced costs in the ΔV region near escape velocity (see the dotted 10% penalty lines). The single-stage expendable vehicle appears as the lowest-cost configuration in only a small regime; however, it is competitive within 10% over a wide range of conditions with the next preferred configuration--a reusable-first, expendable-second-stage vehicle.

The slight operating cost advantage of the preferred configuration is evident from the large area enclosed by the 10-percent-difference dotted lines; it is also evident in the almost indiscernible change in slope across the boundaries where the preferred configuration changes. In these overlapping regimes, quite obviously, the choice of configuration will not be made purely on an operating cost basis. Total program costs will also enter and some discussion of this relationship will follow; one might expect, however, a predilection toward single stages for reliability reasons if the penalty is not too great.

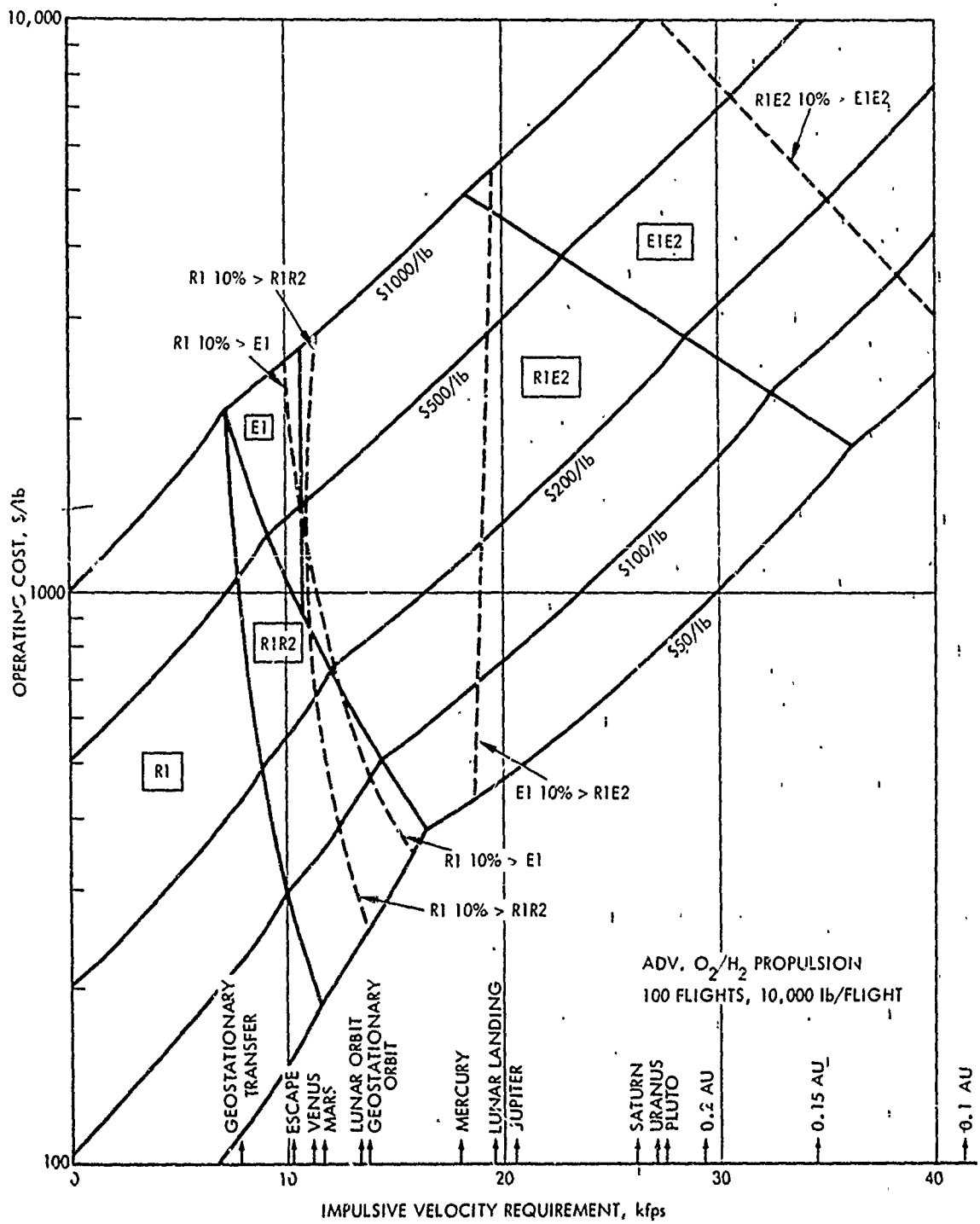

FIGURE 3. Operating Costs of Configurations Giving Minimum Operating Costs. ΔV's Shown for Interplanetary Missions are Hohmann Transfer Values to Mean Orbital Distance. The Total ΔV's of Several Missions Requiring More than one Burn are Shown for Convenience.

In examining Fig. 3, note that the left-hand intercept is approximately* at the EOS delivery cost; at this zero ΔV value, the entire cost for reusable systems is essentially that of carrying the payload to earth orbit. As the ΔV increases, more and more propellant and hardware must be carried to orbit for each pound of payload, increasing the cost. Eventually the round-trip ΔV requirement becomes large enough to approach the limits of single-stage capability, and at about 10,000 ft/sec (with the assumed O_2/H_2 propulsion system) either expendable single-stage or two-stage reusable vehicles become attractive; with still higher ΔV's, the two-stage configurations with the upper stage or both stages expendable are preferred.

In preparing Fig. 3 it was necessary to find the proper staging velocity for the two-stage configurations. It was noted in Section III that R&D costs equal to those of the more expensive stage were assumed if the propellant weights in the two stages were equal. In most cases use of this criterion led to minimum total program costs; in addition, as operating costs are not sensitive to staging velocity, it was found that this criterion also led to near-minimum operating costs. However, at very high ΔV requirements, the penalties associated with equal staging become severe so that it was necessary to determine an appropriate cost-minimized staging velocity (which in principle will be different when minimizing total program costs than when minimizing operating costs and will vary with EOS costs). In general, it was found that equal weight stages were fully satisfactory for the E1E2 configuration up to 15,000 fps but penalties were hardly appreciable (3%) even at 28,000 fps; with R1E2, penalties did not become appreciable until higher velocities (no penalty at 28,223 fps; 6.5% at 34,455 fps). For R1R2 configurations, operating costs were near minimum (2%) for equal weight stages up to about 15,000 fps; total program costs were still a minimum (at $100/lb EOS costs) at 17,818 fps.

*The intercept is not precisely at the EOS value with procedures used here for it is assumed that one engine and guidance system are needed even for infinitesimal ΔV requirements; with reusable systems, the cost is negligibly above the EOS value, but with expendable systems the intercept is considerably above the EOS value.

The relationships between the various configurations will become clearer on examining Figs. 4-6. In these figures gross weight (less payload) in low earth orbit (Fig. 4), operating costs (Fig. 5), and total program costs (Fig. 6) are shown for the various configurations. Several configurations are included on these plots that did not show up as minimum operating cost systems and hence did not appear on Fig. 3. These include the expendable-first/reusable-second (E1R2) and the reusable stage with two drop tanks (R1 + 2DT). The E1R2 configuration appears in two modes--an equal propellant weight mode, and a mode where the second stage imparts no velocity to the payload, but does return the guidance equipment. The R1 + 2DT configuration assumes tank droppage at the destination velocity.

In Fig. 4, the initial vehicle-associated mass in earth orbit is shown for the various configurations. Weights are given over a ΔV regime starting roughly at escape velocity. Again each configuration accelerates 10,000 lb to the added velocity plotted; with the exception of the two E1R2 classes, the plot is for minimum operating cost stages or, for most cases of interest, for equal-propellant-weight stages. From this figure it can be seen that the vehicles increase in weight with velocity requirement and generally decrease in gross weight with increased staging and expendability, as would be expected. Also, as expected, the single-stage reusable vehicle increases most rapidly in size with velocity requirement.

Expected operating costs of the different configurations at a single EOS cost ($100/lb) are shown in Fig. 5. The regime below escape is not shown as it is dominated by the single-stage reusable vehicle. Note now that those vehicles which were among the heaviest in Fig. 4 tend to have the lowest operating costs, obviously because the hardware is being reused, until the stages get very large. Above about 14,250 ft/sec (up to 32,500 fps) the R1E2 configuration vehicle has the lowest operating costs as was shown in Fig. 3; however, in the regime shown in Fig. 5, it is clear that little is gained by the more complex R1E2 over the simpler single-stage expendable.

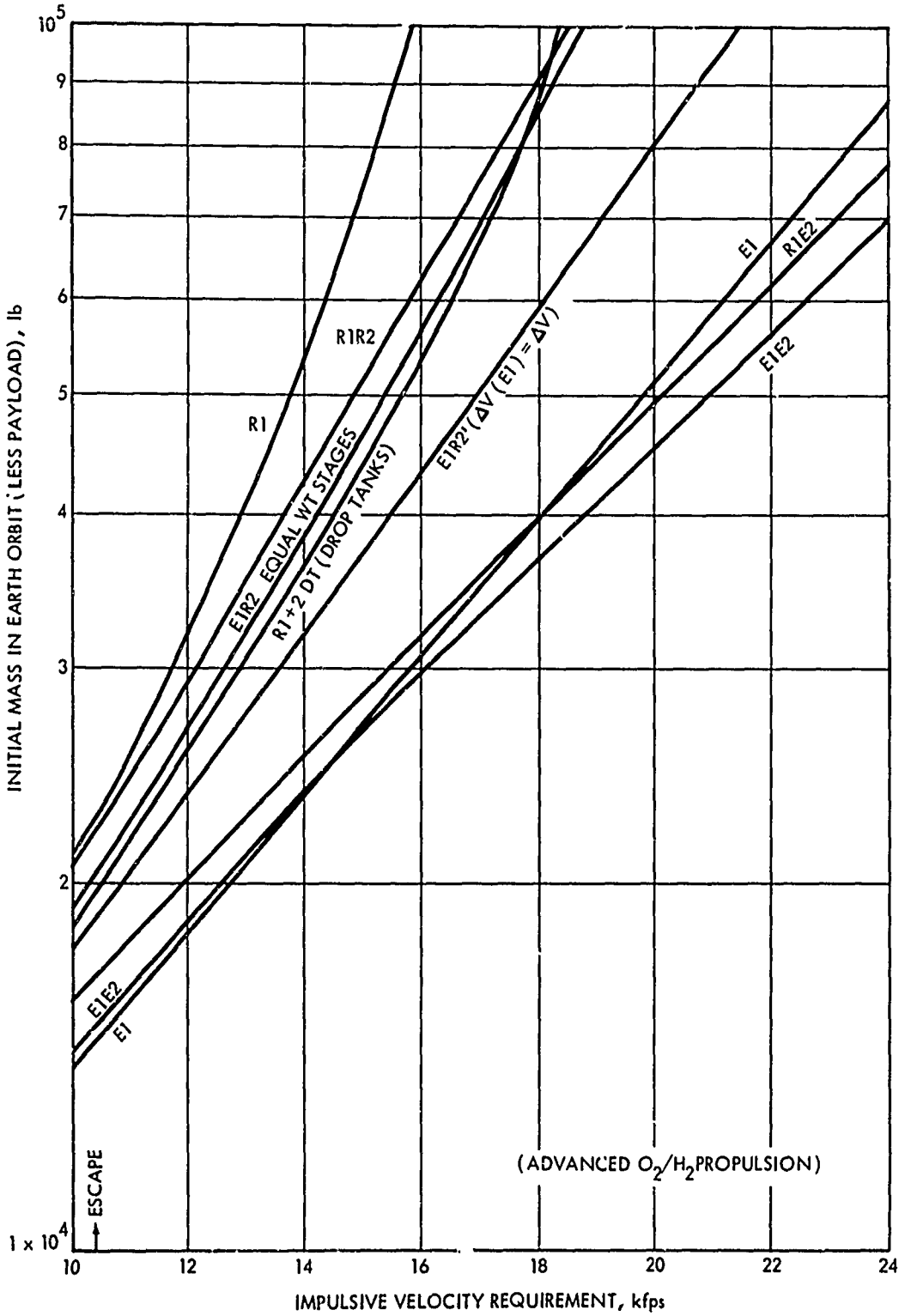

FIGURE 4. Initial Mass in Low Earth Orbit (Less Payload) for 10,000 lb Payload Accelerated

Note further that the gross-weight-to-payload-weight ratios for the reusable systems (derived from Fig. 4) are very nearly equal to the ratio of total operating cost to EOS cost (computed from Fig. 5). For example, for R1 at 14,000 fps, the ratio of total initial mass in orbit to payload is (54,000 + 10,000)/10,000 or 6.4; the operating cost ratio is 580/100 or 5.8. The two-stage reusable vehicle has a total/payload weight ratio of 6.1 at 15,000 fps (Fig. 4); the cost ratio is 5.8 (Fig. 5). Operating costs for resuable systems are nearly proportional to EOS costs and thus insensitive to uncertainties in hardware cost correlations, although they would be sensitive to weight uncertainties. For expendable systems the ratios derived from Figs. 4 and 5 are, of course, not equal: for E1 at 14,000 fps, the weight ratio is 3.4 but the cost ratio is 5.4. The operating costs of expendable systems are sensitive to both hardware costs and weight uncertainties. (These factors will be illustrated later in the section for single-stage systems.)

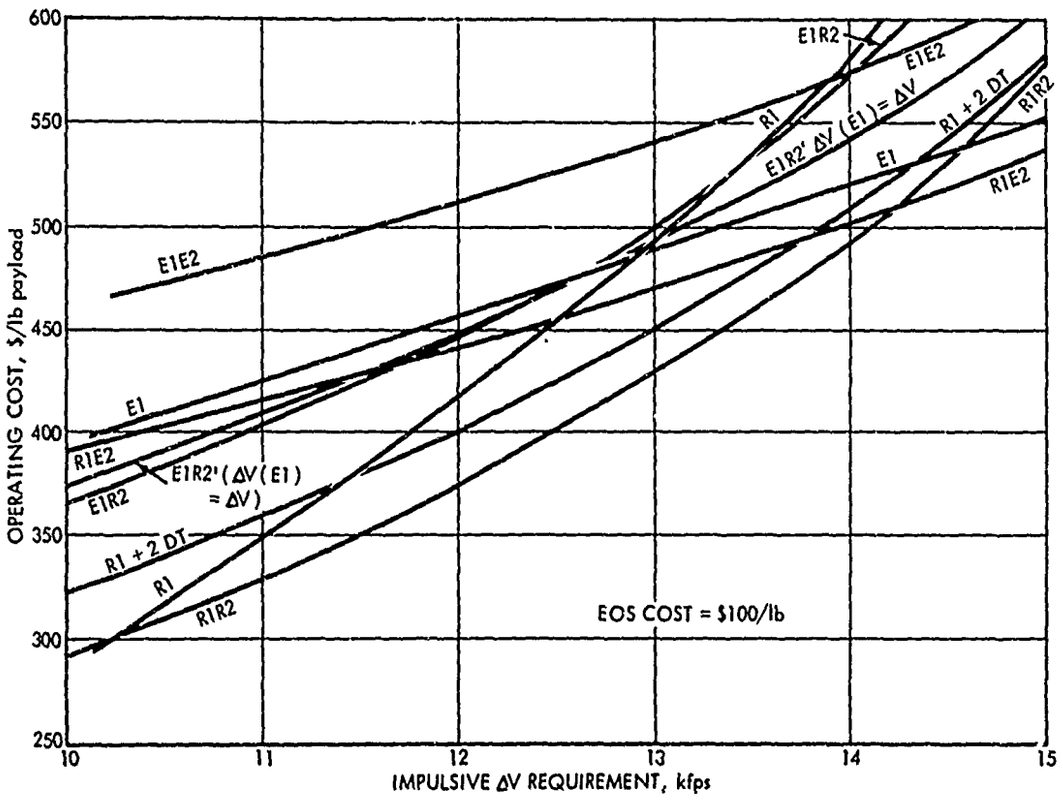

FIGURE 5. Operating Costs of Various Accelerator Stages Delivering 10,000 lb to ΔV From Low Earth Orbit (100 Flights)

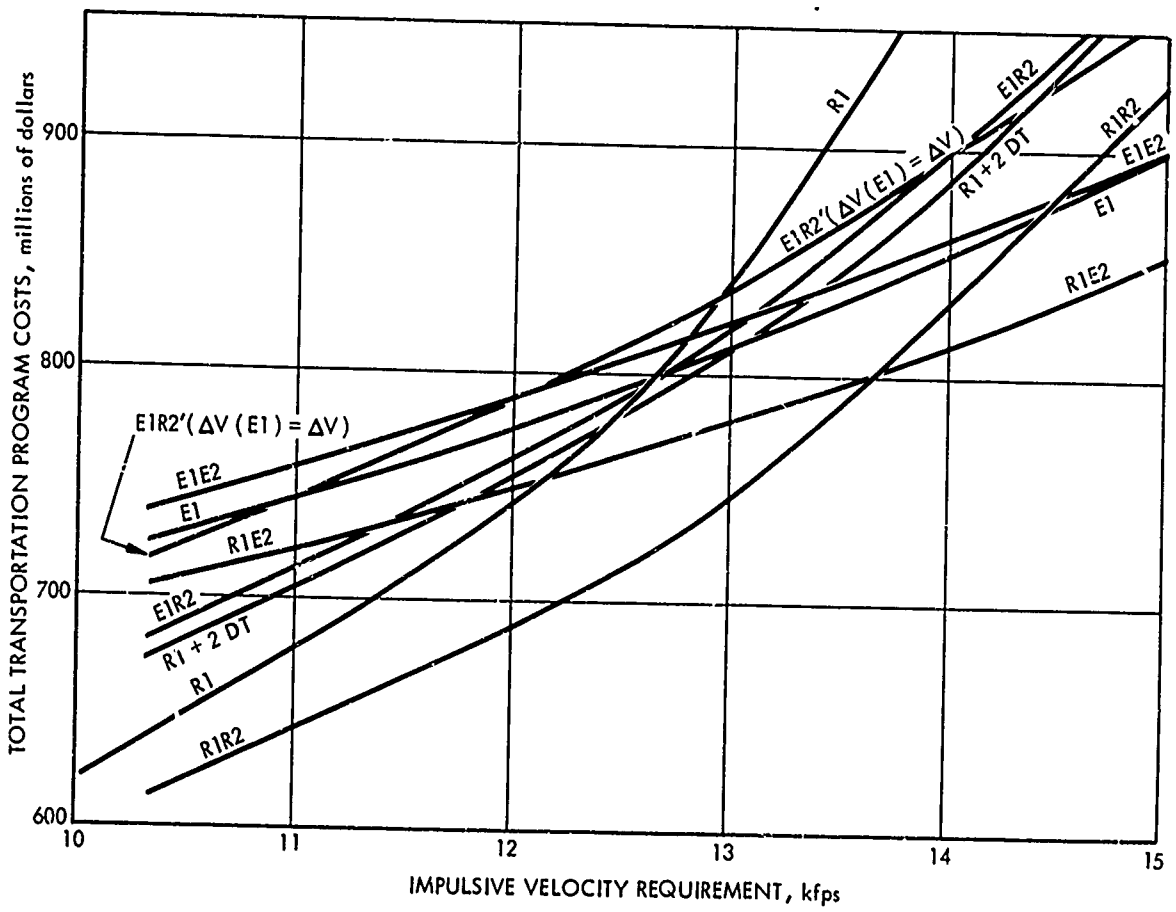

FIGURE 6. Total Program Costs for Various Accelerator Stages Delivering 10,000 lb (100 Flights, Advanced O_2/H_2 Propulsion, \$100/lb EOS)

In addition to operating costs, total transportation program costs are of interest as a function of velocity increment. Total program costs of course vary with size of programs*; for illustrative purposes, the baseline \$100 EOS cost, 100-flight program (each flight delivering 10,000 lb to the indicated ΔV) is assumed and total costs are plotted in Fig. 6, again emphasizing the rather narrow regions above escape wherein crossovers of most interest occur. Note

*Where expendable vehicles are permissible, the greater development costs of reusable vehicles must be recovered by savings in operating costs. This factor will favor expendable vehicles for small programs.

that the two-stage reusable vehicle has lower total program costs than does the single-stage reusable even near escape where operating costs are equal (Fig. 5) due to the reduced stage size and hence reduced R&D cost with a two-stage (equal-propellant) configuration. The same effect enters in comparing R1E2 and E1. The two-stage expendable vehicles cost more than the single-stage expendables however, apparently because of the now much poorer mass fractions at these small stage sizes and the associated inefficiencies involved with use of the fixed-thrust engines that have been assumed.

Figures 5 and 6 were plotted from calculations assuming $100/lb EOS costs. At higher EOS costs the relative positions would change somewhat, as can be inferred from Fig. 3; at higher EOS costs reusability becomes less attractive; the cost of transporting to orbit the greater propellant requirement of the reusable stage tends to outweigh the hardware cost savings obtained through reusability. This point was further investigated making use of the assumption (as already noted) that in spite of the apparent cost savings that might be utilized in some regimes through use of two-stage vehicles, a bias toward the single-stage options would be expected from a reliability standpoint. Consequently, the specific tradeoffs of reusability versus expendability for single-stage vehicles were computed, with results shown in Fig. 7. In Fig. 7 the boundaries separating the domains in which expendability or reusability gives the lower operating costs and total program costs are shown as a function of ΔV requirement; above the lines, expendable vehicles are preferred, and below the lines, reusable vehicles. The boundaries are, of course, lines of equal cost. Four lines are shown, one each for the operating costs and the total transportation program costs for vehicles delivering, respectively, 10,000 lb (100 flights) and 100,000 lb (10 flights) each trip for a total mass delivered of one million pounds in each case. Three observations are to be noted from this plot. First of all, as the velocity requirement increases, the EOS delivery cost must drop in order to justify reusable vehicles. This is consistent with the observation made earlier from Fig. 3. Second, a reusable vehicle can

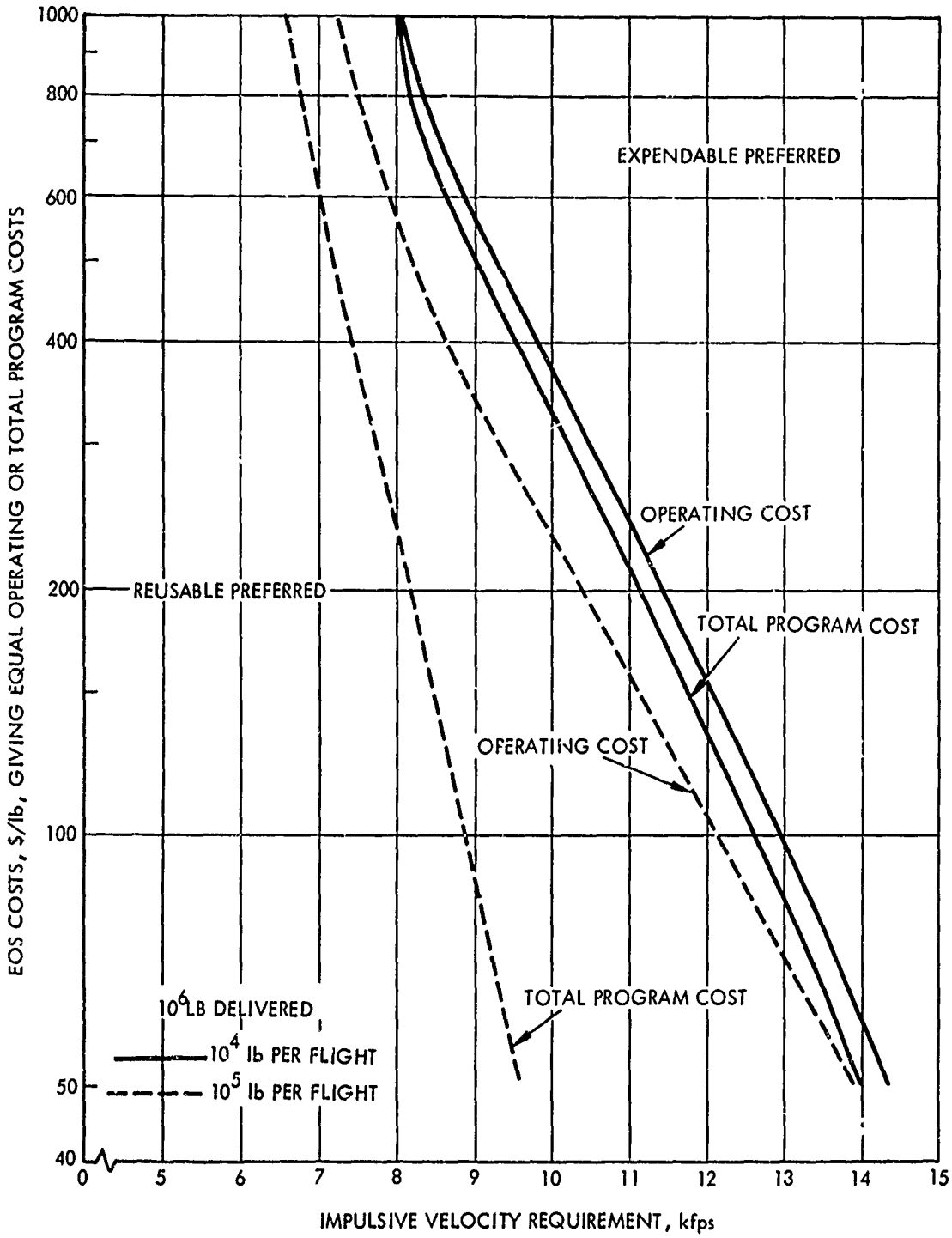

FIGURE 7. EOS Delivery Costs Giving Equal Operating and Total Transportation Program Costs for Single Stage Expendable and Single Stage Reusable Vehicles

be justified to a higher value of EOS costs on the basis of operating costs than on the basis of total transportation program costs. Finally, as the payload size increases, at constant total mass delivered, the advantage of reusability decreases.

The above effects (Fig. 7) can be explained qualitatively. Reusable stages, being considerably larger than expendable stages and using more propellant, involve greater delivery costs to low earth orbit even though they yield lower per-flight hardware costs because of reuse; therefore an upper limit is set for the EOS costs for break-even by the difference in per-flight hardware costs. As the velocity requirement increases, the weight difference grows faster than the hardware cost difference and the break-even EOS cost must diminish as shown in the figure. At the larger 100,000-lb payload size, the reusable vehicle enjoys a smaller improvement in mass fraction over the expendable vehicle (Fig. 2) than it does at the smaller 10,000-lb payload size, and therefore has a proportionately higher difference in weight and, hence, lower break-even EOS costs. When the difference in nonrecurring costs is included, to give the comparative value of total transportation program costs, an additional penalty is levied against the reusable vehicle and an even lower break-even EOS cost is required.

Figure 8 provides some information on the sensitivity of results such as those of Fig. 7 to the cost correlations and the values of I_{sp} and mass fraction used. Four curves are shown, each defining the boundary between reusability-preferred and expendability-preferred domains under different conditions for single-stage vehicles. The second curve from the right is the baseline curve from Fig. 7. The farthest right curve shows the effects of doubling the stage hardware cost over the values derived from the correlations; the break-even EOS cost is essentially doubled. As would be expected, increased hardware costs make reusable vehicles more attractive. The effects of a 10-sec decrease in I_{sp} for both stages, shown on the line just to the left of the baseline curve, are seen to be insignificant. The

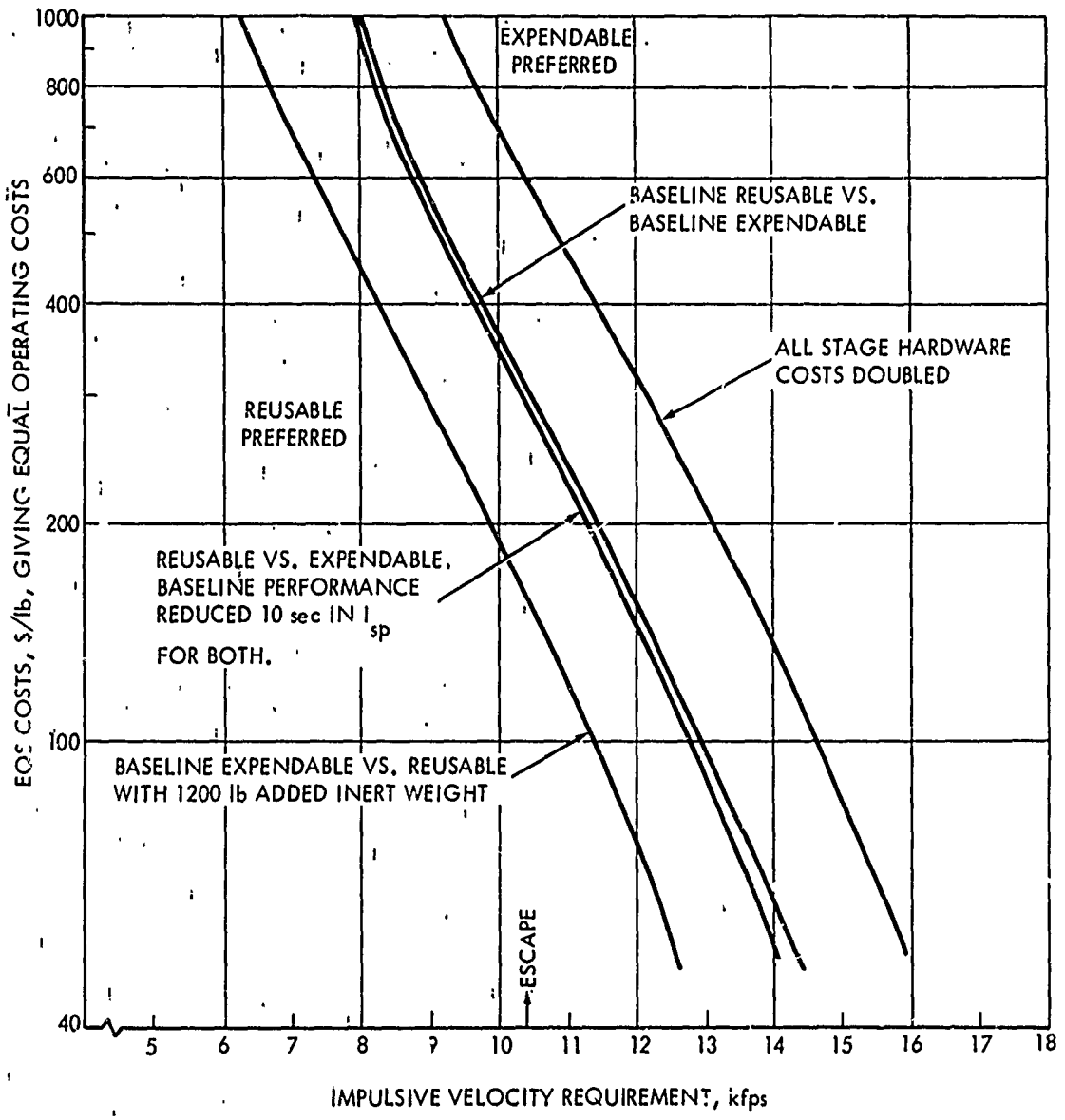

FIGURE 8. Effect on Expendable/Reusable (Single-Stage) Tradeoff of Cost, I_{sp}, and Astrionics Weight Changes ; 10^4 lb/trip; Advanced O_2/H_2 Propulsion

farthest left curve shows effects of an increase in 1200 lb of inert* (costed as guidance equipment) for the reusable stage without corresponding change in the expendable stage. A significant effect can be noted, greatly reducing (by about a factor of two) the EOS delivery cost below which the reusable vehicle becomes attractive.

The general question of how the various configurations vary in size and cost with payload is also of interest but one that is not discussed in depth here. However, gross weight data, using baseline conditions, were computed for the different configurations as a function of payload size for a particular case in which all vehicles were capable of 13,450 ft/sec equivalent impulsive velocity. Results, normalized on a payload ratio basis, are given in Fig. 9. Equal weight stages were utilized for two-stage configurations. The growth factor is defined as the ratio of the growth in vehicle (less payload) weight to the growth in payload from a baseline 10,000 lb. The two-stage reusable vehicle is seen to involve the smallest normalized growth factor, i.e., the smallest growth in vehicle size for an increase in payload. This plot indicates that for a 10-fold increase in payload in the two-stage reusable vehicle there is only a 6.5-fold increase in vehicle weight. For a single-stage expendable vehicle, however, the 10-fold increase in payload requires an 8.7-fold increase in vehicle weight. The ratio changes little beyond the 100,000-lb-paylod level except for the single-stage reusable vehicle, the results for which may or may not be realistic as they are beyond the

*Reusable systems may indeed involve such penalties if complete redundancy and autonomous navigation, etc. are utilized. NASA (Ref. 12) has indicated that the Fig. 2 mass fraction values for vehicles below the 100,000-lb class are 0.03 and 0.05 higher (consistent with a 1200-lb weight difference) than their studies are estimating. Part of the difference may be due to difference in ground rules, i.e., in unmanned space-basing (unconstrained in size) versus manned vehicles with alternative ground- or space-basing constrained to specific EOS cargo bays. Note (Fig. 8), however, that possible optimism in the hardware weight of reusable vehicles tends to balance possible optimism in the costs of expendable vehicles.

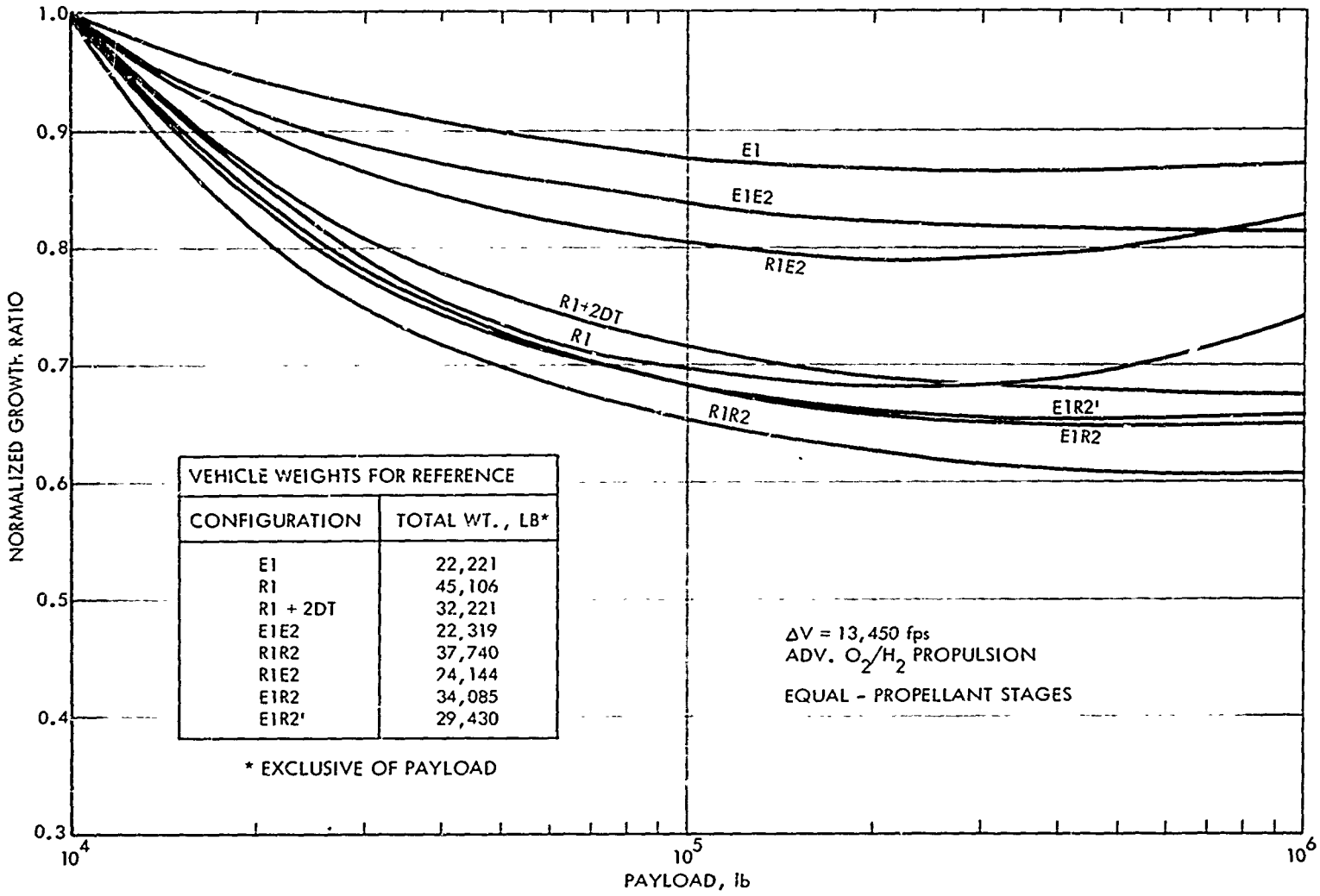

FIGURE 9. Normalized Vehicle Growth Factors Weight Ratio (Without Payload) of Vehicles Carrying Indicated Payload to Vehicles Carrying 10,000 lb of Payload

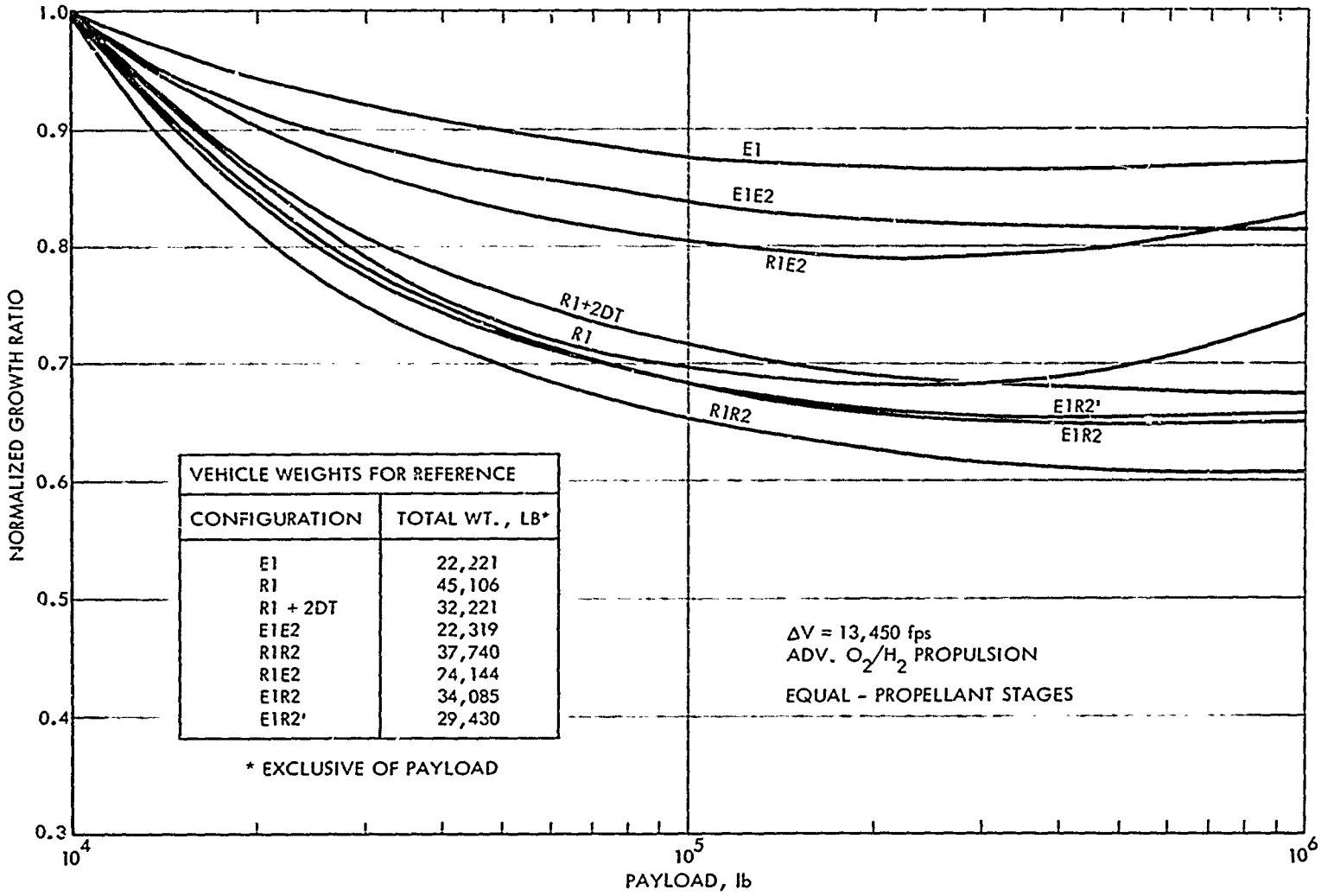

FIGURE 9. Normalized Vehicle Growth Factors Weight Ratio (Without Payload) of Vehicles Carrying Indicated Payload to Vehicles Carrying 10,000 lb of Payload

region on which the correlations were based. (According to the correlations used, at tank sizes approaching those of the single-stage million-lb-payload reusable vehicle, better mass fractions are achieved with dual or multiple tanks than with single tanks.)

The question of the desirability of advanced propulsion was also investigated for the single-stage configuration; it will also be examined for the other missions in later sections. As noted earlier, it is assumed that 15-klb-thrust current oxygen/hydrogen propulsion systems are, or can be, available, either with no development costs if expendable, or with $50 million in development costs if reusable. Also, advanced oxygen/hydrogen engines of 15-klb thrust are estimated to require $215 million, and advanced fluorine/hydrogen $270 million for development, whether reusable or expendable. On an operating cost basis, it is clear that the higher performing systems will always involve lower operating costs, even with fluorine, since an EOS cost of $100/lb will completely mask the cost of propellant, and higher performance implies less propellant required. The advantages of advanced propulsion systems will increase with stage velocity requirements, and as a consequence will be greater for reusable than for expendable systems. If the program is large enough, i.e., if enough flights are made, the savings in operating costs will eventually pay off the extra development cost. This point will be illustrated in the next section for the lunar mission.

A crossover of advanced versus current propulsion system costs will also occur as a function of velocity requirement at a given program size and EOS cost. This propulsion system crossover is shown for reusable single-stage vehicles and for expendable single-stage vehicles in Fig. 10 where, again using baseline values, total transportation costs for a 100-flight program with 10,000-lb payload each flight are plotted for current and advanced O_2/H_2 and advanced F_2/H_2 systems as functions of velocity increment. It is clear from this plot that a crossover between current and advanced systems occurs at this program size and EOS costs for the reusable vehicles at about 14,000 ft/sec (or lunar-orbit requirement) but that very high velocity increments (greater than 21,000

ft/sec) are required before advanced propulsion would involve lower total costs for this hypothetical million pound delivery program with expendable vehicles. At higher EOS costs, the advantages of advanced propulsion would, of course, be increased.

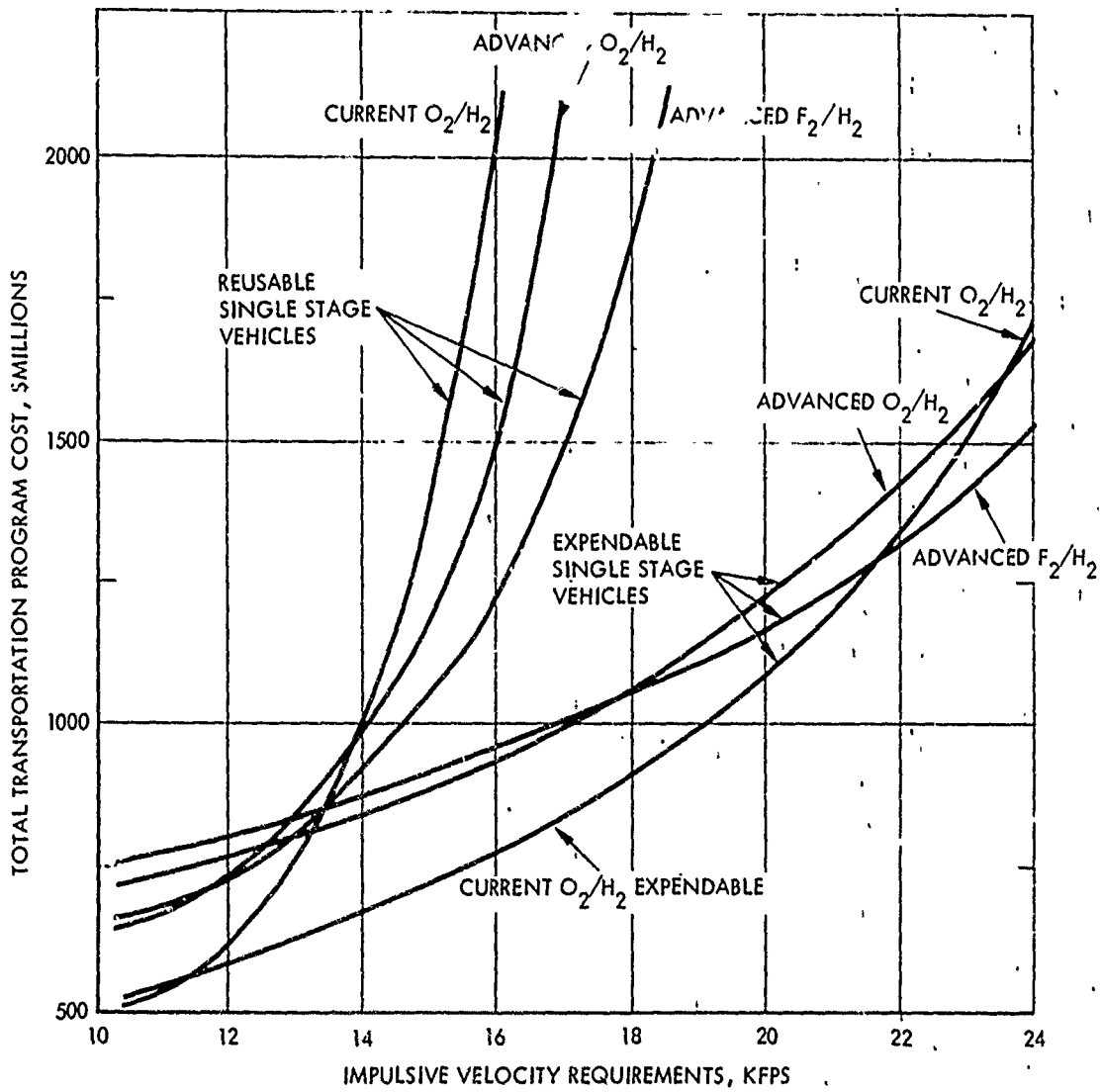

FIGURE 10. Effects of Propulsion Technology on Total Transportation Program Costs (Unmanned Accelerators; 100 Flights; 10,000-lb Payload Each Flight; $100/lb EOS Costs)

C. LUNAR AND GEOSTATIONARY MISSION STUDIES

1. Mission Requirements

Minimum lunar-orbit and geostationary-orbit energy requirements are quite similar, as shown in Table 3. Time requirements are, of course, different by about a factor of 12 for the two missions. In addition, phasing requirements enter into the lunar-orbit-rendezvous mission which affect either total energy or total time between trips. The velocity requirements for the lunar mission in Table 3 correspond to 3-day transfer time and an 18-day stay time in lunar orbit typical of the lunar-orbit rendezvous mode of lunar exploration.

2. Lunar Exploration

Turning first to studies of the manned lunar exploration mission, it was assumed that an extensive lunar exploration program is to be carried out but the total magnitude of this program is not known at the beginning of the effort. It was further assumed that 10,000 lb, possibly a manned capsule plus moon samples, etc., are to be returned each trip from the lunar surface back to earth orbit.

There are a number of modes by which lunar exploration can be accomplished, involving rendezvous and/or staging at various energy points in the earth-moon space, various expendable/reusable options, and undoubtedly various optima in stage commonality to minimize total R&D costs. A complete study of these options is beyond the scope of this work. However, two modes have been investigated in some detail. The first of these, lunar-orbit rendezvous, is similar to the mode used in current APOLLO flights, except that in this case the lunar lander is reusable and is refueled but not delivered each time by the vehicle which goes between earth and lunar orbits. The second mode, illustrated because it is a less familiar concept (Fig. 11), assumes that the translunar injection vehicle is put in a free-return (to low earth orbit) trajectory around the moon, again with a separate lunar lander. The lunar lander in the first case performs rendezvous with the earth-to-lunar orbit vehicle while in lunar orbit and the second with the vehicle during its descent from the earth-moon null point to

TABLE 3. LUNAR AND GEOSTATIONARY MISSION DATA

Initial EOS Parking Orbit	263 nmi x 31.5 deg
Geostationary Mission	
Characteristic velocity requirements (each way, from parking orbit)	
Geostationary Orbit Transfer (including optimum split of plane change)	7,888 fps (perigee)
Circularization	6,003 fps (apogee)
TOTAL	13,891 fps
Lunar Orbit Mission	
Lunar-Orbit Rendezvous	
Translunar injection	10,466 fps
Lunar orbit insertion	3,000 fps
TOTAL to lunar orbit	13,466 fps
Descent from lunar orbit to surface	6,000 fps
TOTAL to lunar surface	19,466 fps
Translunar (and transearth) coast time	3 days
Lunar orbit stay time	18 days
Free-Return Rendezvous	
Translunar injection	10,466 fps
Direct lunar descent	9,000 fps
TOTAL to lunar surface	19,466 fps
All Missions:	
Thrust parallel with velocity vector for g-loss estimates	
Velocity losses as given in the Appendix	

its lunar periapsis (perilune) in the free-return trajectory. Although the split is different, total velocity requirements for these two modes to the lunar surface are essentially the same and for purposes here have been assumed to be identical.

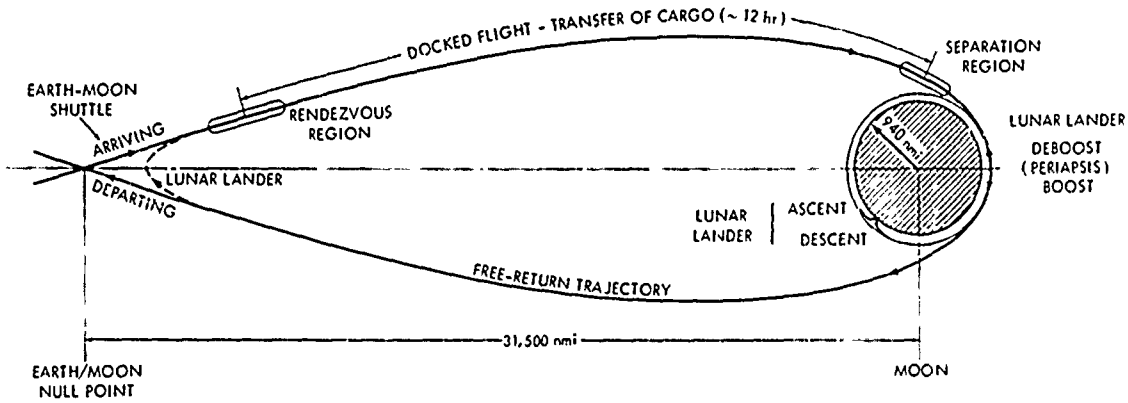

FIGURE 11. Free-Return Rendezvous Schematic (Not to Scale)

In addition to questions of rendezvous mode, the lunar exploration study involved questions of preferred configuration and optimum sizing, both of which would be expected to vary with EOS cost. The approach was to select two simple configurations (the single-stage and the stage-and-one-half) for the translunar injection vehicle and carry out a sizing study. Other configurations and effects of EOS costs and propulsion choice were then investigated at the vehicle size level selected. Preliminary calculations showed that of the two simple configurations studied, the single-stage vehicle was preferred for the free-return rendezvous mode and the stage-and-one-half for the lunar-orbit rendezvous mode.

Using this choice of configurations, total transportation program costs in the two modes were determined as a function of vehicle size for different traffic rates to the lunar surface. The lunar lander was sized to match the outbound payload-plus-propellant capacity of the translunar vehicle and the 10,000-lb return requirement. Results

are shown in the next two figures. In Fig. 12 total 10-year transportation costs versus outbound payload capacity per flight are shown at several different traffic levels: 30,000 lb/yr, 100,000 lb/yr, 300,000 lb/yr, and 1,000,000 lb/yr to the lunar surface. A minimum bound in vehicle size is set at which 10,000 lb are taken out and 10,000 lb returned each trip. The most interesting observation about Fig. 12 is that a minimum-cost regime exists at about 58,000 lb of payload to the lunar surface (= 100,000 lb to lunar orbit) with little dependence on program size. Figure 13, for the free-return rendezvous mode, shows virtually the same result. Careful intercomparison of Fig. 12 and Fig. 13 will show that the free-return mode is slightly (a few percent) less expensive than the lunar-orbit mode. Furthermore, the free-return rendezvous mode appears to have some attractive characteristics in terms of safety and ease of plane change. (An interesting third alternative exists, not pursued further here, in which the "translunar" vehicle and the lunar lander are the same size and the rendezvous orbit is a highly eccentric earth orbit rather than a lunar orbit.)

The plots in Figs. 12 and 13 also show the interesting result that an ambitious lunar program involving 300,000 lb/yr to the lunar surface for 10 years could be carried out utilizing either approach for only about $300,000,000 per year (about the cost of one APOLLO flight) in transportation costs, including amortizing the development costs over 10 years, if the EOS cost is $100 per pound. Essentially, it costs 10 times as much to put material on the moon, with this propulsion technology and at this payload return quantity, as it does to deliver material to earth orbit. Obviously if the EOS cost were higher, the total transportation cost would be higher (almost proportionately so).

Returning now to the configuration question, the cost penalties associated with use of the simpler one- and one-and-a-half-stage configurations instead of two-stage transfer vehicles were determined. The procedure used was to consider the various configurations on a cost basis, computing only the costs associated with the transfer

vehicle, but with all vehicles delivering the same payload to the same ΔV for comparison. The procedure was thus somewhat simplified over the procedures used in preparing Figs. 12 and 13, i.e., boil-off and certain other losses were assumed to be equal for the different configurations, so that results, which are presented in Table 4, are not precisely comparable with results in Figs. 12 and 13. The results in Table 4 should, however, be internally comparable.

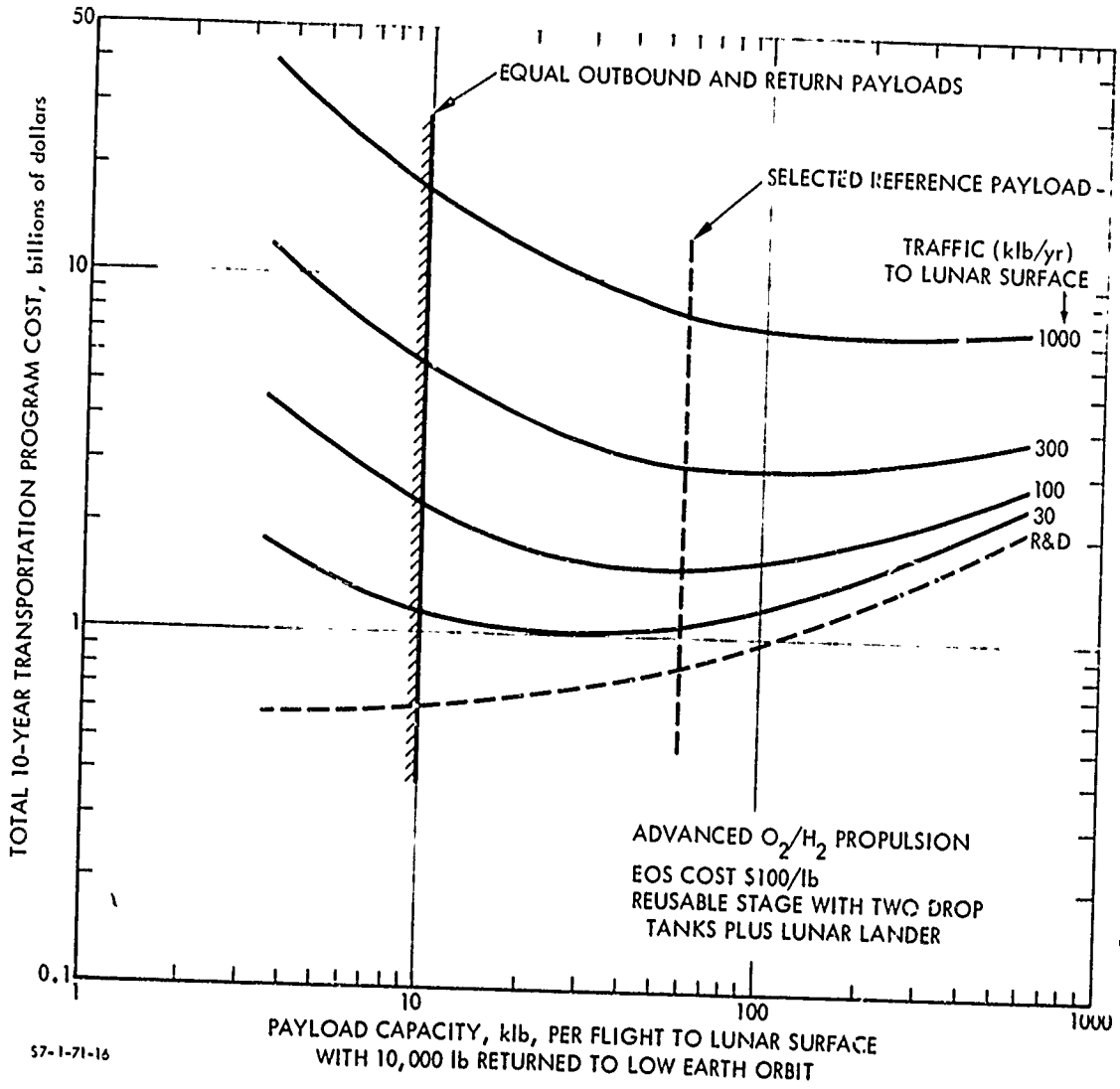

FIGURE 12. Total 10-Year Transportation Program Costs Using Lunar-Orbit Rendezvous

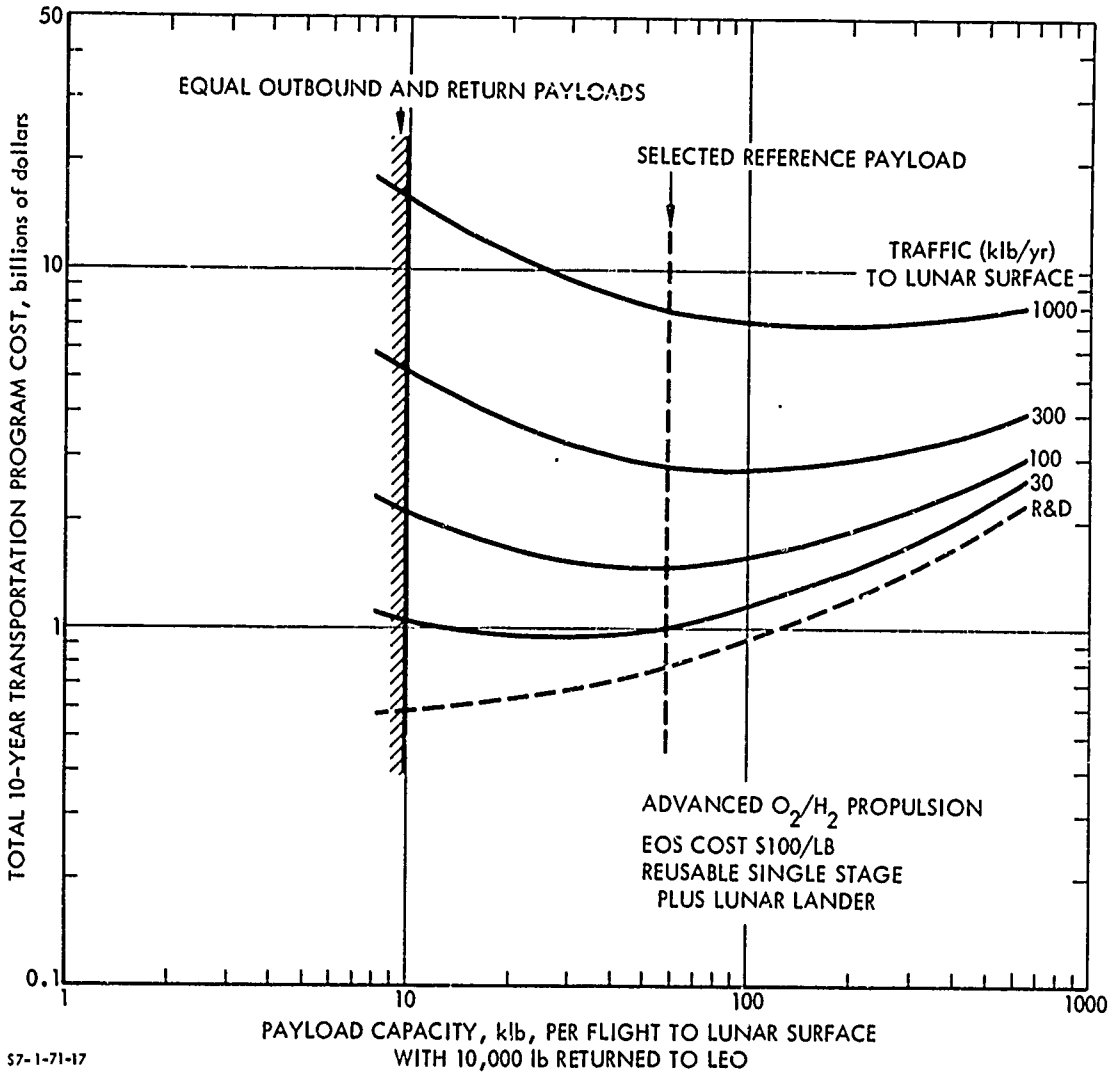

FIGURE 13. Total 10-Year Transportation Program Costs Using Free-Return Rendezvous

The data in Table 4 show total transportation program costs (excluding lunar lander costs) and per-mission costs for 18 flights, each delivering 58,000 lb to the lunar surface (approximately 1,000,000 lb total) and each returning 10,000 lb to low earth orbit. The lowest cost configuration at each EOS cost for each rendezvous mode is underlined. Note that, in general, the two-stage reusable vehicle (the two stages having the same propellant weight) provides lowest costs, but that these costs are, at $100/lb or more, within 12% of the transportation

program costs of the single stage in the free-return rendezvous mode and within 13% of the reusable vehicle with drop tanks in the lunar-orbit rendezvous mode, with per-mission costs diverging to a lesser degree. The differences decrease on a percentag[illegible]sis with higher EOS costs. Note that at very high EOS costs, the use of a large expendable stage and a small reusable stage purely for payload and guidance equipment return becomes attractive. Whether the cost savings of two-stage vehicles would justify their greater complexity would need to be determined by in-depth study.

TABLE 4. CONFIGURATION COST COMPARISONS FOR LUNAR EXPLORATION

LUNAR-ORBIT RENDEZVOUS[a]

EOS Cost	R1		R1-2DT		R1R2[b]		E1R2[b]		E1R2'[b]	
	TPC	OCPM	TPC	OCPM	TPC	OCPM	TPC	OCPM	TPC	OCPM
50	1092	24.13	1026	22.17	863	20.43	907	[illegible]	1037	[illegible]
100	1512	47.50	1373	41.45	1215	39.98	1248	43.36	1366	42.77
200	2354	94.24	2067	80.02	1918	79.07	1931	81.29	2023	[illegible]
500	4878	234.48	4150	195.73	4029	196.34	3979	[illegible]	[illegible]	188.90
1000	9085	468.19	7621	388.57	7547	[illegible]	7393	[illegible]	7284	371.6

FREE-RETURN RENDEZVOUS[d]

EOS Cost	R1		R1+2DT		R1R2		E1R2		E1R2'	
	TPC	OCPM	TPC	OCPM	TPC	OCPM	TPC	OCPM	TPC	OCPM
50	926	19.82	951	20.74	808	19.04	917	22.91	981	23.57
100	1272	39.04	1278	38.95	1136	37.26	1183	41.00	1302	41.40
200	1964	77.49	1934	75.36	1792	73.71	[illegible]	77.19	1944	77.07
500	4041	192.89	3901	184.62	3760	183.04	3784	[illegible]	3870	184.08
1000	7502	385.11	7178	366.72	7040	365.25	7047	366.70	7080	362.42

[a]$\Delta V \cong 14{,}070$ fps, including losses.

[b]The two stages have the same propellant weight, minimizing R&D cost, except for E1R2', where the second stage is used only for the return trip. Costs are based on 18 flights delivering 58,000 lb to lunar surface and returning 10,000 lb each trip, with aid of lunar lander. Costs associated with development and emplacement of the lunar lander are not included; these are larger for the free-return rendezvous than for the lunar-orbit rendezvous mode.

[c]Underlines denote lowest values.

[d]$\Delta V \cong 10{,}860$ fps, including losses.

The advanced propulsion question was also considered for the lunar mission case; i.e., the size of program needed to justify advanced propulsion development for this mission was of interest. In Fig. 14, total transportation program costs for vehicles utilizing the lunar-orbit rendezvous mode (which is more demanding from a propulsion standpoint), all delivering an optimal 58,000 lb to the lunar surface, are plotted versus the total payload delivered. In this case, advanced oxygen/hydrogen propulsion brings the total program cost lower than conventional oxygen/hydrogen after a total payload of about 1,000,000 lb has been delivered (approximately 18 trips) to the lunar surface. Fluorine/hydrogen crosses at a lower value, at about 500,000 lb (8-9 trips) delivered. This is because, according to the cost correlations used, the fluorine/hydrogen vehicles* have a lower R&D cost due to their smaller size at constant payload per trip than does the advanced oxygen/hydrogen system.

3. Geostationary Mission Studies

The geostationary mission studies involved determination of operating costs for different vehicle configurations, payload sizes, and program sizes; in addition, effects of off-loading and of advanced propulsion were studied. The rationale for considering only operating costs was that the geostationary mission would likely be only one of a number of missions which the space-based vehicle could carry out; hence total program costs could not be ascribed to this mission alone.

The first factor investigated was the effect of program size, assuming 10,000 lb per trip, one-way payloads, advanced O_2/H_2 propulsion, and $100/lb EOS costs. Results are shown in Fig. 15 for eight configurations. The greater effects of learning on the expendable vehicles as compared to reusable vehicles are to be noted. Note that over a wide range of traffic levels, the two-stage reusable vehicle provides the lowest operating costs; at very large traffic levels, the reusable first-expendable second stage provides lowest operating costs.

*The lunar lander in this case also uses F_2/H_2.

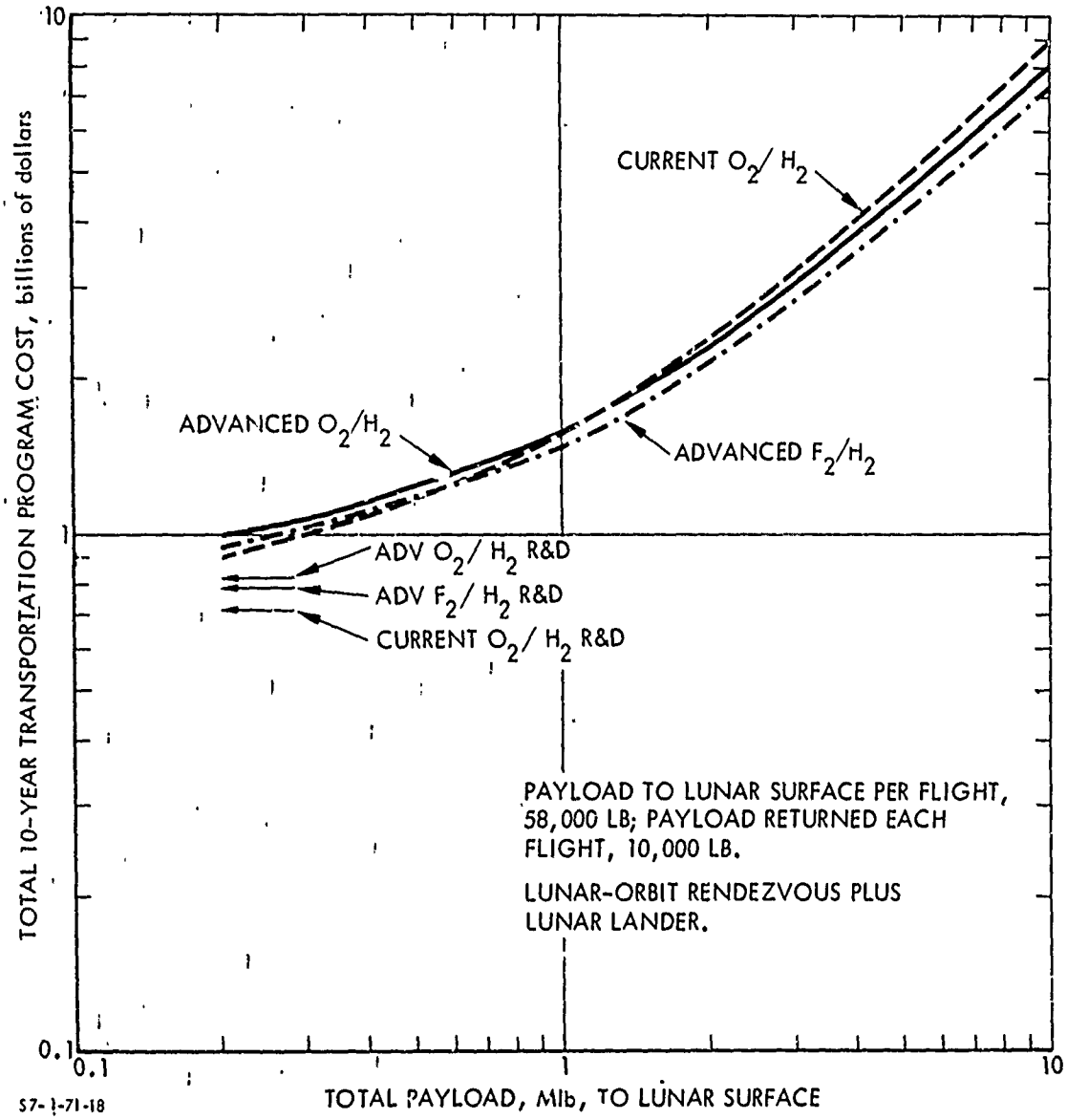

FIGURE 14. Total 10-Year Transportation Program Costs Versus Program Size for Different Propulsion Systems

Turning now to the program size used in earlier portions of this report (10^6 lb), it was of interest to determine operating costs as a function of payload size. In doing this only four configurations were studied, the simple stages R1 and E1, the drop tank reusable, and the lowest cost but more complex R1R2. Results are shown in Fig. 16.

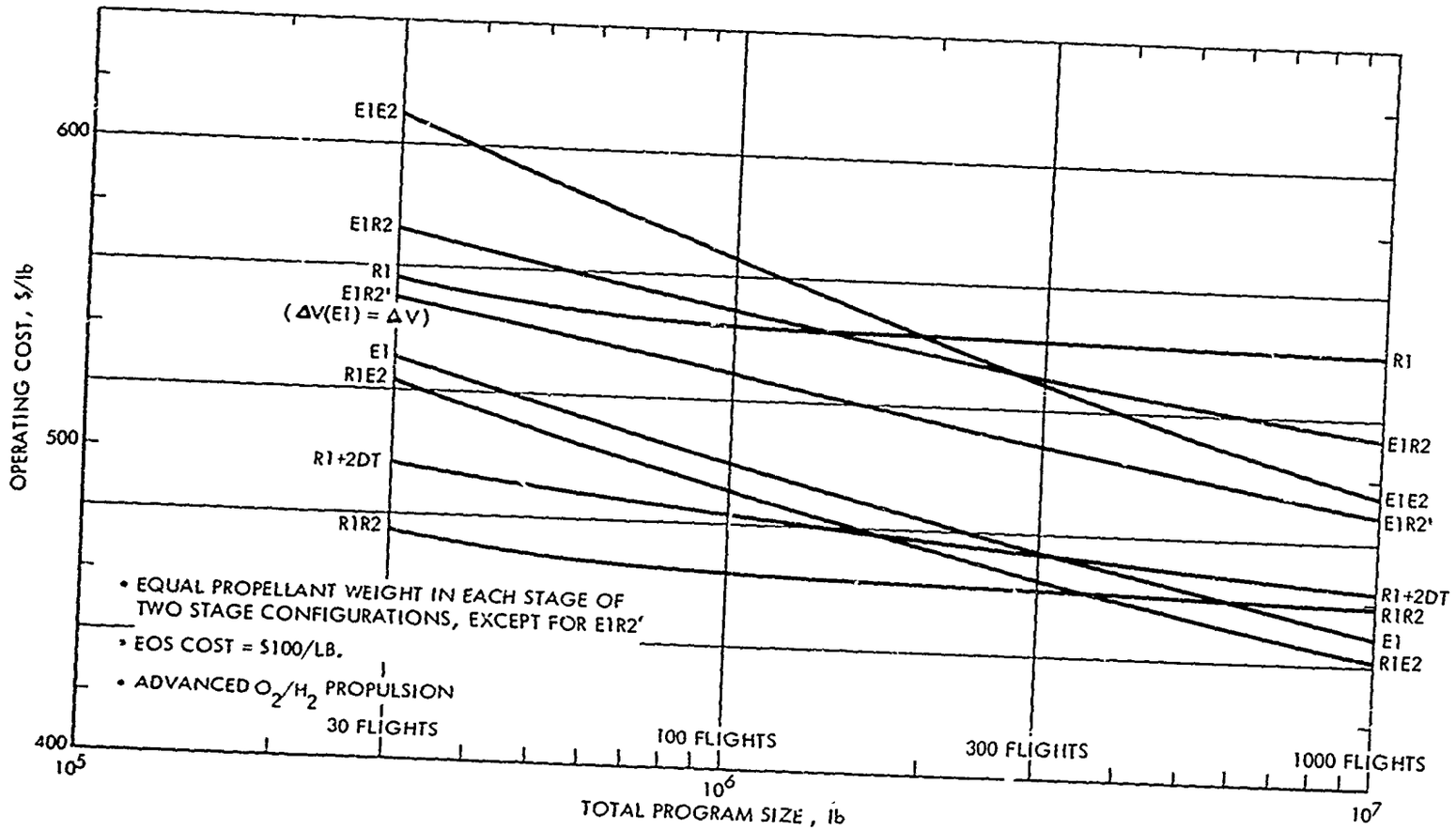

FIGURE 15. Operating Costs of Different Configurations Delivering 10,000-lb Payload One Way to Geostationary Orbit as a Function of Total Traffic

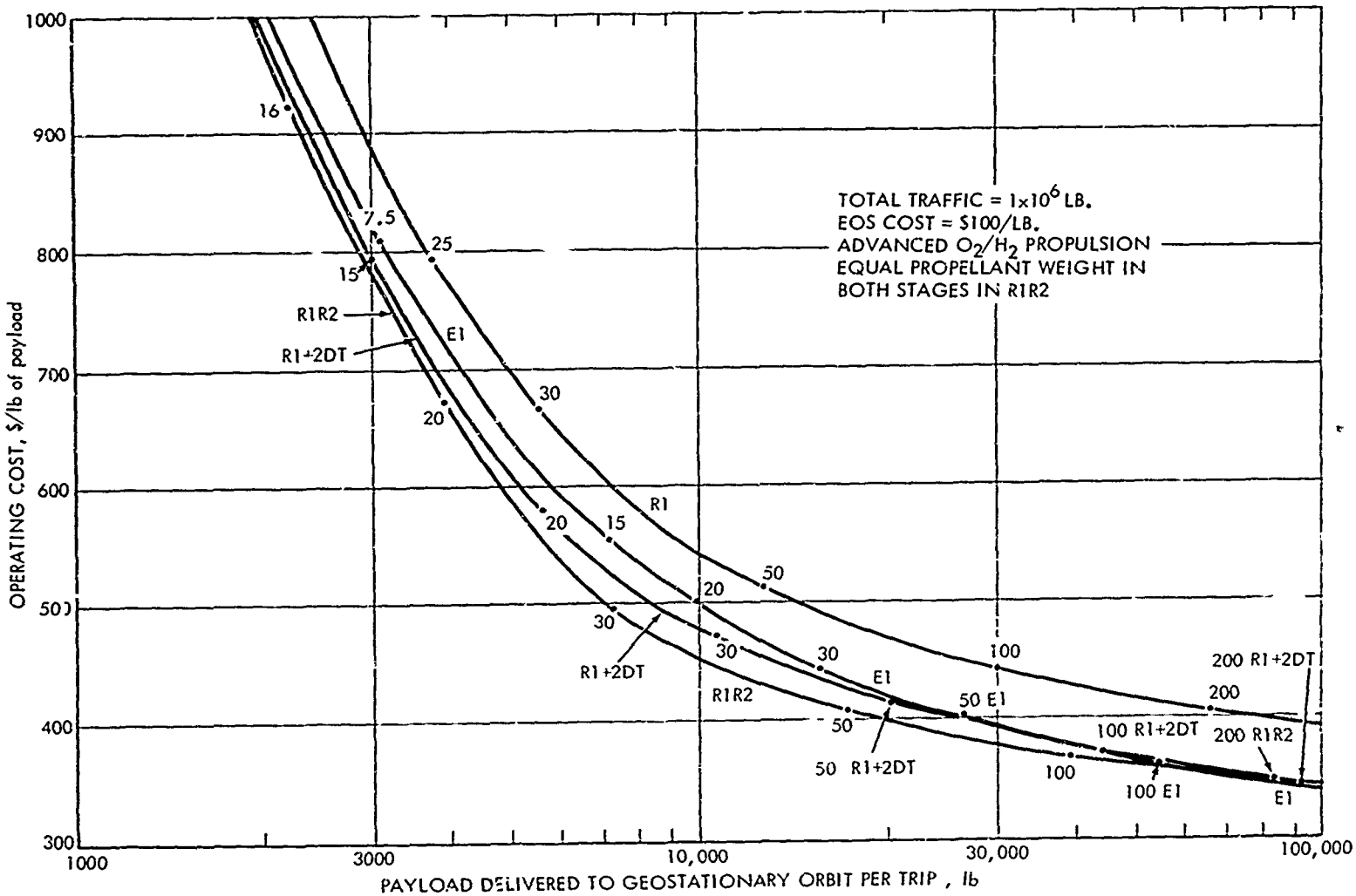

FIGURE 16. Operating Cost Comparison of 4 Configurations Delivering Payload One Way to Geostationary Orbit. Propellant Loadings in Kilopounds Indicated

Again, now over a wide range of payload sizes, the two-stage reusable vehicle provides the lowest operating costs, with the single-stage expendable becoming slightly lower in cost at large payload sizes. The two-stage reusable vehicle utilizes much more propellant than does the single-stage expendable, and would involve greater development costs; it could however, of course also be used to bring payload back from high orbits.

The question of costs of off-loaded vehicles is also an important one when actual programs with various sized payloads are being considered. When a payload less than the design capacity of a fixed vehicle is to be delivered, an economical practice is to load only that fraction of the capacity of the propellant required to perform the mission, thereby reducing somewhat the costs incurred in delivering the propellant to orbit via the EOS. The operating costs resulting from use of this off-loading process for reduced payload are shown in Fig. 17, for two configurations--the expendable single stage and the reusable stage plus drop tanks, both using advanced O_2/H_2 propulsion. Two off-loaded fixed vehicles having 10,000- and 30,000-lb design payload capabilities are compared with "rubber" vehicles.

Note from Fig. 17 the considerable increase in per-pound costs as large vehicles are used at less than capacity. Note further that the expendable vehicle increases in cost somewhat less rapidly than does the reusable vehicle with drop tanks. Thus, a reusable vehicle with drop tanks capable of delivering 30,000 lb has operating costs of $710/lb if off-loaded to deliver 10,000 lb versus $510/lb for a vehicle delivering its design capacity of 10,000 lb. The expendable vehicle on the other hand has operating costs, for a 30,000-lb maximum payload vehicle, of $565/lb at this 10,000-lb payload level, whereas the optimal 10,000-lb expendable vehicle has costs of $427/lb. Thus a 39% cost increase per pound is noted with the reusable versus a 32% increase for the expendable vehicle. These results suggest that vehicle design capacity should be selected toward the lower end of the payload size/frequency distribution, and that clustered small vehicles should be used to deliver large monolithic payloads.

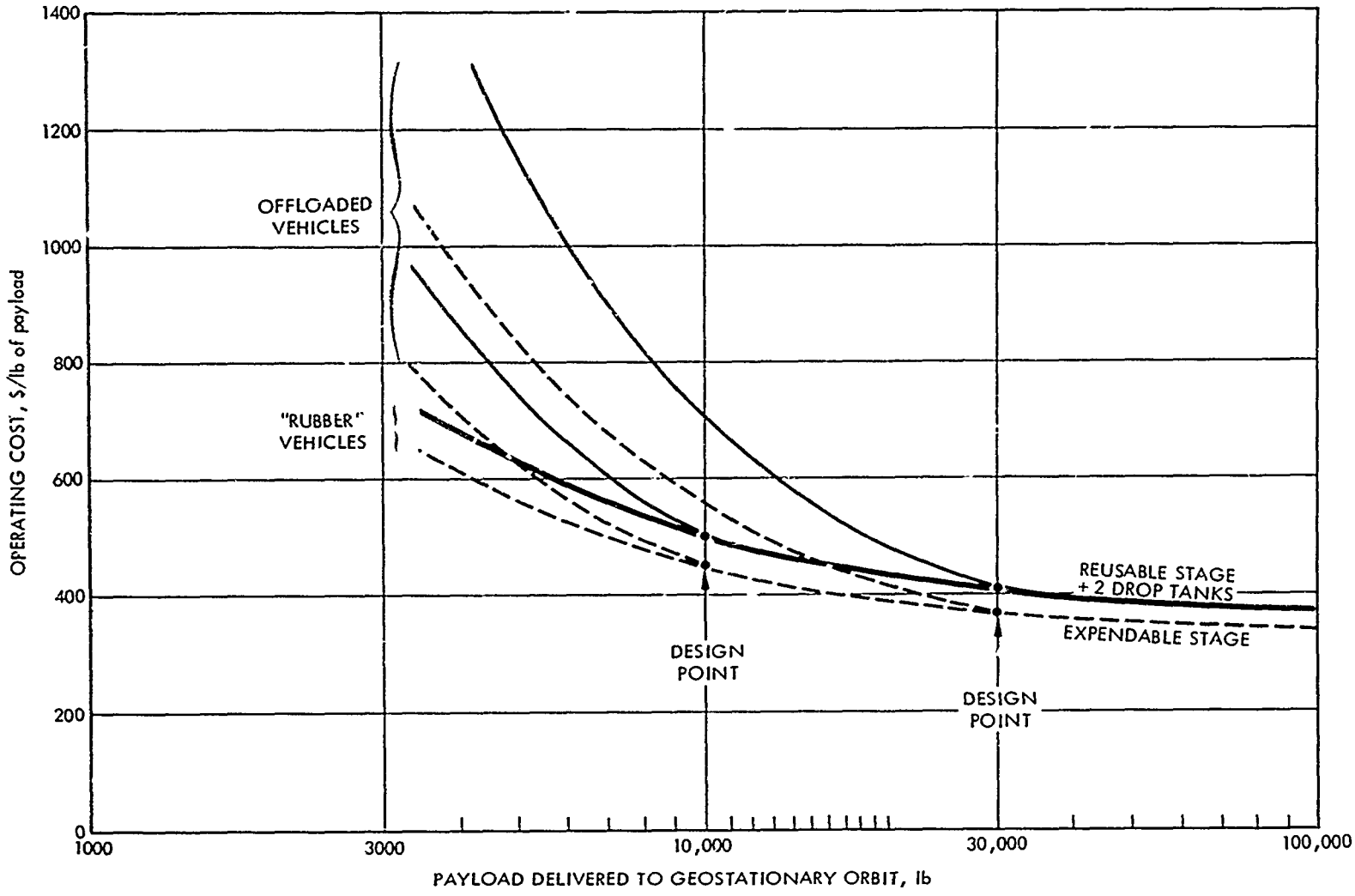

FIGURE 17. Comparison of Off-Loaded Vehicles to Rubber Vehicles in Terms of Recurring Costs of Operations. Advanced O_2/H_2 Vehicles Delivering Payload One Way to Geostationary Orbit.

Figure 18 shows operating cost trends with different propulsion systems in this mission for the single-stage reusable vehicle, the configuration which gains the most from improved propulsion. Lines of constant propellant weight are indicated to give an appreciation of gains at constant propellant weight. Thus at 50,000 lb of propellant, advanced F_2/H_2 delivers 16,800 lb at a cost of \$375/lb; advanced O_2/H_2 delivers 12,700 lb at a cost of \$508/lb, and conventional O_2/H_2 delivers 9100 lb at \$674/lb; comparison at constant propellant volume would, if meaningful, be even more dramatic due to the greater density of the advanced propulsion systems. The data plotted in Fig. 17 are very sensitive to the inert weight uncertainties.

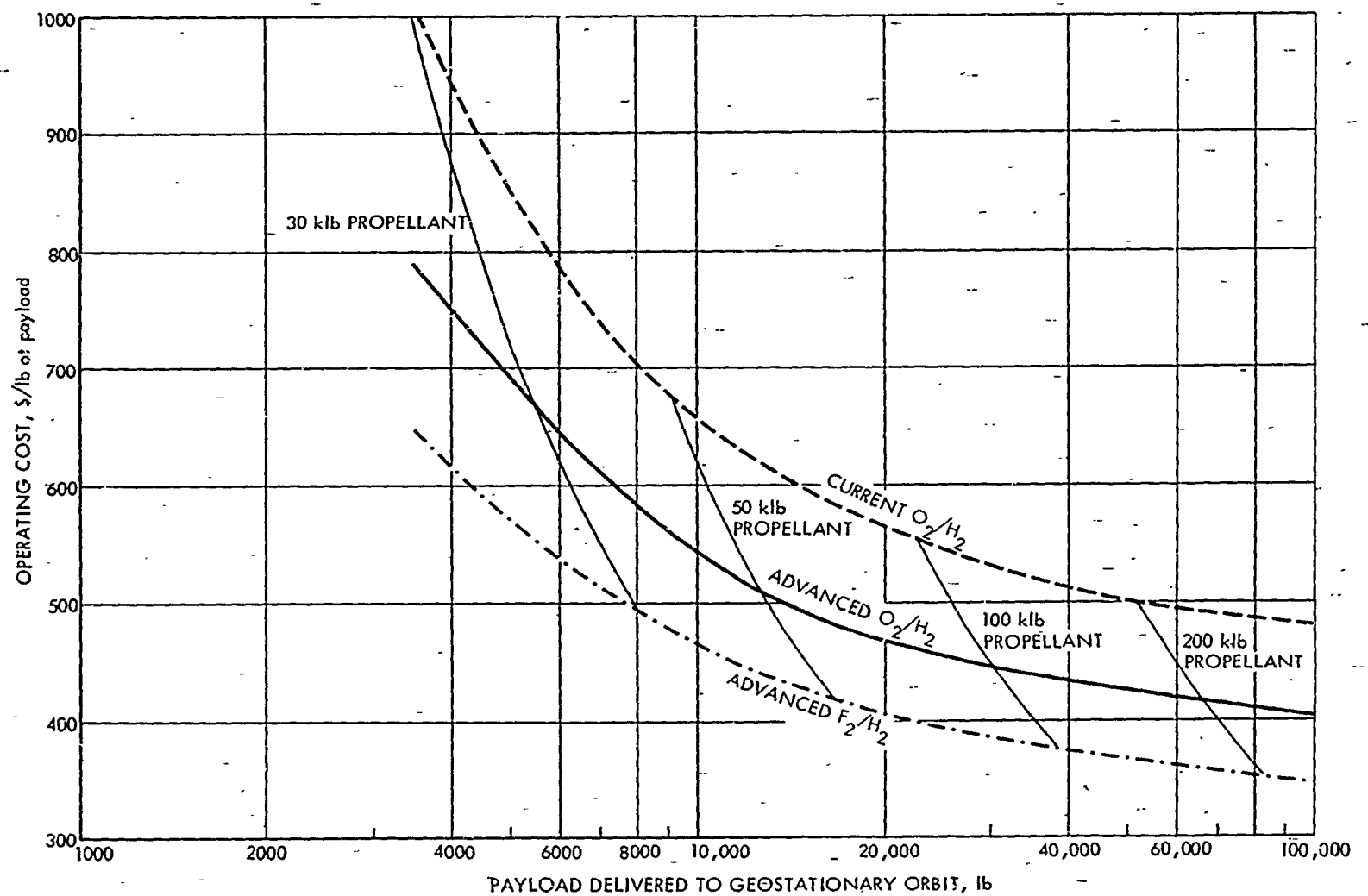

FIGURE 18. Operating Cost Comparison of Single-Stage Reusable Vehicles Delivering Payload to Geostationary Orbit Using Different Propulsion Systems. (Advanced Propulsion Shown to Maximum Advantage; Results Highly Sensitive to Mass Fraction.)

VI. CONCLUDING COMMENTS

The conclusions drawn from this effort were described in Section I. Here, only a few final comments and speculations will be offered.

As noted in the Introduction, this study has been limited to vehicles with a maximum of two stages, with (together) no more than two propulsion modules. The extension of work of this type to vehicles utilizing more conventional staging ratios, involving perhaps two parallel propulsion modules in a first stage, driving a third module in a second stage would be of interest; presumably lower total program costs would result in some regimes over those developed here. Configurations involving other clusters of stages could also be considered. Ultimately, this revised approach involves consideration of what might be done with some specific (existing or postulated) stage in terms of payloads and mission, an equally valid, if different, problem than the one addressed here.

Finally, in spite of uncertainties in the projections and correlations, it is clear that total transportation costs become surprisingly small even for fairly demanding missions, once the hurdle of getting material to low earth orbit at low costs is achieved. The figure of $300,000,000 per year (average costs for a 10-year period) to carry 300,000 lb/yr to the lunar surface (60,000 lb out and 10,000 lb back for each of 5 trips) was noted if earth-to-orbit costs of $100/lb can be achieved. The approximately constant cost growth factor between lunar-delivered payload and earth-orbit payload of about 10 to 1 makes the impact of low EOS costs clear. If it costs $100/lb to earth orbit, the additional cost to the lunar surface by modes considered here is $900/lb, leading to a total ($1000/lb) that appears to be acceptable: at $1000/lb to earth orbit, the cost growth is

$9000/lb to the lunar surface and what appear to be totally unacceptable figures ($10,000/lb) result. In short, at $100/lb EOS costs, the maintenance of a manned lunar base would not appear to involve excessive costs.

For interplanetary missions, the growth factors are larger than for lunar missions. As shown in Fig. 3, the transportation costs remain almost directly proportional to EOS costs over the full range of mission velocities investigated. It follows that achievement of low cost to earth orbit will have powerful impact on the transportation costs of space missions and eventually will reduce significantly the resources required for such missions. In a real sense, therefore, the achievement of low cost to earth orbit may "shrink" the solar system in a manner reminiscent of the "shrinking" of the earth which has resulted from low-cost jet transport.

REFERENCES

[redacted]

3. R.G. Finke and R.C. Oliver, "Comparison of Chemical and Nuclear Propulsion for Lunar and Cislunar Transportation Systems," IDA Paper P-687, October 1970.

4. North American Rockwell, Space Division, Orbit-to-Orbit Shuttle, Current Activity. Doc. No. 80 PDS 106241, September 1970.

5. General Dynamics, Preliminary Conceptual and Program Definition Studies for a Space Tug System, Report No. GDC-BNZ-70-001, February 1970.

6. D.E. Koelle and H. Kellermein, "Space Tug Design and Economy Optimisation," paper given at XXIInd International Astronautical Congress, Brussels, 20-25 September 1971.

7. D.G. Humphries, "The Choice of Space Tug," paper given at XXIInd International Astronautical Congress, Brussels, 20-25 September 1971.

8. General Dynamics, An Evaluation of Current and New Systems for an Orbit to Orbit Shuttle (OOS), PD 71-0036, February 1971.

9. D.C. Gregory, "Shuttle Design Starts in Orbit," Astronautics and Aeronautics, pp. 46-51, August 1971.

10. W.H. Stafford, H. Harlin, and C.R. Catalfairo, "Performance Analysis of High-Energy Chemical Stages for Interplanetary Missions," NASA Technical Note, D-1829, Undated.

11. Pratt and Whitney Aircraft, Oxygen-Hydrogen and Fluorine-Hydrogen Rocket Engine Parametric Data, PDS-2687, 19 January 1968.

12. D. Hart, private communication, August 31, 1971.

APPENDIX

ADDITIONAL ASSUMPTIONS AND CORRELATIONS

H_2 boil-off = 0.025 lb/day/ft^2 of tank area

Meteoroid protection (lb/ft^2) and insulation = $0.21 \left(\frac{\text{Total flight life}}{45 \text{ days}}\right)^{0.25} \times \alpha$

where $\alpha = \left(\frac{\text{Propellant vol}}{68{,}000 \text{ ft}^3}\right)^{0.167}$

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 computed as follows:

The velocity loss for injection into a transfer ellipse to synchronous altitude is given by

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

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

$$\Delta V_{gain} = (I_{sp}/445)^{0.27} (W_o/T)^2 [2.48 - 0.00072 (W_o/T)^2]$$

The velocity loss in attaining escape velocity is given by

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

The velocity losses for attaining velocities beyond escape are calculated in terms of

$$\Delta V_{loss} = \Delta V_{loss\ esc} \left(\frac{\Delta V}{\Delta V_{esc}}\right)^{x}$$

where

$$x = \frac{\ln\left(\Delta V_{loss\ synch} / \Delta V_{loss\ esc}\right)}{\ln\left(\Delta V_{synch} / \Delta V_{esc}\right)} - 0.55 \ln\left(\frac{\Delta V}{\Delta V_{synch}}\right)$$

For operations from 263 nmi, these losses are reduced by the factor

$$\frac{g_{263}}{g_{100}}$$

A velocity "pad," or allowance for off-nominal performance, of 3/4 of 1 percent beyond each computed velocity increment is also included.

Weights

Contingency	= 3% of dry weight
Ullage volume	= 3% of propellant volume
Residuals	= 1% of usable propellants
Subsystems (fraction of dry weight)	= 3% (chem)
	= 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

$$W_{tank} = S\ (2.32 + MP + 4S/10{,}000\ ft^3) \times \left(\frac{\text{propellant density}}{20.24\ lb/ft^3}\right)^{1/2}$$

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)

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

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

H_2 boil-off = 0.025 $lb/day/ft^2$ of tank area

$W_{interstage}$ = 0.01* (sum of weight above)

Correlations

First Unit Costs (1969 dollars):

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

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

Subsystems = $2.4 \times 10^4\ [W_{ss}(lb) + W_{gc}(lb)]^{0.725}$

Learning curve slope = 90% (applied to individual units on each doubling of quantity)

---

*Per 15-klb-thrust engine.

R&D Costs (1969 dollars):

$$\text{Chemical engine} = 50\times10^6 + 4.8\times10^6[T_{vac}(\text{lb})]^{0.32}\,(p_c/1000\ \text{psi})^{0.2} \times \beta$$

where $\beta = 0$, current O_2/H_2

$= 1.5$, advanced O_2/H_2

$= 2.0$, F_2/H_2

Airframe $= 0.759 \times 10^6\ [W_{af}(\text{lb})]^{0.578}$

Subsystems $= 25 \times$ first unit cost

Checkout Costs per flight = 1% vehicle hardware cost

NOTE: Production and R&D cost correlations were derived principally from published contractor costs for the SATURN, TITAN, and ballistic missile programs. Engine and airframe R&D costs are comparable to those obtained from current NASA cost methodology for the Space Shuttle showing agreement within about 10%. Hardware costs, however, do not follow the NASA cost correlations. The airframe production costs have been taken to be one-quarter of NASA experience to reflect the new low-cost fabrication techniques expected to be used for the Space Shuttle drop tanks. The engine production costs are also about one-quarter of NASA predictions and about one-half of USAF experience; it is presumed that by the 1980's rocket engine production efficiencies will have improved by that ratio. Checkout and handling costs are typical of Air Force practice and do not include agency overhead costs, as would be typical of NASA practice.