1963

XIV

8

GENERAL DYNAMICS
Convair Division

A2136-1 (REV. 5-65)

GD/A63-0065

NEXUS
CONCEPT OF A LARGE REUSABLE EARTH LAUNCH VEHICLE

by

Krafft A. Ehricke*

and

Freeman D'Vincent**

GENERAL DYNAMICS/ASTRONAUTICS
San Diego, California

*Director of Advanced Studies

**Engineering Staff Specialist
Advanced Studies Office

ABSTRACT

Aspects of Earth-to-orbit delivery are discussed and a cost analysis of the logistic operation and the cost of orbital operations are presented. Probabilities of success of orbital delivery and the operational and economic aspects of establishing large orbital installations and maintaining a large transportation volume in the 1975/85 time period are compared for the two cases of using a large number of Saturn V versus a smaller number of 1-stage chemical Post-Saturn launch vehicles. Performance parameters of chemical, chemonuclear and nuclear launch vehicles are compared. The concept of a blunt launch vehicle configuration, referred to as NEXUS, is presented in detail. Applications of this configuration to chemonuclear propulsion and to a 50 ft diameter version of Saturn V with recoverable first stage are discussed.

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

1. Introduction

The history of transportation technology shows that requirements imposed by the need for a growing transportation volume are met in three steps: First, the number of existing carriers is increased; second, the capacity of the individual carrier is raised until some practical size limit is reached; finally, through technological advances, a quantum jump is achieved, raising the carrier capacity through greater speed of transportation or by attaining a higher payload fraction or by achieving both simultaneously with a new and more advanced carrier.

This process occurs in response to a demand for growing transportation capacity on land, on sea and in the air. Space is no exception. Throughout many decades of the past, adequate prognosis of future transportation needs and careful planning of how to satisfy these needs has been of crucial importance to the competitive growth and survival of private transportation business in this country. As far as space transportation is concerned, Earth is "this country" and qualified nations represent the individual enterprises engaged in competition for technological eminence and the manifold benefits expected to result therefrom. The approach to the problem is, therefore, essentially the same.

It is the object of this paper to suggest a concept of how a growing demand for space transportation capability can be met in a timely and economical manner when it arises. The question, if it should arise in the first place, is not subject to discussion here. If and when it arises, it will have to be faced first in the field of Earth-to-orbit transportation. Therefore, Earth launch vehicle

(ELV) planning is necessarily the first order of business.

Applying the three steps of transportation volume growth, referred to above, to the ELV problem, they take the following form:

(1) Increase of the number of Saturn V launches per unit time.

(2) Develop a chemically powered Post-Saturn ELV of 4 to 8 times the orbital payload capability of Saturn V.

(3) Develop a chemonuclear or nuclear ELV of superior payload fraction (ratio of payload weight to lift-off weight).

This paper contains results of a Post-Nova launch vehicle study conducted since March 1962, and presently continuing, under the direction of the Marshall Space Flight Center, Future Projects Office. This study is one of the study series assisting in the selection and definition of the next large launch vehicle after Saturn V. It is primarily aimed at the question of probable operational life of the vehicle concept which is determined by the state-of-the-art expected during the late seventies and early eighties. Results of this study have produced conceptual designs for several promising launch vehicles, now transferred to the main NOVA studies. The chemical NEXUS concept presented in this paper is one of these configurations.

2. Orbital Operation

The Earth-to-orbit transportation volume is determined by the number and the extent of orbital operations. Orbital operation is defined here as a process of establishing or of maintaining and servicing orbital installations. Orbital installations can be "permanent" (space stations) or temporary (lunar or planetary vehicles). The four principal operational modes for establishing orbital installations are depicted in Fig. 1.

Servicing and maintenance becomes a significant factor only for permanent orbital installations and consists primarily of rotation of personnel, delivery of food and other expendable necessities; and, of delivery of spare parts and replacements. The transportation requirements for servicing and maintenance are always considerably below those needed for the initial establishment. The latter ones, therefore, pace the expansion of the ELV transportation capacity.

3. Earth-To-Orbit Delivery

Delivery is defined here as the process of payload transfer from the launch pad to a rendezvous condition with a point in the target orbit. The payload is in rendezvous condition when it moves in the immediate vicinity (20 to 100 ft distance) of the target point (e.g. orbital operations center, orbital launch facility or a partially completed orbital establishment) at very nearly the same instantaneous radial velocity and the same angular momentum as the target point. Successful establishment of planned rendezvous conditions completes the delivery process. Everything thereafter is categorized here as orbital operation (primarily, mating or fueling of modules).

The sequence of principal events during delivery is summarized in Tab. 1 for a two-stage vehicle exemplified by Saturn V and for a more advanced single-stage vehicle. It is seen that in this model the mission of the ELV proper is completed--as far as delivery is concerned--with the successful separation of the payload. The ELV may deliver the payload directly into the target orbit into a near-rendezvous condition, leaving it to the propulsion system attached to the payload to carry out a comparatively small terminal maneuver to attain full rendezvous condition. If no direct delivery occurs, the payload either is launched into a parking orbit, or the method of intercept delivery (ref. 1) is applied: that is, the ELV enters an elliptic orbit, slightly overshooting the target orbit. Prior to intersection with the target orbit the payload is separated and, with its own propulsion system accomplishes rendezvous. In either case, the number of events remains the same for the ELV mission, while the number of principal maneuvers for the payload propulsion system is either one or two (Fig. 2). In the case of 1-burn (for payload propulsion system) intercept-delivery, the number of events is basically the same as for the sequence of events outlined above. The overall probability of delivery is

$$P_D = R_{ELV} \, R_{E-7} \, R_{E-8} \tag{1a}$$

The reliability of the events E-1 through E-6 directly related to the ELV is

$$\prod_{E-1}^{E-6} R_E = R_{ELV} \tag{1b}$$

The reliability R_{ELV} is assumed to vary as shown in Fig. 3 for the 2-stage Saturn V as the result of the cumulative launches between 1970 and 1990. The assumed reliability of events E-7 and E-8 is also shown in Fig. 3. These reliability curves are based on component reliability estimates and on experience curves for various ballistic missiles and space boosters (ref. 2). Since a reliability analysis is not the purpose of this paper, these curves are presented here as one of several reliability models which will be used in the subsequent cost analysis. The reliability curve for the 1-stage ELV is based on the assumption of all-chemical propulsion with initial operational availability in 1975. The propulsion system is assumed to consist of advanced O_2/H_2 engines and a configuration described as NEXUS configuration below. Since the vehicle does not stage and since the number of principal events involved in the delivery is, therefore, smaller, a higher rate of growth is indicated. It is believed that the reliability figure of 0.945 in 1990 for a 1-stage vehicle operational since about 1975, is conservative.

4. Operating Cost of Establishing an Orbital Installation

4.1 Total Operating Cost

The total operating cost, C, of establishing a particular orbital installation consists of the sum of the total operating cost of the logistic operation, C_{log}, of which the cost of transportation into orbit is the major part, plus the total operating cost of the orbital operation, C_{orb},

$$C = C_{log} + C_{orb} \tag{2}$$

The total operating cost is defined here, as in ref. 3, as being the sum of direct and indirect operating cost. The direct operating cost comprises the cost of article production, propellant, transportation from factory to launch site, launch cost, maintenance and repair or refurbishing, flight crew (if any) and other recurring costs. The indirect cost includes the cost of range operation, GSE per launch complex, launch facility, article development and other non-recurring cost. The term "article" refers to either the ELV or the net payload package and its associated propulsion system, needed for events E-7 and E-8 (Tab. 1). The individual cost elements listed above must, of course, be checked as to their applicability to the particular case under consideration.

The logistic operation consists of launch preparations and delivery, beginning with lift-off and terminating with rendezvous condition of the payload. Cost considerations enter both phases. Reliability is primarily a problem in the delivery phase and has been specified in Sect. 3 above.

4.2 Cost of Logistic Operation

A large number of interesting cost analyses have been prepared in the past five years, regarding the cost of development and operation of launch

Tab. 1 SEQUENCE OF EVENTS MODEL FOR EARTH-TO-ORBIT DELIVERY

EVENT	2-STAGE VEHICLE (SATURN V)	1-STAGE VEHICLE
E-1	Launch of S-IC	Launch
E-2	Cut-off of S-IC	Cut-off of some engines
E-3	Staging of S-IC	Jettisoning of fairing
E-4	Ignition of S-II	Cut-off of residual engines
E-5	Jettisoning of fairings	Separation of payload
E-6	Cut-off of S-II	Rendezvous maneuver of payload
E-7	Separation of payload	
E-8	Rendezvous maneuver of payload	

vehicles of various sizes. Some of the related work and additional references are found in references 3 through 5. Moreover, cost and development time predictions become increasingly vague the more ambitious the project. For these reasons, and since detailed cost analysis is not the objective of this paper, approximations will be used which assure the development of a possible and perhaps likely cost model which can be used for parametric purposes and which takes into account the effect of increasing experience with large launch vehicles, and the effect of the rising cost of production and living.

The direct operating cost for vehicles of the Saturn V type which are expendable and land-launched, consists to about 90 percent of vehicle production cost. Therefore, for the present discussion, the variation in direct operating cost can essentially be reduced to a discussion of the production cost. This cost decreases with increasing cumulative production number due to growing experience and improving production efficiency. The cost increases with time, however, due to rising labor and material cost and due to continued product improvement of vehicle subsystems and components. Thus, the production cost is primarily a function of the cumulative production number and of time. From the volume of sales and deliveries of aircraft corporations in the 1950/60 period, it was shown in ref. 3 that the production cost, in terms of dollars per lb hardware delivered, increased almost by a factor of six during that decade, from \$21 to \$118. This includes the effect of transition from aircraft to missiles and, to a lesser extent, from missiles to spacecraft. The cost of airliner production roughly doubled during the past decade.

The effect of transition from missiles to spacecraft on cost will make itself felt in the 1960/70 period and will be caused primarily by a trend toward further reduction in production numbers resulting in more man-hours per unit weight, the (rightful) demand by NASA and DoD for higher product quality and reliability, by further increase in the proportion of electronic and other high cost equipment and material (heat shields, etc.), by a further rise in average salaries due to an increase in the proportion of highly skilled personnel all the way from design to manufacturing and testing and by other factors, connected with more studies, analysis and research. This trend may be expected to continue into the 1970/80 decade with the advent of nuclear propulsion, nuclear electric power generation, extensive application of cryogenic technology and, on the operational side, with the introduction of orbital facilities, lunar bases and manned planetary operations into the technological frame of reference.

This cost-increasing trend is taken into account for the 1970/1985 time period by means of an exponential function e^z. For Saturn V,

$$z = 0.03\,(0.9Y + 0.3) + 0.00022\frac{\nu}{Y^2} - 0.00000008\frac{\nu^2}{Y^6} \qquad (3)$$

where Y represents the cumulative number of years, starting with mid-1970 (i.e. Y = 1 in mid-1971). The term in parenthesis indicates that the first term, which represents the increase in cost with progressing time, is not directly proportional to time, modulating the growth of z, taking into consideration factors such as saturation of plants with highly skilled and experienced personnel (i.e. level-off trend in the process of transition in the personnel composition), amortization of the investments in basic production capabilities (for cryogenic fluids and special "space age" materials) and in other basic facilities (large simulators, special test facilities). The second term represents the time correlated effect of the production number ν. With increasing production, the cost per unit weight of hardware will decrease. However, this decrease will be different if a certain production number is to be attained during the "first" year ($Y = 1$) than two or three or more years later. In the first case, the cost-reducing effect will be less pronounced because of extensive tooling and facility investments and because rapid increase in the production number of a relatively young product is bound to cause mistakes which must be charged against the production cost. This effect, however, should diminish rapidly with time, as expressed by Y^2 in the denominator.

The third term modulates the effect of ν in the first year ($Y = 1$) and takes into account that for Saturn V the year 1971 is not really "$Y = 1$". The third term is designed to give the increase in production number to very high values of Saturn V in 1970 a greater cost-reducing effect than it would be justified for the first operational year of a new product.

Counteracting this cost-raising trend is the fact that with increasing production number, independent of time, there is going to be a reduction in cost due to progress on the learning curve, more efficient production methods, smaller reject quantities and amortization of the production facilities and equipment proper. This trend can mathematically be defined by the experience curve $a\nu^{-b}$ which, in logarithmic form represents a straight line. From data concerning the V-2 rocket the B-29, B-47, B-52 and others for production numbers up to 1000, presented in ref. 3, a value of $b = 1/6$ is indicated, corresponding to about 90 percent learning. Most of the experience cases mentioned before are, however, based on time periods which are shorter than the 15 years considered here. In practice, progress on the learning curve makes b a function of time also. Neglecting this aspect here (since some of this, at least, already has been incorporated in z) we then apply the following relation to the determination of the production cost per lb of hardware

$$K_{prod} = a\nu^{-b} e^{z} \tag{4a}$$

which becomes, under the given assumptions for the Saturn V ELV,

$$K_{prod} (\$/lb) = a\nu^{-1/6} \exp \quad 0.03 (0.9Y - 0.3) + 0.00022 \frac{\nu}{Y^2}$$

$$- 0.00000008 \frac{\nu^2}{Y^6} \tag{4b}$$

This relation is plotted in Figs. 4 and 5 for a = 80. Fig. 4 shows the effect of varying the production number in a given year. For a fixed production number the increasing cost of living is a dominant factor. Increasing the production number is more effective, in terms of cost reduction, in later years, since by then earlier investments are amortized, the entire manufacturing and quality control process is more "debugged" and the capability of handling increases in production efficiently and without costly errors is increased. For comparison, the relation $a\nu^{-b}$ is shown, which does not take cost-increasing effects into account. Fig. 5 illustrates the effect of time on the production for given production number. As would be expected, the cost-increasing effects are most dominant at prolonged manufacturing at a low production level. This effect is less pronounced, and may even be reversed temporarily, at higher production numbers.

The associated direct cost of delivery per lb of net payload (direct cost effectiveness) is

$$C^{**}_{O,D} \approx \frac{K_{prod}}{0.9} \frac{W_d}{W_\omega} \text{ (\$/lb Pld)} \tag{5}$$

for the expendable, land-launched Saturn V, where W_d is the dry weight and W_ω the net payload weight of the vehicle. The associated indirect cost effectiveness is

$$C^{**}_{I,D} = m\, C^{**}_{O,D} \text{ (\$/lb Pld)} \tag{6}$$

where m = m(N), N being the annual launch rate. Cost analyses made in connection with the Post-Nova study indicate a trend as shown in Fig. 6, for m as function of the cumulative launches N, hence of ν, assuming that the vehicles are not stockpiled in significant quantities. The total cost effectiveness of payload delivery into orbit with Saturn V is, therefore, approximately

$$C^{**} \approx \frac{K_{prod}}{0.9} \frac{W_d}{W_\omega} (1 + m) \text{ (\$/lb Pld)} \tag{7}$$

Where $W_d \approx 440,000$ for both, first and second stage of Saturn V. The net payload is 250,000 lb or less. The total operating cost per launch is then

$$C' \approx \frac{K_{prod}}{0.9} W_d (1 + m) \tag{8}$$

The total operating cost for a given orbit lift operation is,

$$C_{op} \approx N_{op}\, C' \tag{9}$$

where N_{op} is the number of launches required for a given transport volume. The annual total operating cost follows to be

$$C \approx N\,C' \qquad (10)$$

where N is the number of launches per annum.

Two models of growth (case A and case B) of the Saturn V launch rate have been assumed and are shown in Fig. 7. They probably bracket the actual case. The associated cumulative net payload weight delivered into orbit is shown in Fig. 8 for three Saturn V payload levels. Case A assumes a comparatively moderate growth of cumulative payload in orbit, reaching 90 to 115 million lb by 1985. In case B, between 185 and 230 million lb will have been delivered into orbit by 1985. Mean total cost effectiveness and associated parameters have been determined for each of these cases, based on Eqs. (7) through (10) and Figs. 3, 4 and 6. They are listed in Tab. 2. Based on these values an approximate variation of total cost effectiveness of Saturn V versus time is shown in Figs. 9 and 10 for case A and B, respectively. All that can realistically be said about these figures is that they are likely and perhaps tend to be optimistic rather than conservative. This is done intentionally, because, if a larger, Post-Saturn ELV compares advantageously with Saturn V, such result is more conclusive if Saturn V is treated optimistically.

For the Post-Saturn vehicle a payload capability into orbit of 10^6 lb has been assumed. A reusable version with an average operational life of 10 launches and an expendable version of a single stage chemically powered ELV have been considered. Corresponding to the orbital transportation models A and B assumed in Fig. 8 for 250,000 lb payload, the same payload build-up has been

Tab. 2 COST EFFECTIVENESS AND ANNUAL LAUNCH COST OF SATURN V FOR VARIOUS ORBITAL PAYLOAD LEVELS IN THE 1970-1985 PERIOD

TIME PERIOD	1970/75		1975/80		1980/85	
CASE	A	B	A	B	A	B
Total orbital payload (W_ω = 250 k) (10^6 lb)	15	30	30	65	65	130
(W_ω = 220 k) (10^6 lb)	13.2	26.4	26.4	57.1	57.1	114.2
Mean production cost, $\overline{K}_{prod}$ ($/lb)	48	44	48	44	50	45
Mean ratio of indirect to direct oper. cost, m	0.77	0.7	0.64	0.61	0.54	0.475
Probability of successful delivery, P_D	1.0	1.0	1.0	1.0	1.0	1.0
Total number of launches, N_{op}	60	120	120	260	260	520
Annual number of launches, N	12	24	24	52	52	104
Total cost of effectiveness, C^{**} ($/lb Pld):						
W_ω = 250 k	166	145	157	137	150	130
W_ω = 220 k	189	165	178	156	171	147
Cost per launch, C' (10^6 $)	41.5	36.4	39.2	34.3	37.6	32.6
Total operating cost, 5-yr period, C_{op} ($)	2500	4360	4700	8930	9800	16,900
Annual total operating cost, ($/a)	500	875	940	1780	1950	3,380
Probability of successful delivery, P_D	0.792	0.792	0.87	0.87	0.93	0.93
Total number of launches, N_{op}	76	152	138	300	280	560
Average annual number of launches, $\overline{N}$	15.2	30.4	27.6	60	56	112
Total operating cost, 5-yr period, C_{op} ($)	3165	5525	5405	10,300	10,550	18,200
Annual total operating cost ($/a)	634	1110	1080	2060	2,110	3,640
Total cost of effectiveness, C^{**} ($/lb Pld)						
W_ω = 250 k	210	183	181	158	160	140
W_ω = 220 k	240	209	200.5	180	183	158

followed with the Post-Saturn vehicle (Fig. 11) for which 1975 has been assumed to be the first full year of operational state. The cost effectiveness of such more or less hypothetical vehicle can vary considerably, depending on many detailed assumptions which cannot be discussed here. Typical variations of the total cost effectiveness with time for the type of vehicle under consideration are shown in Fig. 13 for the cases A and B and for the reusable and the expendable version. Typical uncertainty limits whose range is characteristic for all curves have been indicated for the upper curve as the apparently most likely curve for the case in question. The total cost effectiveness figures for the expendable versions probably are on the conservative side. Reusability is indicated to pay off more on the long run than initially where lower reliability will permit fewer vehicles, if any, to live through their full operational life of 10 launches and where recovery and refurbishing operations are less routine.

4.3 Cost of Orbital Operation

The cost of the orbital operation is composed of the orbital labor cost and of the ground operational cost. The latter, consisting primarily of tracking the orbital installation and of associated data evaluation is small compared to the cost of orbital labor, because it is based on tracking facilities and crews which are also used for tracking other satellites and deep space vehicles, as well as in connection with Earth-to-orbit logistic operations.

The bulk of the direct cost of orbital operations is connected with the establishment and the maintenance of a human labor force in orbit for the duration of the particular orbital operation (i.e. primarily the establishment of an orbital installation as defined in Sect. 2). A small amount (approximately 10 to 15 percent of the orbital labor cost) will have to be added for maintenance and/or replacement of orbital support equipment (OSE, the analog of the GSE). This OSE and its transport into orbit constitutes the bulk of the indirect (non-recurring) cost of the orbital operation. Of all these cost items, the orbital labor cost appears to be by far the largest single cost item, although exceptions are possible. Fortunately, the orbital labor cost is comparatively most accessible to a general analysis. The cost of the OSE depends upon the type of orbital operation; the cost of maintaining and servicing the OSE is largely a function of the duration of the particular orbital operation or of the sequence of orbital operations, all assumed to be using the same OSE.

For the hourly orbital labor rate, in terms of dollars per labor hour based on the period of the particular orbital operation, T_{op}, the following equation was developed,

$$(C_{h,OL})_{T_{op}} = \text{f (cost of special job training) + f (cost of transportation to and from orbit) + f (cost of living) + f (cost of housing) + f (cost of operating and maintaining orbital housing for the work force)} \tag{11}$$

$$(C_{h,OL})_{T_{op}} = \frac{C_{T\,rng}}{24 N_D T_D f_w} + \frac{1}{24\, T_{op} f_w} \quad \frac{T_{op}}{T_D} C_{T_r} + C_L + \frac{C_{OH} + C_{OM}}{\overline{N}_P} \tag{12}$$

where (1 day = 24 hrs),

$C_{T\,rng}$ (\$) = cost of special training of person for his orbital job, as paid for by the Government

N_D = number of orbital duty periods, T_D, of a given person during the period of orbital operation, T_{op}

T_D (days) = orbital duty period (i.e. time between ascent into orbit and return)

f_w = fraction of 24-hr period spent working

T_{op} (days) = total period of the particular orbital operation

T_{op}/T_D = number of transportations to and from orbit per equivalent person (if the ratio is not a full number, the next high full number must be taken)

$C_{Tr, P}$(\$) = transportation cost per person to orbit and back

C_L (\$) = cost of living

C_{OH} (\$) = cost of orbital housing for the labor force (no development cost)

C_{OM} (\$) = cost of operation and maintenance of orbital housing for the labor force

$\overline{N_P}$ = average number of personnel in the particular orbital labor force

In particular, it is

$$C_{Trng} = Y\, C_y \tag{13}$$

Y (years) = number of years of special training for orbital work

C_y (\$/yr) = annual cost of special training

$$C_{Tr, P} = (C^{**}_{ETO} + C^{**}_{OTE})\, W_P \tag{14}$$

C^{**}_{ETO} (\$/lb) = Earth-to-orbit personnel transportation cost (\$/lb of person and personal equipment)

C^{**}_{OTE} (\$/lb) = Orbit-to-Earth personnel transportation cost (\$/lb of person and personal equipment)

W_P (lb) = weight of person and personal equipment

$$C_L = T_{op}\, C^{**}_{Tr}\, (\dot{w}_F + \dot{w}_W + \dot{w}_X) \tag{15}$$

C^{**}_{Tr} (\$/lb) = cost of cargo transportation into orbit per pound of **cargo**

$\dot{w}_F$ (lb/d) = daily consumption of food which has to be replaced by **supply** from Earth (mostly solid, since water is recycled)

$\dot{w}_W$ (lb/d) = water loss per person per day

$\dot{w}_X$ (lb/d) = expendables per person per day (e.g. tooth paste, tissues, etc.)

$$C_{OH} = C_{production} + C_{launch} = W_{OH}\,\overline{C}_{prod} + W_{OH}\,C^{**}_{Tr} \qquad (16)$$

W_{OH} (lb) = weight of orbital housing facility

$\overline{C}_{prod}$ (\$/lb) = mean production cost of facility

$$C_{OM} = 0.003\,\frac{T_{op}}{365}\,W_{OH}\,C^{**}_{Tr} + C_{GO} + C_{crew} \qquad (17)$$

assuming that the cost of maintenance corresponds in the average to transporting daily 0.3% of the weight of the orbital housing facility into orbit at \$150/lb transport cost.

$C_{GO}(\$) = C_{Daily} \cdot T_{op}$ = cost of ground operations, i.e. average daily cost times period of operation (tracking, etc.)

C_{crew} (\$) = cost of crew to run and maintain the orbital housing facility. This cost is assumed presently to be zero, since "facility duty" can be handled by the labor crew; however, if a special crew or a specialist were required, apart from the work force, the associated cost would be carried under C_{crew}

Based on values selected for the various parameters, listed in Tab. 3, the orbital labor cost has been computed for orbital operations lasting 360 days, 180 days and 720 days, respectively. The results are shown in Figs. 14 through 17. Fig. 15 shows a cost breakdown for a 360 day operational period. This cost breakdown is typical also for the two other operational periods shown. From this figure it is seen that the cost of special training of the orbital labor force represents the dominant cost item and that the period of orbital duty and the number of orbital duty periods are the most important variables. If the

Tab. 3 COMPUTATION OF ORBITAL LABOR COST

Period of given orb. operation, T_{op} (d)		180	360	720		
Orbital duty period, T_D (d)	30	60	90	180	270	360
Number of orbital duty periods per person, N_D						
for T_{op} = 180 d	1, 2, 3	1, 2	1			
360 d	1···6	1···4	1, 2, 3	1		
720 d	1···12	1···6	1···4	1, 2, 3	1, 2	1
Fraction of 24-hr period spent working, f_W			1/3			
Number of years of special training, Y (years)			2			
Annual cost of special training per person, C_Y (\$/a)[1]			400,000			
Average weight of person & personal equip., W_P (lb)			200			
Cost of personnel transp. to & from orbit, $C^{**}_{Tr,P}$ (\$/lb W_P)			100			
Daily consumption of expendable food, $\dot{w}_F$ (lb/d/person)			3.66			
Daily water loss (to be replaced), $\dot{w}_W$ (lb/d/person)			0.35			
Daily expendables (to be replaced), $\dot{w}_X$ (lb/d/person)			0.74			
Cost of cargo transportation to orbit, C^{**}_{Tr} (\$/lb cargo)			150			
Average number of personnel, $\overline{N_P}$ (persons)			50			
Weight of orbital housing, W_{OH} (lb)			200,000			
Mean production cost of orbital facility, $\overline{C}_{prod}$ (\$/lb)			80			
Average daily cost of ground operations, directly charged to the orbital operations budget, C_{daily} (\$/d)			10,000			

[1] Special training comprises all ground and orbital training required to render a person capable of handling expensive orbital payloads professionally and with high confidence level as a fully effective member of the orbital team.

orbital duty period is brief, the cost of special training remains the dominant factor even if the cost is one half or one third of the $400,000 value assumed in Tab. 3. The cost of personnel transportation is a comparatively small item in the framework of a 360 day orbital operation. However, Fig. 18 shows that its contribution increases with decreasing period of orbital operation and increasing number of personnel rotations, expressed by the ratio of period of orbital operation to period of duty of the individual.[1)] The cost of living contribution nominally is not a function of T_{op}, since both, food requirement and number of labor hours vary in the same manner with T_{op}. The contribution of orbital housing to the hourly labor rate exceeds that of transportation by a factor of 4 and higher, as shown in Fig. 19 for various average numbers of personnel, $\overline{N}_P$, and on the basis of the specifications listed in Tab. 3 and in Fig. 9.

These data show:

1. For periods of orbital operations of 100 days or more, personnel transportation costs play a comparatively minor role in the overall hourly labor rate, provided the number of crew rotations does not exceed 2 for a 100 day operation and 12 for a 720 day operation. This conclusion is correct even when doubling the transportation cost of $100/lb assumed here (postulating an all-recoverable 2-stage personnel transport vehicle).

[1)] As far as transportation cost is concerned, it does not matter, whether or not the same individual is involved in another period of duty during the same period of orbital operation; i.e. N_D has no effect on the transportation cost, only on the contribution of the special training cost to the hourly labor cost.

2. The cost of supplying the orbital crew with expendable items, primarily (on a per person basis) food (3.66 lb/d/p), make-up water (0.35 lb/d/p) and miscellaneous, ranging from food containers, filters, sanitary supplies, etc. (0.74 lb/d/p) contributes approximately \$89 to the hourly labor rate, based on a transportation cost of \$150/lb..

3. The principal cost item, aside from the cost of special training, is orbital housing and its maintenance. Even for an operational period of one year, its contribution to the hourly labor rate is between \$350 and \$850 for crew sizes between 50 and 20 persons.

4. If the cost of ground and orbital training of the individual is taken into account, the hourly labor cost varies within wide limits, being now strongly dependent upon the individual's orbital duty periods during the total period of a given orbital operation.

5. Unless the cost of special training can be kept at a level of \$50,000 to \$60,000 per person per year (for a two-year period), it is of great economic importance to maintain long orbital periods of duty (at least 90 days); or, if this meets with difficulties from the standpoint of work efficiency, to assure at least three tours of duty ($N_D = 3$) of 30 days per individual for orbital projects ranging from 180 to 720 days.

The overall orbital labor cost is a function of the various items discussed above and of the period of the orbital operation. Assume, for instance, that a 100 day orbital operation is planned, involving a crew of 30 persons, which is not rotated, but stays up for 100 days. Then the hourly labor rate becomes, not counting special training,

Cost of living (Fig. 18)	89 \$/hr
Personnel transportation (Fig. 18, $T_{op}/T_D = 1$)	24.5 \$/hr
Housing (Fig. 19, $\overline{N_P} = 30$)	2000 \$/hr
Hourly labor rate without special training	2113.5 \$/hr

resulting in a cost of 2113.5 · 800 · 30 = \$50.724.000. At the cost of special training specified in Tab. 3, the amount of 30 · 2 · 400,000 = \$24,000,000 must be added, yielding a total of \$74,724,000, or an overall hourly labor rate of \$3110.

5. Operational Considerations and Probability of Success of Orbital Delivery and Establishing Orbital Installations

5.1 Criteria

In principle, any ELV can amass any amount of weight in orbit, given a sufficient number of successful launchings. In practice, the establishment of an orbital installation whose weight or volume exceeds the payload weight or volume capability of a single given ELV affects the cost of establishing the installation through the following parameters:

(1-a) number of deliveries required,

(1-b) probability of successful delivery,

(1-c) probability of successful orbital mating and/or fueling.

Tab. 4 relates these parameters to six criteria, grouped in three categories. The number of launchings affects ground operation and ELV procurement cost, especially if the vehicles are expendable. The reliability of the overall operation determines also the procurement cost of modules in excess of those basically needed, to replace losses during delivery failures and failures during orbital operation. Finally, level of effort, duration and cost of orbital labor determine essentially the cost of the orbital operation during the establishment phase.

Tab. 4 <u>CRITERIA AFFECTING THE COST OF ESTABLISHING AN ORBITAL INSTALLATION</u>

Parameter	Criterion					
	No. of launchings	Procurement Volume		Orbital Operation		
		ELV	Modules	Level	Duration	Labor Cost
Number of orbit deliveries	*	*		*	*	*
Probability of successful delivery	*	*	*		*	*
Probability of successful orbital mating and/or fueling	*	*	*	*	*	*

5.2 Probability of Successful Establishment of Orbital Installations

The total requirement on the transportation system is determined by the probability of success desired for the delivery operation: in addition, by the orbital operation associated with the payload weight delivered and the desired level of its probability of success.

First, it is assumed that the payload packages are modules of a larger system which is assembled in orbit by mating these modules. We consider two cases:

Case A: All modules delivered are mated. Failure to mate one module to several modules already mated is assumed to lead to the loss of the two modules concerned, but not of the other modules. Thus, if module II of a I-II complex fails to be mated with module III, both modules II and III are assumed to be made unsuitable, but not module I. Module II must be separated from I and two new modules II and III delivered and mated with each other and with I.

Case B: The delivered modules are mated to individual complexes of 3, 4 or 5 modules each. In case of failure to mate, the same rules apply as in case A.

Case A and B are identical where 3, 4 or 5 modules are concerned. They are different for larger number of modules. Case A applies primarily to the establishment of large space stations, case B to lunar or planetary space vehicles.

The probability of success in establishing an orbital installation is thus determined by the cumulative probability of delivery of a number of modules, P_D^*, multiplied by the cumulative probability of mating a given number of modules, P_M^*. The probability of n or more successes in $n_D = n + j$ deliveries is

$$P_D^* = \sum_{n_D = n}^{n+j} A_j \, P_D^{\,n} (1 - P_D)^j \tag{18}$$

where P_D is given by Eq. (1a) and A_j follows from Tab. 5. Thus, for 3 or more

Tab. 5 COEFFICIENT A_j IN EQ. (18)

	A_j														
j	n=1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
0	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1
1	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
2	1	3	6	10	15	21	28	36	45	55	66	78	91	105	120
3	1	4	10	20	35	56	84	120	165	220	286	364	455	560	680
4	1	5	15	35	70	126	210	330	495	715	1001	1365	1820	2380	3060
5	1	6	21	56	126	252	462	792	1287	2002	3003	4368	6188	8568	11,628

successful deliveries out of 5 attempts, i.e. $n_D = 3 + 2$, it is

$$P_D^* = P_D^3 + 3\,P_D^3\,(1 - P_D) + 6\,P_D^3\,(1 - P_D)^2 \qquad (19)$$

and so forth.

The probability of m or more successful matings of m + 1 modules in m to m + k attempts under the ground rules specified for case A above is given by the following equations.

$$k = 0 \quad P^*_{M,\,m} = P_M^m$$

$$k = 1 \qquad = P_M^m \left[1 + (1 - P_M)\right] \qquad (20)$$

$$k = 2 \qquad = P_M^m \left[1 + \sum_{k=1}^{2} (1 - P_M)^k\right] + (m-1)\,P_M^m\,P_M\,P_d\,(1 - P_M) \qquad (21)$$

$$k = 3 \qquad = P_M^m \left[1 + \sum_{k=1}^{3} (1 - P_M)^k\right] + (m-1)\,P_M^m\,P_M\,P_d$$
$$(1 - P_M)\left[1 + (1 - P_M)\right] + (m-1)\,P_M^m\,P_M\,P_d\,(1 - P_M)^2 \qquad (22)$$

$$k = 4 \qquad = P_M^m \left[1 + \sum_{k=1}^{4} (1 - P_M)^k\right] + (m-1)\,P_M^m\,P_M\,P_d\,(1 - P_M)$$
$$\left[1 + \sum_{a=1}^{2} (1 - P_M)^a\right] + (m-1)\,P_M^m\,P_M\,P_d\,(1 - P_M)^2$$
$$\left[1 + (1 - P_M)\right] + (m-1)\,P_M^m\,P_M\,P_d\,(1 - P_M)^3$$
$$+ 2\,(m-1)\,P_M^m\,P_M^2\,P_d^2\,(1 - P_M)^2 \qquad (23)$$

$$k = 5 \qquad = P_M^{\ m}\left[1+\sum_{k=1}^{5}(1-P_M)^k\right] + (m-1)\,P_M^{\ m}P_M P_d(1-P_M)$$

$$\left[1+\sum_{a=1}^{3}(1-P_M)^a\right] + (m-1)\,P_M^{\ m}P_M P_d(1-P_M)^2$$

$$\left[1+\sum_{a=1}^{2}(1-P_M)^a\right] + (m-1)\,P_M^{\ m}P_M P_d(1-P_M)^3$$

$$\left[1+(1-P_M)\right] + (m-1)\,P_M^{\ m}P_M P_d(1-P_M)^4$$

$$+\,2(m-1)\,P_M^{\ m}P_M^{\ 2}P_d^{\ 2}(1-P_M)^2\left[1+(1-P_M)\right]$$

$$+\,2(m-1)\,P_M^{\ m}P_M^{\ 2}P_d^{\ 2}(1-P_M)^3 \qquad (24)$$

where P_M is the probability of mating successfully two modules and P_d the probability of successfully demating a damaged module from a module aggregate. It is assumed in the above equations that P_M and P_d are the same for all modules or mating processes. The number of modules which must be delivered into orbit is always $(m+1) + (k+2)$. This number, then, determines the possible number of successful launches required. Thus, if $m = 3$, $k = 2$, preparations for the establishment of this 4-module orbital installation must plan for eight deliveries. If one delivery failure is included, a total of nine launch vehicles (if non-reusable) and of nine modules (if interchangeable) would have to be procured to attain the associated overall success probability.

$$P^* = P_D^* P_M^* = \sum_{8}^{9} A_1 P_D^{8} (1 - P_D) \left\{ P_M^{3} \left[1 + \sum_{k=1}^{2} (1 - P_M)^k \right] \right.$$

$$\left. + 2 P_M^{2} P_M P_d (1 - P_M) \right\} \qquad (25)$$

In case B, Eqs. (18) through (24) are also applicable, but m is restricted to 5 or less and the fact that the process of establishing these installations is to be repeated, say p times, must be taken into account. Thus, in the example leading to Eq. (26), assume that p = 3 orbital installations of m + 1 = 4 modules each would have to be established. Then the overall success probability is

$$P_p^* = (P_D^* P_M)^p \qquad (26)$$

The number of modules to be procured may have to be larger in this case than in case A if they are not interchangeable.

A third case is also considered:

Case C: An orbital installation is to be supplied with fuel or other necessities. Failure to fuel does not destroy the module to be fueled and, therefore, requires only delivery of another supply vehicle (tanker) rather than two additional deliveries in the cases A and B.

For case C, Eq. (18) applies also to the orbital operation. The probability of s or more successes in p = s + q attempts to fuel or service the installation in any other manner is

$$P_S^* = \sum_{p=s}^{s+q} A_q P_S^{s} (1 - P_S)^q \qquad (27)$$

where A is found from Tab. 5 for $q = j$ and $n = s$. The probability of success of the entire operation is then in case C

$$P^* = P_D^* P_S^* \tag{28}$$

Probabilities P_D^*, P_S^* and P_M^* are shown in Figs. 20 through 23 for relevant ranges of individual probabilities P_D, P_S and P_M.

6. Comparison of Operational and Economic Aspects of Establishing Orbital Installations with a Large Number of Saturn V ELV Versus a Smaller Number of Post-Saturn ELV

6.1 Approach

The preceding Sections have laid the foundation for a comparison of the alternatives (1) and (2) presented at the end of Sect. 1. A comparison of two vehicles which do not exist, under operational conditions which have not yet been experienced, performing loosely defined tasks under economic conditions 12 years in the future is necessarily uncertain. Moreover, no exhaustive treatment of the subject is claimed in the frame work of this paper. However, if the two cases are treated consistently, the resulting trends should nevertheless be of significance for future planning.

It is attempted to show that not only transport cost effectiveness is involved; but that the associated cost of payload procurement and of orbital operations also plays an important role.

The technique of comparison is illustrated in an example, listed in some detail in Tab. 6. The task is to establish an orbital installation which consists of four complexes of 10^6 lb each. Saturn V, given a useful payload of 250,000 lb, is compared with a chemical Post-Saturn vehicle of 10^6 lb useful payload. The year selected is 1975, assumed in this paper (as an example, not a prediction) to be the first operational year of "the" Post-Saturn vehicle. This year puts the Post-Saturn, therefore, in a particularly unfavorable position reliability wise. The nevertheless comparatively high reliability resulting in a probability of successful delivery of $P_D = 0.75$ is justified on the basis that a 1-stage-to-orbit vehicle (no auxiliary systems jettisoned), a 10-vehicle test

Tab. 6 COMPARISON OF ECONOMIC AND OPERATIONAL ASPECTS OF ESTABLISHING FOUR COMPLEXES @ 10^6 lb WEIGHT IN ORBIT IN 1975 WITH SATURN V AND POST-SATURN

	Saturn V									Post-Saturn ELV		
Line Col.	1	2	3	4	5	6	7	8	9	10	11	12
1 n	4	4	4	6	6	6	8	8	8	4	4	4
2 j	0	1	2	0	1	2	0	1	2	0	1	2
3 $m = 3$: k	0	0	0	1	1	1	2	2	2	No mating req'd		
4 N_L	4	5	6	7	8	9	8	9	10	4	5	6
5 N_{Mod}	4	5	6	7	8	9	8	9	10	4	5	6
6 P_D^*	.498	.417	.9445	.351	.688	.877	.295	.493	.765	.316	.632	.829
7 P_M^* ($P_M = .95$; $P_d = .99$)	.855	.855	.855	.90	.90	.90	.983	.983	.983	--	--	--
8 $P^* = P_D^* \cdot P_M^*$	.425	.699	.806	.316	.619	.79	.29	.484	.751	.316	.632	.829
9 h	0	0	0	0	0	0	0	0	0	--	--	--
10 $N_L^* = 4\ N_L$	16	20	24	28	32	36	40	44	48	--	--	--
11 $N_{Mod}^* = 4\ N_{Mod}$	16	20	24	28	32	36	40	44	48	--	--	--
12 $P_p^* = (P^*)^4$	.033	.239	.422	.01	.147	.389	.007	.055	.318	--	--	--
13 h	1	1	1	1	1	1	1	1	1	--	--	--
14 $N_L^* = 5\ N_L$	20	24	28	32	36	40	41	48	52	--	--	--
15 $N_{Mod}^* = 5\ N_{Mod}$	20	24	28	32	36	40	44	48	52	--	--	--
16 P_p^* ($h = 1$)	.109	.527	.75	.037	.371	.716	.027	.168	.635			
17 Launch rate	1 per week (7 days) per launch pad									1/30d/launch pad		
18 Launch pads	4 + 1 spare = 5									3 + 1 spare (off-shore pltfms)		
19 Orbit. crew size (p)	$\overline{N}_p = 40$									$\overline{N}_p = 20$		
20 Launch period (d)	28	35	42	49	56	63	70	77	84	30	40	50
21 T_{op} (d)	38	45	52	59	66	73	80	87	94	40	50	60
22 Nomin. labor hrs.	11,800	14,400	16,600	18,900	21,100	23,400	25,600	27,800	30,100	6400	8000	9600
23 Cost of living (Fig. 18)	$89/hr	89	89	89	89	89	89	89	89	89	89	89
24 Housing ($\overline{N}_p = 40$)	$4200/hr	3450	3000	2630	2350	2140	1940	1790	1650	No housing req'd		
25 Personnel Transp.	40 (persons) · 200 (lb/p) · 100 ($/lb) = $800,000									$61/hr	49	41
26 Total Orb. Labor	$51.3M	51.7	52.2	52.1	52.4	52.9	52.8	53	53.2	$.96M	1.1	1.25
27 Total hourly rate	$4350	3590	3140	2760	2480	2260	2070	1908	1770	$150/hr	138	130
28 C^{**}	$160/lb Pld (Fig. 9, 1975); 160 · 250,000 = $40 M per ELV									$120/lb Pld; $120M/ELV		
29 Max. # ELV procured	16	20	24	28	32	36	40	44	48	4	5	6
30 Max. launch cost	$640M	800	960	1120	1280	1440	1600	1760	1920	$480M	600	720
31 Total cost w/o Pld.	$691.3M	851.7	1012.2	1064.3	1332.4	1492.9	1652.8	1813	1973.2	$481M	601.1	721.25
32 Max. payload wt. procured (10^6 lb)	4	5	6	7	8	9	10	11	12	4	5	6

NOTES FOR TAB. 6

1. See text for discussion of Table and explanations beyond those given below.

2. Line 6: $P_D = 0.84$ for Saturn V; $P_D = 0.75$ for Post-Saturn

3. Line 9: Process of mating 4 modules to obtain a 10^6 lb complex has to be repeated 4 times to obtain the required 4 complexes @ 10^6 lb; h = 0 means that no additional attempt to mate one more 10^6 lb complex is planned. This results in the overall number of launches, modules to be procured and probability of success, indicated in lines 10, 11, 12, respectively.

4. Line 13: One additional attempt to assemble a 10^6 lb complex is planned (h = 1), resulting in improved probability of success (line 16) of obtaining the required 4 complexes @ 10^6 lb, but at correspondingly higher procurement and launch cost.

5. Line 20: Designates the period within which all launches take place.

6. Line 21: The orbital operation is 10 days longer than the launch period to account for mating and/or checkout of the last module (Saturn V) or complex (Post-Saturn) which may arrive at the last day of the launch period.

7. Line 22: Based on 8 labor hours per day of T_{op} on the number of days of T_{op} and $\overline{N_P}$ (line 19).

8. Line 24: In the case of Post-Saturn, the orbital crew is expected to live in the complex for the duration of the period T_{op} of orbital operation

9. Lines 26, 27, 30: The cost of special training of the orbital crew (cf. Sect. 4.3) is <u>not</u> included. (M = million dollars)

10. Line 28: Cost effectiveness for Post-Saturn is derived from $160 \cdot 0.75 = 120$; where \$160/lb is given in Fig. 13 for the expendable version on the basis that its reliability is about 0.75. Since the effect of reliability has been considered here separately (lines 6 & 8), the cost effectiveness figure has been reduced to a value corresponding to 100% reliability. The cost figures in Fig. 13 account for the effect of reliability statistically over large numbers of launchings, whereas the values in lines 6 and 8 refer to success probability for the given, limited, number of delivery attempts.

11. Line 29: The procurement figures for Post-Saturn refer to expendable version, but are unlikely to be lower for the expendable version, in view of the short launch period (for which the vehicles have to be readied in advance) and in view of the low reliability (1975 being postulated as the first operational year in this example).

program, and careful, advanced quality control and checkout methods have been assumed.

For Saturn V the task amounts to transporting 16 modules into orbit and mating them to 4 10^6 lb-complexes. Lines 1 through 8 determine, for the assembly of one 10^6 lb complex, the overall probability of successful delivery (P_D^*), based on $P_D = 0.84$; the probability of successfully accomplishing four times three matings (P_M^*), based on $P_M = 0.95$, $P_d = 0.99$ (Sec. 5.2); and the overall probability of success P^*. The assumptions specified for orbital mating (rather than fueling) in Sect. 5.2 are used. Three alternatives are considered. First, in columns 1 through 3, no extra mating attempt beyond the minimum of 3 matings is planned ($k = 0$); whereas the number of extra delivery attempts is increased from zero ($j = 0$) to two ($j = 2$). The delivery probability grows, therefore, according to Eq. (18), for $n = 4$ and $j = 0, 1, 2$, i.e. 4 successful deliveries in 4, 5 and 6 delivery attempts. In practice, when 6 deliveries are planned, the first four deliveries may be successful; in which case the fifth and sixth ELV and payload would be available, in case of a mating failure. However, this additional mating capability is not part of the procurement and launch plan represented by col. 1 through 3, which merely aims at maximizing the probability that the four required modules will actually be delivered. In col. 4 through 6, an additional mating attempt is specifically planned ($k = 1$). This means that the plan must provide for a minimum of 6 launches ($n = 6$) under the assumption made in Par. 5.3 that failure to mate renders the two modules involved unsuitable, against which j is again varied from 0 to 2 to increase the confidence level of successful delivery of 6 modules.

In col. 7 through 9, finally, k = 2 and j = 0, 1, 2. Because P_M is larger than P_D, increasing j is more effective than increasing k. Thus, the highest confidence level is obtained with j = 2, but k = 0 (col. 3). Trying to accumulate excess modules in orbit (k = 0) in case they are needed there degrades the overall probability of success unless P_D is higher.

In lines 9 through 12 the effect is shown of carrying out 4 times each of the 9 alternatives for establishing one 10^6 lb complex (lines 1 through 8), on the procurement requirements and on the overall probability, under the assumption that no additional attempt to assemble a 10^6 lb-complex is planned (h = 0). If one attempt is planned (h = 1), the figures in lines 13 through 16 are obtained. Lines 17 through 30 estimate the cost of delivery and of orbital labor. The procurement requirements for h = 0 are used. Instead of attaching a dollar figure to the payload modules, they are compared on a weight basis, since it is plausible that their specific cost ($/lb) is comparable for Saturn V and Post-Saturn. In regard to this weight and the associated number of Saturn V payloads it should be pointed out that this number represents the maximum number of interchangeable modules which could possibly be used. Actually, to procure one module for each delivery failure which could possibly occur and two modules for each mating failure which could possibly occur is excessively cautious and might be justified only if the procurement lead times for the modules is much longer than the launch period and if the importance of timely execution of this operation is so crucial that it must not be endangered by lack of an adequate module supply, however remote the probability. Since the probability of occurrence of every conceivable module-damaging failure which could possibly occur, is almost zero, the probability that these many modules will be needed is likewise zero; in other words, the risk to the success of the overall operation, entertained by reducing the procurement of modules by some 10 to 20 percent is very small. The same applies to the procurement of Saturn V vehicles, since it is most unlikely that the maximum number of launch failures (one or two, respectively, during the assembly of each 10^6 lb complex) actually will occur. Again this statement is based on the assumption that all Saturn V vehicles and their payload interfaces are alike and, therefore, freely interchangeable. If all 16 modules are significantly different (in the sense that a change to convert one module into another (given this is feasible) would require significantly more time than the planned launch period), then a significantly larger number of modules than indicated in line 31 must be procured to be consistent with the overall probability of success. The economic importance of having, in a planetary or lunar ship or in a space station, as many modules interchangeable as possible is so apparent that it will strongly influence the design philosophy in this direction (especially, since interchangeability of modules of manned planetary ships is also of considerable practical importance in case of troubles en route). However, practice shows that this goal is never reached completely. Therefore, it can be expected that the majority, but not all of the modules will be interchangeable. Since this tends to raise the "safe" procurement level, compared to all-out interchangeability, the numbers given in line 31 could possibly have to be met.

In any case, for comparable success probability the procurement cost when using Saturn V is considerably higher than for Post-Saturn. Although the individual delivery reliability of the latter is a good deal inferior than that of Saturn V, the confidence level (.829) attained with $6 \cdot 10^6$ lb payload procurement (j = 2, k = 0) is significantly higher than that attainable with Saturn V for the same condition (.422, line 12, col. 3). Conversely, for a success probability of at least .75 (line 16, col. 3), $7 \cdot 10^6$ lb of payload would have to be procured. The reason for this is, of course, that no orbital mating is required. A maximum amount of preparation is done on the ground where it can be done more efficiently and far less expensively. The launch costs are likewise higher for Saturn V, since for each Post-Saturn launch at least 4 Saturn V launches are required, whereas the launch cost of an O_2/H_2 Post-Saturn ELV of the payload capability envisioned here appears to be only about 3 times as high as that of a Saturn V.

The example indicates the following:

1. The best alternative available to Saturn V, in terms of competitiveness with Post-Saturn would be n = 4, j = 2, k = 0, h = 1 (col. 3, lines 14 through 16).

2. This case compares with Post-Saturn, n = 4, j = 2 as follows:

   overall success probability: .75 vs. .829

   max. number of launches: 28 vs. 6

   max. launch cost (expendable ELV): \$1120M vs. \$720M

   max. payload wt. procurement: 10^7 lb vs. 10^6 lb.

3. If this payload is inexpensive (e.g. H_2), the last point is negligible. In this case, however, no housing would be available for the orbital crew of Post-Saturn, which would bring the orbital labor cost

roughly to the same level as listed for Saturn V. Thus, if inexpensive payload is hauled, the economic superiority would be based primarily on its higher cost effectiveness, resulting in a saving of the order of $400M for the entire operation. The economic superiority of Post-Saturn is not raised significantly if the cost of special training of the orbital crew is taken into consideration. At the level specified in Tab. 3, this cost is $16M higher for Saturn V, based on line 19 of Tab. 6.

4. If the payload is moderately expensive, say, $300/lb, the economic disadvantage of Saturn V is emphasized further, because, for reasons of mission success confidence, 10^6 lb more payload weight must be procured, adding $300M to the $400M in higher vehicle procurement and launch cost. Moreover, in this case, the payload is likely to possess accommodations for personnel (e.g. flight crew). Their temporary use by the orbital operations crew is likely to be more feasible in the Post-Saturn case where no mating, only checkout of the complexes is involved. Therefore, it is likely that no orbital housing will be required for Post-Saturn, adding another $50 to 65 million and bringing the total cost difference for this comparatively small orbital operation to the order of $750 million in favor of Post-Saturn V. It is important to note that this advantage is due to about 50 percent to lower transportation cost, the other half being derived from lower payload weight procurement and simplification of the asso-

ciated orbital operations. Even if the savings were only half as large, they would still be very significant.

In conclusion, it should be pointed out that the relative position of Saturn V can be improved, if a mating technique is used in which failure to mate does not result in the destruction (or mission unfitness) of both modules concerned, but of only one, preferably the one to be attached (so as to eliminate the need for demating a module). In that case, a mating failure results in a requirement for one, rather than two additional deliveries. This raises the overall probability of success significantly, even if the probability of mating success proper (P_M) is not raised.

This condition can certainly be assumed to exist in the case of fueling rather than mating. This case is, of course, predicated on the specification, not made previously, that the 10^6 lb complex is a vehicle. It must further be assumed that the vehicle uses chemical propellants (O_2/H_2 or denser), since the size of a 10^6 lb nuclear-powered hydrogen carrying vehicle is too big for the presently specified payload volume of Saturn V (about 52,000 ft^3) limited by facility limitations and design criteria. However, O_2/H_2, assuming a mean density of 24 lb/ft^3, requires only about 38,000 ft^3 for a propellant load of 900,000 lb. Thus, the complete vehicle can be carried aloft, partly fueled and subsequently fueled by tankers. To account for problems connected with mounting the entire vehicle in the nose section, and taking into consideration insulation weights for the tanker atop Saturn V, it is assumed that in this case, a minimum of 5 launches is required, 1 for the vehicle and 4 tankers.

6.2 Long Range Transportation Requirements

Although the example in Sect. 6.1 indicates an impressive potential cost superiority of Post-Saturn, the difference nevertheless is small compared to the development cost of such a vehicle which is expected to lie between 5 and 10 billion dollars. It is, therefore, necessary to establish a justification on the basis of sustained long-range transportation requirements.

As basis for this comparison has been selected the Case A transportation level (Fig. 7) which calls for an average successful orbital delivery of 6 million lb in 1975/79 and of 12 million lb in 1980/84. The delivery costs are based on the cost effectiveness values and associated varying success probabilities shown in Figs. 9 and 13. The orbital labor cost data are based on the cost analysis presented above and specifically on the resulting hourly rates, plotted in Fig. 24 for the conditions noted on the graph. For Saturn V a net payload of 250,000 lb is assumed. In computing the orbital labor cost, the orbital personnel required in the second 5-year period has been increased by 50% for Saturn V (corresponding to a reduction of the nominal period of duty by 50%) and by 33 percent for Post-Saturn. The results are shown in Fig. 25 for the expendable and the recoverable version of the Post-Saturn ELV. The cost figures shown are cumulative with progressing time. The upper three curves show both, delivery and orbital labor cost. The lower two curves show the orbital labor

Tab. 7 RELIABILITIES OBTAINED WITH SATURN V FOR THE SAME TASK AS IN TAB. 6, EXCEPT THAT MATING IS REPLACED BY LAUNCHING COMPLETE VEHICLE IN ORBIT AND FUELING

		Saturn V			
Line	Col.	1	2	3	4
1	n	1	1	1	1
2	j	0	0	1	1
3	N_L	1	1	2	2
4	N_V	1	1	2	2
5	$(P_D = .84)\ P^*_D$	.84	.84	.9745	.9745
6	Tankers s	4	4	4	4
7	q	0	1	0	1
8	N_L	4	5	4	5
9	N_T	4	5	4	5
10	$(P_s = .95)\ P^*_S$	.814	.977	.814	.977
11	N_L for 1 compl. veh.	5	6	6	7
12	$P^* = P^*_D\ P^*_S$	.684	.82	.792	.952
13	h	0	0	0	0
14	$P^*_p = (P^*)^4$	.219	.452	.393	.835
15	N^*_L	20	24	24	28

cost. The cost effect of the two approaches on payload is not included, since it cannot be assessed in this general form. However, the qualitative trend established in this respect by the example in Sect. 6.1 should apply also to the general case. Fig. 25 shows:

1. The cost superiority of the Post-Saturn appears to be based in the first place on size, in the second place on the more gradually developing effect of reusability, thirdly on savings in orbital labor cost which are the least certain factor, but are believed to be treated here conservatively, i.e. reducing the difference between Saturn V and Post-Saturn more than might be the case in a more specific 10-year orbital delivery program.

2. It is, therefore, not necessary that Post-Nova attains reusability from the start. It is more significant that its design and configurational characteristics permit the development to a reusable mode of operation in the course of approximately the first five years of its operational life.

3. If the cost of developing Post-Saturn is taken as 6 billion dollars, then this investment should be amortized during the first ten years of its operational life, if the transport requirements develop as assumed in Fig. 25, even if no reusability is attained during this period. In case of reusability, amortization is indicated after about 8 years of operational life.

4. It is, therefore, important that the Post-Saturn configuration selected, has a low rate of obsolescence. This is assured if

the vehicle is characterized by:

(a) a shape which offers as few volume restrictions as possible to a payload weight of this magnitude

(b) highest possible operational simplicity and reliability

(c) advanced chemical engines (high-pressure O_2/H_2) and a design which permits the vehicle to be adapted to more advanced propulsion systems, as the state-of-the-art advances, specifically to the use of nuclear and airbreathing engines.

7. Parametric Performance Evaluation of Advanced Concepts

7.1 Introduction

The parametric evaluation of advanced vehicle concepts comprises an appraisal of the payload fractions to be expected by the different vehicle types as well as a discussion of the merits of basic configuration types.

Definitions

It is $\lambda + b + \Lambda = 1$ (29)

where λ gross payload fraction of the given stage

b wet inert weight fraction (burnout weight fraction minus λ) of the given stage

Λ useful propellant weight fraction of the given stage

and
$$\lambda = \frac{1}{x}\left(\frac{1}{\mu} - 1\right) + 1 = 1 - \frac{1}{x}\left(1 - \frac{1}{\mu}\right) = 1 - \frac{\Lambda}{x} \tag{30}$$

$$b = \frac{1 - x}{x} \frac{\mu - 1}{\mu} \tag{31}$$

$$\Lambda = \frac{\mu - 1}{\mu} \tag{32}$$

where x = mass fraction of the given stage

μ = mass ratio of the given stage

and
$$x = \frac{W_p}{W_b + W_p} \tag{33}$$

$$\mu = \exp\left(\frac{\Delta v_{id}}{g\, I_{sp}}\right) = \exp\left(\frac{\tau}{I_{sp}}\right) = 1 + \frac{W_p}{W_\lambda + W_b} \tag{34}$$

$$W_b = k_f F + k_p W_p + \ldots = W_p \left(\frac{1-x}{x}\right) \qquad (35)$$

where $k_f = W(F)/F$ the thrust-dependent portion of W_b, $k_p = W(P)/W_p$ the propellant-dependent portion of W_b. Other factors may be added for additional significant fractions of W_b. For the above equation, it is for given values of x and k_f

$$k_p = \frac{1-x}{x} - k_f \frac{F}{W_p} = \frac{W_b - k_f F}{W_p} \qquad (36)$$

or, alternatively,

$$k_f = \frac{W_p}{F} \left(\frac{1}{x} - 1 - k_p\right) \qquad (37)$$

$$x = \frac{W_p}{k_f F + W_p (1 + k_p)} \qquad (38)$$

Finally,

$$W_p = (\mu - 1)(W_\lambda + W_b) = \frac{(\mu - 1)(W_\lambda + k_f)}{1 - k_p (\mu - 1)} \qquad (39)$$

where Δv_{id} = ideal velocity of the given stage

W_p = useful propellant weight of the given stage

W_λ = payload weight of the given stage (useful + operational pld.)

W_b = wet inert weight of the given stage

and $\tau = \Delta v_{id/g}$ (40)

Burnout weight of stage, $W_B = W_b + W_\lambda$ (41)

Initial weight of stage, $W_A = W_B + W_p$ (42)

Partial derivatives pertaining to the payload fraction of the given stage or of a single-stage vehicle

$$\frac{\partial \lambda}{\partial \mu} = - \frac{1}{\mu^2} \frac{1}{x} \tag{43}$$

$$\frac{\partial \lambda}{\partial I_{sp}} = \frac{\tau}{I_{sp}^2} \frac{1}{\mu} \frac{1}{x} = \frac{\ln \mu}{\mu} \frac{1}{I_{sp}} \frac{1}{x} \tag{44}$$

$$= \frac{\tau}{I_{sp}^2} \left(\lambda + \frac{1}{x} - 1 \right)$$

$$\frac{\partial \lambda}{\partial x} = \frac{1}{x^2} \left(1 - \frac{1}{\mu}\right) = \frac{\Lambda}{x^2} = \frac{1 - \lambda}{x} \tag{45}$$

$$\frac{\partial \lambda}{\partial (\Delta v_{id})} = \frac{1}{g\, Isp} \frac{1}{\mu} \frac{1}{x} = - \frac{\ln \mu}{\Delta v_{id}} \frac{1}{\mu} \frac{1}{x} \tag{46}$$

Finally, if subscripts 1 and 2 designate the first and second stage, respectively, and subscript 12 designates values pertaining to the overall vehicle,

$$\lambda_{12} = \frac{W_{\lambda 2}}{W_{A1}} = \lambda_1 \lambda_2 \tag{47}$$

$$\mu_{12} = \frac{W_{B2}}{W_{A1}} = \mu_1 \mu_2 \tag{48}$$

$$b_1 = \frac{W_{b1}}{W_{A1}} = \text{weight jettisoned (from second stage)} \tag{49}$$

$$b_2 = \frac{W_{b2}}{W_{A2}} = \text{weight jettisoned (from third stage or payload stage)} \tag{50}$$

Partial derivatives pertaining to a two-stage vehicle,

$$\frac{\partial \lambda_{12}}{\partial I_{sp1}} = \lambda_1 + \lambda_2 \frac{\tau_1}{I_{sp1}^{\,2}} \left(\lambda_1 - \frac{1}{x_1} + 1\right) \tag{51}$$

$$\frac{\partial \lambda_{12}}{\partial I_{sp2}} = \lambda_2 + \lambda_1 \frac{\tau_2}{I_{sp2}^{\,2}} \left(\lambda_2 - \frac{1}{x_2} + 1\right) \tag{52}$$

$$\frac{\partial \lambda_{12}}{\partial x_1} = \lambda_1 + \lambda_2 \left(\frac{1 - \lambda_1}{x_1} \right) \tag{53}$$

$$\frac{\partial \lambda_{12}}{\partial x_2} = \lambda_2 + \lambda_1 \left(\frac{1 - \lambda_2}{x_2} \right) \tag{54}$$

7.2 Two-Stage O_2/H_2 Vehicles with Conventional and Advanced Engines

Fig. 27 shows characteristic weight curves for an advanced land or sea-platform launched vehicle with 10^6 lb payload and an ideal velocity capability of 30,000 ft/sec (non-rotating Earth) for the conditions specified on the graph. The propellant factor of St. 1 and St. 2 has been varied between 0.88 and 0.92. Post-Saturn preliminary design studies indicate for a conventional 2-stage vehicle, $x_1 \sim 0.88$, $x_2 \sim 9.99$. Small variations in I_{sp} have been included. The curves show that the maximum of λ_{12} can shift over a wide range of $\Delta v_{id1}/\Delta v_{id2}$ in response to comparatively moderate variations of I_{sp} and/or x. Generally, the trend is, as expected, that with increasing superiority of the second stage (in terms of I_{sp} and x) the $\lambda_{12,\ max}$ - staging velocity decreases. It is of interest to note, however, that an increase in the upper stage propellant factor, x_2, by five percent (from 0.88 to 0.92) is of greater importance than an increase in specific impulse, $I_{sp\ 2}$, by about five percent (from 430 to 450). This is shown by curves (1), (4) (5% $I_{sp\ 2}$ - increase), and (5), (5% x_2 - increase). Compared to curve (1), curve (4) raises λ_{12} from 0.0525 to 0.0575 and reduces the St. 1 ideal velocity from 13,200 to 12,000 ft/sec; whereas curve (5) shows an increase to λ_{12} = 0.0637 and reduces $\Delta v_{id,1}$ to 10,000 ft/sec. In the first stage too, an improvement in x_1 (curve (6)) pays off more than an improvement in $I_{sp\ 1}$ (curve (3)), although the effectiveness in increasing λ_{12} is less than for the same improvements in the second stage, while the effect on $\Delta v_{id,1}$ is, of course, the opposite, since a higher performance first stage should burn longer for maximum λ_{12} than a first stage of lower performance. The partial derivatives, Eqs. (44), (45) show that increase in specific impulse pays off increasingly with increasing τ, increasing λ, decreasing x and especially with decreasing I_{sp}; raising x pays off increasingly well with increasing λ and decreasing x.

The use of advanced O_2/H_2 engines makes itself felt in higher specific impulse, while the mass fractions remain stable. A variety of advanced O_2/H_2 engines has been proposed by leading engine development companies ever since the development of Centaur began. They are all characterized by higher chamber pressure (1500 - 2500 psi) and, in some cases involve novel chamber and nozzle configurations. Fig. 28 estimates the increase in payload fraction attainable with 2-stage ELV's by the higher mean specific impulses foreseen with advanced O_2/H_2 engines. The increases lie between 1/2 and 1% of the take-off weight. The increases in I_{sp} used in Fig. 28 are conservative. With successful development of altitude compensation which permits the use of high expansion ratios from the ground up, mean effective specific impulses up to about 440 sec for the entire ascent may be attainable. In such case, it is of interest to look at single-stage to orbit configurations.

7.3 Single-Stage-To-Orbit O_2/H_2 Vehicles Using Advanced High-Performance Engines

From the standpoint of operational simplicity and reliability, single-stage-to-orbit vehicles are particularly attractive, since the need for staging and for two separate recovery operations (first and second stage) is avoided; and compared to two-stage vehicles in tandam arrangement, the need for an air start of the second stage engines is also eliminated. Particular attention is, therefore, given the single-stage-to-orbit concept in the Post-Saturn study. The draw-backs of this concept are connected with the need to carry a comparatively greater structural weight into orbit and to have to recover a correspondingly larger weight. In consequence, launch weight and launch thrust level increases. It is, therefore, in this case even more important to take advantage of high-I_{sp} systems. In the case of chemical propulsion, this means advanced O_2/H_2 engines. Likewise, much emphasis has to be placed on a large propellant factor x. Although the greater size of the single-stage vehicle renders the attainment of a larger propellant factor more feasible, in principle, it is doubtful that it can be boosted to the extent necessary to match the combined propellant factor of a 2-stage vehicle. This factor is

$$x_{12} = \frac{1}{1 + \dfrac{1}{\dfrac{x_1}{1-x_1}\,\dfrac{W_{p2}}{W_b^{1}}}\;\dfrac{1}{\dfrac{x_2}{1-x_2}\,\dfrac{W_{p1}}{W_{b2}}}}$$

since W_{p2}/W_{b1} and W_{p1}/W_{b2} are large numbers, the second and third term in the denominator are quite small so that x_{12} comes close to unity. Thus, it is unlikely that the single-stage-to-orbit chemical Post-Saturn will attain payload fractions as high as the two-stage version. On the other hand, with proper design, the penalties in terms of higher take-off weight, thrust requirement and higher recovery weight can be made acceptable in exchange for the practical advantages of single-stage-to-orbit operations.

Fig. 29 shows the payload fractions attainable with single-stage-to-orbit vehicles as function of the mean effective specific impulse and the propellant factor X for two ideal velocities. The mean specific impulse for surface-to-orbit flights can be taken as 1.13 times the surface - I_{sp} in engines with fully effective altitude compensation and as 1.2 to 1.23 for engines with no altitude compensation, whose surface - I_{sp} is much lower. For high- p_c O_2/H_2 engines without altitude compensation, the mean effective I_{sp} should lie between 420 and 430 seconds and, with altitude compensation, between 430 and 440 seconds. Fig. 29 shows that for this range of I_{sp} - values, λ for Δv_{id} = 30,000 ft/sec will lie between 0.016 and 0.023, if x = 0.90 can be achieved, compared to $0.05 \leq \lambda \leq 0.064$ for the two-stage vehicle (using curves (3), (4) and (5) in Fig.27). The sensitivity of λ against changes in x are seen to be quite high. The partial $\partial\lambda/\partial x$ has values around 1.0 (1.08 to 1.01), whereas $\partial\lambda/Isp$ lies between $5 \cdot 10^{-4}$ and $7 \cdot 10^{-4}$; that is, it takes an I_{sp} increase of 2 to 14 sec to compensate for an increase in x by 0.01.

7.4 Chemonuclear and Nuclear Launch Vehicles

Fig. 30 shows the payload fractions attainable with a chemonuclear launch vehicle of the HELIOS design (ref. 6) with chemical rocket lift-off stage for a range of I_{sp} and x values expected to bracket actually attainable values. From the slope shown by all curves at $\Delta v_{id,1} = 1000$ ft/sec, it is apparent that the highest payload fraction is attained at $\Delta v_{id,1} = 0$, i.e. no chemical rocket lift-off stage at all. This fact determined the basic philosophy underlying the HELIOS design, namely, to restrict the chemical portion of the flight as much as possible, consistent with requirements for payload protection against scattered neutron radiation in the denser atmosphere.

However, it is not sufficient to provide a small chemical lift-off stage of fixed performance. The comparatively steep gradient of all curves indicates the desirability of varying the flight performance of the chemical lift-off stage, consistent with individual payload radiation sensitivities, so as to maximize the payload fraction for individual flights as much as practicable. This flexibility is designed into the HELIOS concept by the parallel arrangement of chemical rocket stage with the nuclear stage and by the use of the same fuel in both stages, thereby rendering it comparatively easy to provice, within wise limits, for each mission on optimum compromise between performance (high payload fraction) and protection of the payload against scattered nuclear radiation.

In comparing the HELIOS payload fractions with those attainable with an all-nuclear single-stage-to-orbit (Fig. 31) it is seen that the latter values are only slightly higher, in the order of 1.5 - 2% of the take-off weight. For example, in Fig. 30 for HELIOS, curve (4) at $\Delta v_{id,1} = 6000$, $\Delta v_{id,2} =$ 24,000 ft/sec shows a payload fraction of 0.181. For a comparable all-nuclear ascent, the mean specific impulse is no higher than about 820, yielding with x = 0.85, a payload fraction of 0.201; an increase of 2%. Although it is not to be implied that 2% is a negligible gain in payload fraction, it can nevertheless be stated that the penalty incurred by the use of a small chemical lift-off stage, as in HELIOS, is not severe. Certainly, the loss in payload could be more excessive if radiation sensitive payload would have to be protected against scattered neutron radiation during the first 100,000 ft of ascent through the atmosphere.

On the other hand, HELIOS is so designed that the chemical lift-off stage can be omitted and full advantage be taken of an all-nuclear ascent, whenever the type of payload (e.g. LH_2) permits this approach.

Because of its high payload fraction a nuclear ELV is characterized by particular low initial weight. This fact favors the nuclear ELV in combination with the use of airbreathing boosters. An estimate of the payload fractions attainable with a HELIOS using an airbreathing (air/H_2) stage (Fig. 32) instead of an O_2/H_2 rocket lift-off stage indicate a slight superiority over the all-nuclear single stage ELV. This is because the increase in λ_2, due to the reduction of the ideal velocity required by the nuclear stage, is not quite wiped out by the λ_1 of the airbreathing stage ($0.8 \leq \lambda_1 \leq 0.9$), resulting in a λ_{12} for the entire system which offers prospects of being superior to that of the all-nuclear ELV (ref. 7). It is worthwhile to note, in Fig. 32, the sensitivity of the payload fraction to changes in the mass fraction of the air-breathing lift-off stage.

Considering the last availability of large nuclear engines (compared to the availability of O_2/H_2 engines, even of the advanced type) and the difficulties in clustering nuclear high-thrust engines, nuclear and chemonuclear ELV's with minimum-type non-nuclear lift-off stage are not likely to attain, in the early seventies, orbital payloads of the order of 10^6 lb.

If the ELV is to take advantage at all, by the middle or late seventies, of the superior specific impulse of nuclear engines at orbital payloads of the order of 10^6 lb or higher, then the chemical stage must be enlarged and a vehicle must be developed in which the chemical stage carries a larger share of the energy burden due to its superior thrust capability which is bought at the expense of mean specific impulse. Moreover, the nuclear propulsion system must consist of clustered engines.

At this point it is worthwhile to consider seriously the mating of advanced O_2/H_2 engine technology with nuclear technology both of which will mature considerably in the course of the sixties and the early seventies. It is further assumed that airbreathing boosters of adequate thrust level and, if possible, capable of supersonic combustion would be highly desirable, especially for a nuclear ELV. The time and cost required for such development, however, makes the operational date for the aero-nuclear ELV very uncertain. As far as the nuclear engines are concerned, single, multi-million pound thrust engines with solid core reactors will not be available within the next 10 to 15 years. Therefore, designs must be considered which permit the use of smaller nuclear engines, in an open cluster.* Finally, it is felt highly desirable to retain the

* An open cluster is defined as a cluster in which the nuclear engines are 30 ft or more apart.

advantages connected with a parellel arrangement of chemical and nuclear engines because of the many practical advantages connected with the nuclear engine start. At the same time, the vehicle can be developed as a 2-stage or a 1-stage to orbit ELV.

The adaptation of a recoverable chemonuclear ELV of the HELIOS design concept has, therefore, the following features:

One-stage or two-stage to orbit

Advanced O_2/H_2 engines

Chemical engines parallel with nuclear engines

Open cluster of nuclear engines at periphery; chemical engines in center

Vehicle recoverable from orbit and re-usable.

The orbital payload fraction attainable with a 2-stage version is shown in Fig. 33 to be intermediate between the advanced chemical 2-stage ELV (Fig. 28) and the 1-stage nuclear ELV (Fig. 31), as is to be expected. Also, as expected, it is somewhat inferior to the HELIOS version (Fig. 30), because a large share of the ascent energy is furnished by the chemical system. Thus, it appears possible, for example, to attain $\lambda = 0.1$ with 20,000/10,000 velocity distribution, $x - 0.92$ and $I_{sp} = 430$ (mean effective), yielding a 10^6 lb orbital payload ELV for a take-off weight of the order of 10^7 lb.

Figure 34 presents the conditions for a single-stage-to-orbit chemonuclear ELV. Here, the chemical stage burns up to a certain velocity, then shuts down, whereupon the nuclear stage carries the vehicle the rest of the way. Obviously, this is not necessarily the best way. An alternate mode is to start, from a certain altitude on, to blend the nuclear thrust in which cutting back the chemical thrust correspondingly. Subsequently, as the nuclear thrust/weight ratio increases, and as the centrifugal trajectory pressure grows, requiring less thrust/weight ratio in the first place, the chemical thrust is gradually reduced further and finally cut-off entirely.

For the first mentioned case of separate chemical and nuclear burning, the wet inert weight $W_{b,c}$ of the chemical module is not jettisoned, but carried along by the nuclear module. In terms of the initial weight $W_{A,n}$ of the nuclear stage, $W_{b,c}$ represents the fraction

$$\frac{W_{b.c}}{W_{A,n}} = \frac{W_{b,c}}{W_{A,c}} \; \frac{W_{A,c}}{W_{A,n}} = \frac{b_c}{c} \tag{55}$$

where $\lambda_c = W_{A,n}/W_{A,c}$ is the payload fraction of the chemical module.

Therewith the payload fraction λ_n of the nuclear stage is modified to

$$\lambda_n' = \lambda_n - \frac{bc}{\lambda c} \tag{56}$$

since the additional hardware carried along all the way reduced the payload of the nuclear stage in direct proportion (i.e. trade-off factor = 1 in this case). The apparent mass fraction of the nuclear stage is now

$$x_n' = \frac{W_{p.n}}{W_{p,n} + W_{b,n} + W_{b,n}} \quad \frac{\Lambda n}{\Lambda n + b_n + \frac{bc}{\Lambda c}} \tag{57}$$

whence one has for λ_n' also

$$\lambda_n' = 1 - \frac{1}{x_n'} \left(1 - \frac{1}{\mu n}\right) \tag{58}$$

or, conversely, solving Eq. (58) for x_n',

$$x'_n = \frac{1 - \frac{1}{\mu_n}}{1 - \lambda'_n} \tag{59}$$

Finally, the payload fraction of the entire ELV is given by

$$\lambda' = \lambda_c \lambda_n = \left[1 - \frac{1}{x_c}\left(1 - \frac{1}{\mu_c}\right)\right]\left[1 - \frac{1}{x'_n}\left(1 - \frac{1}{\mu_n}\right)\right] \tag{60}$$

where x_c is the mass fraction of the chemical module, x'_n the modified mass fraction of the nuclear module, μ_c the mass ratio of the chemical stage $\left[\mu_c = \exp(\Delta v_{id,c}/v_{e,c})\right]$ and μ_n the mass ratio of the nuclear stage

$\mu_n = \exp(\Delta v_{id,n}/v_{e,n})$

On the basis of these relations, Fig. 33 shows the payload fraction to be far more sensitive to changes in the $\Delta v_{id,c} / \Delta v_{id,n}$ velocity distribution than for the two-stage version, because of the influence of the wet inert weight of the chemical module vehicle tends to reduce drastically the effective mass fraction x'_n of the nuclear module (Graph (C)). The overall vehicle system is extremely sensitive to the mass fraction x_c of the chemical module (Graph (B)) and to the velocity distribution; but it is potentially superior to an all-chemical one-stage-to-orbit ELV and potentially competitive with the two-stage chemonuclear ELV.

8. ELV Configurations

Having surveyed parametrically the range of payload fractions which appears attainable with single and two-stage launch vehicles using a variety of propulsion modes, it remains to compare the relative merit of different vehicle configurations. These can be grouped into

Ballistic

Blunt body

Winged.

The ballistic configuration is defined here as a launch vehicle whose aspect ratio without payload is larger than one. The blunt body configuration is taken as one whose aspect ratio without payload is equal to, or less than, one.

Aside from being able to stand freely and being self-supporting the ELV must provide adequate frontal area to take large low-density payloads and it must be capable of re-entry and of reaching the surface in reusable condition. The latter two requirements have a particularly profound influence on the appraisal of Post-Saturn ELV configurations. The wide range of possible payload densities, coupled with the large payload weight, leads to enormous differences in payload size. This fact will represent a major problem area for any all-purpose Post-Saturn ELV configuration, as is illustrated in Fig. 35. The envelope of two single-stage-to-orbit ELV's of $24 \cdot 10^6$ lb take-off weight, using O_2/H_2 ($x = 0.914$; $W_{H_2} = 3.5 \cdot 10^6$ lb, $W_{O_2} = 16.5 \cdot 10^6$ lb) is shown, in case (a) as a blunt body, in case (b) as a ballistic configuration. Either case is shown with an interplanetary vehicle section as payload[1] and

1) Planetary vehicle of about 900,000 lb weight fully fueled, without the Earth escape booster used for hyperbolic injection. Since such vehicle cannot fill out the entire volume, its apparent mean density is low. In the present case the mean density for the given nose fairing envelope is between 2.5 and 3 lb/ft^3.

with a conical nose section which, from the apex on down, is assumed to be filled with 10^6 lb of LH_2 or of water. It is seen that with the blunt body configuration, even in the most extreme case, the overall vehicle height is comparable to the maximum value considered for Saturn V; whereas the ballistic configuration becomes some 530 ft tall. For land launched or sea-platform launched vehicles, such length means additional heavy investment in high-ceiling assembly and checkout buildings and facilities and in test and launch facilities. On the other hand, the blunt configuration cannot efficiently accomodate denser payloads of the same weight. The total volume of a characteristic cone (40° cone angle) of $1.57 \cdot 10^6$ ft^3 is approximately seven times as large as the volume required even by liquid hydrogen. Payload densities for lunar operations will lie between the density of LH_2 and that of concrete. It is apparent from Fig. 20 that the need for compatibility with a wide range of payload sizes will be a major factor in determining the Post-Saturn ELV configuration, if the ELV is to have an all-purpose payload-carrying capability. Body diameters between 90 and 160 ft are indicated. The problem is aggravated in the case of winged configurations.

Tab. 8 compares the re-entry and recovery characteristics of the three principal ELV configurations. Emphasis on keeping the re-entry and touchdown operations as simple and reliable as possible suggest the blunt body as the most attractive configuration.

Tab. 8 SURVEY OF RE-ENTRY AND RECOVERABILITY CHARACTERISTICS OF THE THREE PRINCIPAL ELV CONFIGURATIONS

ELV Configurations	Aspect Ratio	Re-Entry and Recoverability
Ballistic	1	Passive: Feasible only with very heavy structure, such as Aerojet's Sea Dragon. Water impact of aerodynamically stable vehicle. Active: (a) Attitude controlled re-entry. Aerodynamically stable configuration. Tank heat shielded. Aerodynamic stability and low W/A achieved by flaps flaring out at aft end. Touchdown by parachute or balloon. (b) By special provisions and in flight operations such as the inflatable drag body proposed by the ROOST concept of Douglas Aircraft Co. (ref. 8) or modifications thereof. Attitude controlled re-entry. Configuration is aerodynamically stable. Very low W/A attainable with inflatable drag body. Buoyancy attainable near surface.
Blunt Body	1	Has favorable shape for high-drag re-entry, large nose radius and low W/A. Attitude controlled re-entry. Vehicle must be aerodynamically stable. Touchdown by parachute and retro-rockets.
Winged	1	Low W/A controllable glide return from space. Vehicle structure must be strong enough to accept landing shocks in a direction which is normal to thrust direction assuming vertical take-off at launch. Horizontal take-off appears not practical at the payload sizes to be considered.

9. The NEXUS Earth Launch Vehicle Concept

The nucleus of a stable, economic, large-payload Earth-to-orbit logistics operation is an Earth launch vehicle (ELV) which satisfies the following requirements:

(a) low rate of obsolescence

(b) economy of operation

(c) high reliability

In order to meet condition (a), the vehicle design (including the propulsion system) must be sufficiently advanced to meet firm as well as potential requirements with no or few changes (operational capability and versatility). On the other hand, the design must be realizeable within the time period by which the transportation system as a whole (which includes more than just the vehicle, e.g. launch and tracking facilities, capability of orbital handling of large masses etc.) has to be operational. The development schedule of this overall transportation system, in turn, is imbedded in the national space development plan. No such plan is officially available as a guide. However, it is generally recognized that the seventies will see the development of a post-Apollo Lunar capability which is likely to require considerably larger payload weights than can conveniently and economically be furnished by the Apollo ELV Saturn C-5. Furthermore, the foreseeable state-of-the-art in the areas of propulsion, orbital operations, deep space vehicle operation, manned space vehicles and manned space flight in the late sixties and early seventies will principally meet the requirements for the first manned flights to Venus or Mars in the seventies. Comparatively favorable mission conditions exist in 1973 and 1975 to Venus and Mars--from the standpoint of mission energy and solar activity--which will not recur before the 1984/86 period. Although these two factors provide an incentive to consider the possibility of planning the first manned planet flight to take place during the first half of the seventies, they are not so dominant as to permit no alternative other than to disregard the 1977 through 1983 period and make a choice between the 1973 to 1975 period and postponement of first manned planetary flights to 1984. Flights in 1977 and 1979 are quite feasible, if a Post-Saturn ELV is operational and if a nuclear engine of 250,000 lb thrust is available to power the planetary vehicle. To illustrate this fact, Tab. 9 lists favorable launch dates to Mars in the 1973 through 1984 period for a 430-day roundtrip mission and shows the associated mass ratios. It is, therefore, reasonable to assume that a Post-Saturn Earth-to-orbit transportation system should be operational no later than 1977-1979, in which case the development of the ELV which is to be its nucleus, must be completed in the 1976/1978 period. This leave approximately 11 to 13 years for the actual development work and determines the frame of reference within which the selection of the more advanced components and subsystems must be made. This is especially true for the propulsion system which promises to be the principal pacesetter. The boldest strides will have to be made in the propulsion area where the biggest performance gains can be achieved.

Requirement (b), economy of operation, calls for reusability at moderate costs for recovery and refurbishing. It also calls for operational flexibility.

Tab. 9 COMPARISON OF A 430-DAY MARS ROUNDTRIP 1973 THROUGH 1984 T_1 = 180 d; T_2 = 200 d; T_{cpt} = 50 d

Dep. Earth	$v^*_{\infty 1}$	$v^*_{\infty 2}$	R_P	Arr. Mars	Dep. Mars	$v^*_{\infty 3}$	$v^*_{\infty 4}$	R_P	μ_1 (F/W_0 = .3)	μ_2 (F/W_1 = 1.2)	μ_3 (F/W_0 = .2)	$\mu_1\mu_2\mu_3$
3-9-73	.33	.28	.84	9-5-73	10-25-73	.29	.26	.87	2.4	2.15	2.2	11.35
4-28-73	.27	.21	.92	10-25-73	12-14-73	.36	.38	.76	2.07	1.73	2.82	10.1
7-17-75	.28	.24	.94	1-13-76	3-3-76	.38	.52	.65	2.12	1.9	3.03	12.2
8-15-77	.36	.33	90	2-11-78	4-2-78	.35	.53	.66	2.77	2 53	2.73	11.6
10-14-79	.27	.36	.95	4-1-80	5-21-80	.29	.52	.61	2.07	2.82	2.2	12.9
12-2-81	.16	.30	.97	5-31-82	7-20-82	.25	48	.61	1.66	2.3	1.94	7.4
1-31-84	.11	22	.99	7-29-84	9-17-84	.26	40	.65	1.55	1 8	2 01	5.6

$v^*_{\infty 1}$, $v^*_{\infty 2}$, $v^*_{\infty 3}$, $v^*_{\infty 4}$ = hyperbolic excess velocities relative to Earth after escape, Mars upon approach, Mars after escape and Earth re-approach, respectively Values are rounded off and given in units of the mean orbital velocity of Earth (about 97,700 ft/sec)

T_1, T_2, T_{cpt} = transfer times Earth to Mars Mars to Earth and capture periods Mars, respectively.

μ_1, μ_2, μ_3 = Mass ratios for Earth departure (350 km orbit), Mars capture (1000 km circ. orbit) and Mars escape, based on initial or terminal thrust/weight ratios indicated and on a specific impulse of 850 sec.

R_P = Perihelion distance during outgoing and return transfer in astronomical units

Notes: 1. Earth departure dates have been selected to prevent R_P to fall below 0.6 A.U. at any mission and to reflect increasing capability for hyperbolic entry into the Earth atmosphere

2. Mass ratios given only for the first three maneuvers, because:

(a) the last maneuver takes place after much mass reduction (essentially to weight of entry capsule),

(b) capability to enter at hyperbolic speed will advance, resulting in changing amounts of propellants for retrothrust to reduce speed to maximum permissible entry velocity,

(c) no entry maneuver or retro-maneuver may be executed, but crew picked up by Earth-launched vehicle via hyperbolic rendezvous.

Thus, terminal conditions may vary greatly.

Comercial aviation, for example, would not be feasible, if airplanes could not take off on a rainy day. In the area of Earth-to-orbit transportation, corresponding requirements mean freedom of constraints of return from orbit other than those dictated under certain conditions by the laws of celestial mechanics. The ELV should be able (a) to return from orbit directly; a condition which is particularly difficult to meet with nuclear engines as they are presently designed (design changes must permit the jettisoning of the nuclear reactor prior to atmospheric entry); (b) to return to the vicinity of the launch site; for blunt bodies this means the ELV must return after 1 - 3 revolutions or must stay in orbit for 14 - 15 revolutions (approx. 1 day) or more, depending on the particular resonance conditions between launch site and orbit. Requirement (a) makes return after 1 to 3 revolutions most desirable; as in every commercial operation, rapid recycling contributed to greater economy; (c) to execute the return, independent of weather conditions in the landing area (barring rare extremes, such as hurricanes); this imposes constraints upon the mode of re-entry as well as of touchdown and discourages the use of aerodynamic devices, such as large balloons, or large parachutes or paragliders.

Requirement (c), high reliability, is a contributory requirement to economy of operation; but more than that, operational reliability is vital for operational safety as well as for the safeguarding of expensive payload or the timely delivery payload on which the lives of persons in orbit or on the Moon may depend. An important prerequisite for the achievement of high reliability is simplicity of flight operational procedures, i.e. minimizing the number of "events" which must take place to assure mission success, so long as this is not done at the expense of component and subsystem simplicity. It means that the vehicle should undergo as few configurational changes as possible in the course of its mission which involves ascent, orbit coasting, descent and touchdown, preferably on water.

The simplest re-entry body is of blunt shape, low $W/C_D A$, and large nose radius. Aside from being very suitable for re-entry, this kind of shape is, for all practical purposes also dictated by the physical dimensions of payloads. For such payload dimensions only a very blunt shape leads to a configuration which is dynamically (load dynamically) not only uncontroversial, but even advantageous; and only a very blunt shape leads to a vehicle plus payload height which avoids excessively tall launch structures, such as 500 - 700 ft tall service structures.

The answer to the many conflicting requirements implied by the multitude of things which tomorrows ELV must be, is primarily: bluntness. Just as the automobile, the airplane and the missile had to find their own characteristic shape, so there is a characteristic shape for large ELV's. It is a blunt-body shape, somewhat reminiscent of the shape of a Galapagos turtle or a mushroom head. Together with its towering payload it represents an aerodynamically and dynamically acceptable ascent configuration (Fig. 36). Without the payload it posses a favorable re-entry configuration (Fig. 37). Its large nose radius (120 ft or more) and the large diameter (140 ft or more) provide a large area which, in connection with the comparatively low W/A of 50 - 80 lb ft^2, offer the potential of some flide control of the descent path in more advanced operational entry modes. This potential is not utilized in the first version under study.

The concept of the blunt-shaped ELV has been named, for brevity, NEXUS, the link between Earth and space. One-stage and one-and-a-half stage versions have been considered. Although the NEXUS concept has been applied primarily to chemical Post-Saturn vehicles, it is an attractive configuration for a chemonuclear ELV, since its large diameter allows open cluster arrangements of nuclear engines. The configuration also offers advantages to a second-generation Saturn V when applied to a recoverable first stage.

10. The Single-Stage NEXUS Vehicle

The single stage NEXUS being larger in diameter than height is unconventional when compared to present day high L/D booster configurations. The demand for a large payload capability coupled with single stage operations results in a vehicle weight and shape which is relatively huge

For comparison, the NEXUS boosts about 4.5 times the Saturn V payload, has a diameter about five times greater and posses about four times the take-off weight The height of the two vehicles on the launch stand is approximately the same (Fig. 36) Because of the magnitude of the NEXUS weight and size, new techniques will be required for the manufacturing, mating and handling during construction of the vehicle.

The NEXUS configuration is determined by the requirement for: simplicity in manufacturing, total vehicle recovery, minimum vehicle height when on the launcher and by simplicity in operation from launch through recovery.

The vehicle features a blunt-body shape with a large nose radius for re-entry. The large radius reduces the aerodynamic heating so that no special protection in the form of high temperature or ablation materials will be necessary as a heat shield The blunt nose combined with extendable flaps provides the vehicle with the required static margin of stability during its re-entry into the atmosphere. Final deceleration from about 350 ft/sec is provided by retro-thrust or the combination of parachute and retro-thrust. The vehicle is designed for water landing. Touchdown rockets are provided for reducing the impact acceleration and for patterning the water surface to provide more ideal entry conditions.

The single stage NEXUS concept is not restricted to one certain size and weight vehicle but can vary greatly in its magnitude. A comparison illustration is shown in Figure 38 . The construction details of the various size concepts are similar and for all practical purposes are independent of the vehicles size and weight. However, for purposes of discussing finite values of vehicle characteristics, a 24 million pound gross weight NEXUS will be presented. An illustration of the vehicle is shown in Figure 39. The purpose of the larger version is discussed below.

With an interplanetary payload the vehicle stands approximately 400 feet. The Nexus itself is 115 feet high, 150 feet in diameter at the heat shield and 164 feet at its base. Although the illustration shows a plug nozzle, the dimensions are approximately the same for clustered advanced expansion-deflection engines. The plug's even distribution of thrust load into the vehicle structure and its favorable center of gravity location (forward for re-entry) makes it an ideally suited main engine configuration for the NEXUS.

The concept features a large spherical-toriconical hydrogen tank with a 120 foot dome radius. Over the tank dome is crushable structure which supports the 0.1 inch thick titanium heat shield. Blow-out panels are provided in the shield at touchdown rocket locations. Being a reusable vehicle, the hydrogen tank is not employed as a load carrying member for external loads when in its chilled condition. It supports its own weight (366,000 lbs), LH_2 weight (3,500,000 lbs), liquid hydrostatic head and internal pressure loads (25 psi). Payload and thrust loads bypass the tank through external load carrying members. This design criteria is a precaution against unpredictable loads that could create a leakage of the tank by racking and material fatigue.

The hydrogen tank is constructed of 5Al-2. 5 Sn titanium sheet varying from 0.23" thick at the dome to 0. 35" skin on the cone. The assembly is butt-welded. Titanium was selected because of its great resistance to sea water corrosion and its favorable strength to weight ratio. The tank is supported by its upper connection at maximum tank diameter and near the bottom where the touchdown rocket reaction structure joins the tank. The upper connection supports the tank during ascent and prevents the shell from buckling if internal pressure is lost on the launch stand. The lower connection supports the tank during touchdown rocket firing and touchdown loads. (The tank shell is warmed by aerodynamic and pressurization gas heating prior to this loading.)

The LH_2 tank is insulated with 0. 25" thick fiberglass honeycomb covered with a sealed layer of fiberglass sheet. This forms a cryogenic insulation capable of providing an exterior tank temperature above zero degrees Fahrenheit.

The liquid oxygen tank is a comparmented spherical torus assembly surrounding the aft portion of the hydrogen tank. Its cross section diameter is about 28 feet and outside diameter nearly 160 feet. It weighs about 121,000 pounds including frames. The tank is assembled of twenty-four spherical sections of .032" formed stainless steel sheets welded together to form the torus. At each spherical intersection is a frame with a internal slosh ring and an external flange for pin joint attachment to the vehicle load carrying structure. Type 301 steel was selected for the oxygen tank because of its compatability with the chemical.

The spherical segmented torus tank with chem-milled sheets ranging from .062 to .032 is 35% lighter than a constant section torus tank.

The load carrying structure supporting the payload, tanks, side fairings and engines weighs about 160,000 pounds. It is constrcted of titanium alloy panel segments stiffened with tees and joined by fusion butt welds. The members form a conical lattice type structure.

The heat shield is a titanium shell about 0.1" thick. It is bonded to styrofoam about one foot thick at the center and increasing to five feet thick at the edges. The styrofoam provides a crushable structure if necessary at touchdown to prevent damage to the hydrogen tank.

A titanium fairing surrounds the entire vehicle protecting it against aerodynamic heating during ascent and re-entry and from water spray while in the sea. Imbedded in the fairing, flush with its surface, are four wedge shaped flaps. They provide center of pressure control during re-entry and potential aerodynamic maneuvering capability during launch. The flaps are hydraulically actuated and weigh about 13,000 pounds each. The wedge shape places the center of pressure further aft on the surface than that of a rectangular flap and minimizes possible flutter tendencies.

The main propulsion system is a truncated plug engine. It is a high pressure, throttleable, altitude compensating oxygen/hydrogen engine producing a maximum of 24 million pounds of thrust. It has an expansion ratio of $\epsilon = 150:1$ and a mixture ratio of about 5:1. Liquid hydrogen is supplied to the engine by a pump system from a hydrogen pressurized LH_2

tank. The tank pressure is about 25 psi. Liquid oxygen is supplied by pumps from the oxygen pressurized LOX tank of 25 psi. Since the LOX tank outlets are below the engine inlet, a saving in residuals is accomplished by reducing the number of active feed lines by valving and drainage systems as the engine is throttled back during ascent.

Because very little thrust vector control appears to be available from a plug engine by throttling or other means, the NEXUS concept incorporates separate control engines. Since the inherent center of gravity of the take-off configuration is close to the engines and since the NEXUS has such a broad base diameter, throttleable control engines mounted at the outer periphery of the base are very effective. Thrust variation of these engines produced a far greater torque than swivelling. Five 400, 000 lb thrust units in each of four quadrants could satisfactorily control the NEXUS during max-q condition

The NEXUS concept does not require gimballed or swivelled engine installations. This is very advantageous (for reasons of weight and reliability) where large cryogenic feed lines and high-pressure engine systems are involved.

The interstage adapter, although not part of the NEXUS recoverable booster, is partially retained after payload separation. The de-orbiting retro-rockets are installed in the lower part of the adapter and aligned so their centers of gravities pass through the vehicles center of gravity. When the payload is released from the booster the upper portion of the adapter containing the trans-stage propulsion separates from the lower portion exposing the retro-rocket nozzles. After retro-firing the adapter section is released by small rockets canted to remove the structure from the normal flight path. The adapter is constructed of light weight honeycomb-sandwich structure.

For final touchdown low impact deceleration is of prime importance in order to minimize the vehicle structural weight. The most favored method of easing the vehicle's impact at touchdown is by providing suitably located retro-rockets. The rockets must provide sufficient impulse to stop the vehicle a short distance above the water, then with less thrust, allow the vehicle to enter the water at a controlled velocity. The allowable impact velocity is determined by the forces the vehicle's structure is able to tolerate. Also, the reaction loads of the touchdown rockets influence on the design of the forward bulkhead. The NEXUS bulkhead being a thin shell of approximately 150 feet in diameter with a 120 foot nose radius is a somewhat more unwieldy design than bulkheads with smaller radii of curvature. Therefore, to accept impact loads, provision of supporting structure to localized areas of the bulkhead appears most attractive and is incorporated in the NEXUS design The weight of this support structure is a function of the touchdown retro-thrust and the water impact loads.

The touchdown impact loads may not act against well defined reinforced areas because their reaction remains unpredictable and the touchdown conditions in general are unknown. An attempt has been made to clarify this by constructing a small scale model shown in Figure 40 . Scale tests were conducted which provided rather interesting results. It became apparent that water could be displaced and patterned by the touchdown jets, the size and shape of the pattern being a function of the number, magnitude, and location of the jets with respect to the (NEXUS) center line. The backflow of water, steam and rocket exhaust would change conditions at the impact point favorably for the entering NEXUS, producing a significant cushioning effect which is provided by bubbling water, an effectively less dense fluid.

Although the preceding discussion, for simplicity, assumed only touchdown rockets without parachutes, or balloons, etc., the weight implications must be understood. For this reason, a parachute study was made which indicated that an optimum weight condition is obtained when both parachutes and rockets are used. The vehicle would first be reduced in velocity to about 75 ft/sec with parachutes, and rockets would then decrease the velocity to zero. This combination weighs approximately 50% of the "rockets only" system.

Preliminary indications of impact conditions and the rocket exhaust influence have been obtained through experimental model tests. The use of small scale models should provide valid information since the action of the water surface is primarily a wave phenomenon which can be scaled using Fraude's Number.

A 1/160 scale test model of the NEXUS forward bulkhead with 49 jet ports was built. Having all ports operating simulates seven million pounds of retro-thrust which corresponds to 3.5 g. Nine ports simulate 1.6 million pounds of retro-thrust or 0.8 g which allows the NEXUS to descend at 0.2 g.

Figure 40 shows the jet impingement on water at a scaled 30 foot altitude. The test simulates the condition where the main touchdown rockets have subsided leaving the final touchdown rockets patterning the water. Figure 40 shows the final touchdown rockets disturbing the water when the nose of the vehicle is beginning to penetrate the zero water line.

There are some general conclusions which may be drawn from the test results and are pertinent to touchdown in general.

(1) The cavity formed in the water is such that vehicle impact will be applied near the outer periphery of the vehicle, and not in the center.

(2) The ground effect is felt at a considerable altitude. Approximately 10% increase in apparent thrust at 100 ft and over 20% increase at 30 ft attitude. This means that rocket weight will be saved, but this effect has not been considered in rocket weight estimate.

(3) When only the final touchdown thrust is active (1.6 Meg. thrust), there appears to be 20% increase in apparent thrust at 30 ft altitude, which drops back to 0 increase at 15 ft altitude, then increases to 100% increase at 0 altitude. This variable condition will need careful evaluation before rocket total impulse, thrust, and probable vehicle impact velocity can be established.

(4) There are several areas as yet unexplored in which model testing may be of considerable benefit:

(a) instability due to interaction of vehicle attitude and ground effect;

(b) effect of gas temperature - some energy will be removed from the jet through heat transfer to the liquid;

(c) water density in the surface layers of water. The effect will be significant to the impact forces on the vehicle;

(d) Rebounding from the water due to thrust, ground effects and buoyancy.

(e) Effects of horizontal velocity on contact pressure and g-tolerances have a significant effect on the condition of touchdown.

Because of these tolerances and in order to keep the impact velocity low, a 30,000 lb rocket weight has been assigned for the final touchdown.

Both altitude and velocity are of concern and magnitudes of these have been assumed from various causes.

	Altitude (feet)	Velocity (feet/sec)
Measurement tolerance		
altitude 4 ft 1/2%	6.2	
velocity 1 ft/sec 1/2%		2.3
Engines		
3% total impulse	.4	8.
thrust build-up time	6.3	
Ocean waves	5.	3.2
Propellant Residuals		
1.5% variable total vehicle (these may be jettisoned)	.2	4.

During descent with the touchdown rocket thrust applied, there are two main considerations. One is the overturning moment due to ground effects and the other is horizontal velocity component due to vehicle attitude. Both problems indicate the need for an active attitude control system during touchdown.

As the vehicle comes close to the water surface, the return flow of water and gas may apply an overturning moment to the vehicle if its attitude is not controlled. This instability is dependent on rocket placement and direction of thrust and is also a function of distance from the water surface.

The touchdown rocket weight varies with terminal velocity of vehicle due to the Δv requirements, and varies with thrust due to the gravity effects.

The best thrust level at which to operate the touchdown rockets would appear to be the highest, however, structure weight also has an influence. The reaction structure inside the LH_2 tank is designed to react rocket thrust, and its weight increases with greater thrust. The trade-off between rocket weight and structure weight optimum at approximately 9,000,000 lbs of retro-thrust.

Final touchdown has been considered to be a second phase of the touchdown operation. The main touchdown rockets are assumed to burn out at some distance above the water surface and to have brought the vehicle to 0 velocity at this point. The final touchdown rockets, at less than 1 g, are still firing, allowing a descent to the water. As the water surface is approached, the ground effect raises the apparent thrust to more than 1 g, and the vehicle is again decelerated. This approach eliminates a high impact velocity.

TAB. 10

INTENTIONALLY

OMITTED

Table 11 Intentionally Omitted

The wet inert weight for the 48 M vehicle is 65% heavier than for the 24 M vehicle. The main controlling factor for the increase lies in the tremendous gain in tank volume with respect to increases in the vehicle's linear dimensions.

The largest single element of the NEXUS inert weight is the huge titanium LH_2 tank. The support rings are welded to the upper portion of the tank where the thrust ring is where the load carrying structure and engine thrust structure attach. Thermal insulation is provided to minimize boil-off while on the launcher and to prevent air from icing on the tank surface.

The segmented LO_2 tank requires insulation and a heat shielding for protection against engine base heating. The control engine thrust is introduced into the tank rings and require special mounting brackets in each of the four quadrant positions. These attachments add 3,000 pounds each.

The titanium load carrying structure acts as the main vehicle structure for load distribution between engines, payload and tanks. The aerodynamic fairing surrounds the vehicle and contains the four control flaps. The JATO attach structure of 6,000 pounds is included in the weights for attach points.

The NEXUS may require assist when heavier than normal payloads are specified. These attachments are distributed around the periphery of the aerodynamic fairing.

The propulsion group includes the main plug engine, the control engines, the valves and lines for the engines and an attitude control system for orienting the vehicle for re-entry and for touchdown.

Residuals comprise nearly 9% of the wet inert weight as pressurization gases in the tanks, unrecoverable and trapped liquids and extra propellant required for propellant utilization error.

The interstage adapter although not a recoverable part of Nexus is included in the wet inert weight. It contains the de-orbit retro-rocket system for re-entry. The recovery system is approximately 18% of the wet inert weight and consists of all items unique with the Nexus touchdown rocket system. These items are; the heat shield, the crushable structure between the heat shield and the LH_2 tank, the retro-rockets and the structure inside the LH_2 tank required to react the thrust load of the retro-rockets.

The comparative marginality of a chemical 1-stage ELV for 10^6 lb payload causes concern as to the development risk undertaken regarding the actual payload available at the end of the development phase.

TAB. 12

INTENTIONALLY

OMITTED

Instead of using JATO's and before deciding that a 1-stage version is too marginal and should be abandoned for a 1 1/2 or 2-stage version, it is of interest to look in the opposite direction, toward larger 1-stage vehicles. Neither a payload of 10^6 lb nor a take-off weight of $24 \cdot 10^6$ lb are limiting figures. Conditions become rapidly more favorable in many respects as the vehicle size is increased. The 48M version has a significantly relaxed I_{sp}-requirement. It can sustain weight increases (reductions in mass fraction) and reductions in specific impulse which would render the 24M version useless, and still retain a payload capability which exceeds 10^6 lb. Considering a larger version not only reduces the development risk but also provides a significant growth potential. The 48M version may become operational with $1.4 \cdot 10^6$ lb payload, for example, and, in the course of further refinements grow into a 1.6 to 2 million pound payload capability. Among other vehicles, the Atlas ICBM is an outstanding example that this can be accomplished.

11. Other Applications of the NEXUS Configuration

The large diameter of the NEXUS vehicle renders it suitable to accomodate large nuclear pulse-type vehicles of the future, such as Orion. This is one aspect which lends this configuration a low rate of obsolescence.

Another application is its growth into a chemonuclear vehicle, retaining a chemical plug engine system in the center, reducing considerably the oxygen/ tank and adding a torus-shaped LH_2 tank, in the place of the large Lox tank of the chemical version, under which the control engines as well as the nuclear engines are mounted in an open cluster of 4 engines @ 750 k or 12 engines @ 250 k. This appears feasible with 24M version whose diameter at the control engine distance is about 140 ft, yielding a circumference of 440 ft. thus keeping the nearest distance between any two of the 12 engines to some 36 ft.

In conclusion it should be pointed out that application of the NEXUS shape to the first stage of Saturn V not only would result in a recoverable first stage for this ELV; but it also would improve further the usefulness and mission versatility of this ELV, especially for the preparation of planetary missions, because the volume limitation of the present configuration, discussed at the end of Par. 6.1 above, would be removed. Fig. 42 compares the present configuration of Saturn V with that of a 50 ft diameter Saturn V-R with blunt, recoverable first stage. It could transport a considerably larger volume. Fig. 43 shows a concept of the Lox/RP-1 recoverable first stage with uprated F-1 engines.

REFERENCES

1. Ehricke, K. A., Orbital Operations, Advances in Space Science and Technology (F. Ordway, III, edit.), vol. 5, Academic Press, New York, 1963

2. Space Technology Laboratories, Inc., Reliability of U. S. Rockets, STL Rep. No. 8659-6054-RS-000, Nov. 1962

3. Koelle, H. H. and Huber, W. G., Economy of Space Flight, Chapt. 1.9, p.1-65 of Handbook of Astronautical Engineering (H. H. Koelle, edit.) McGraw-Hill Book Company, Inc., New York, 1961

4. Marshall, A. W. and Meckling, W. H., Predictability of the Costs, Time and Success of Development, RAND Corp., Rep. P-1821, October 1959, revised December 1959

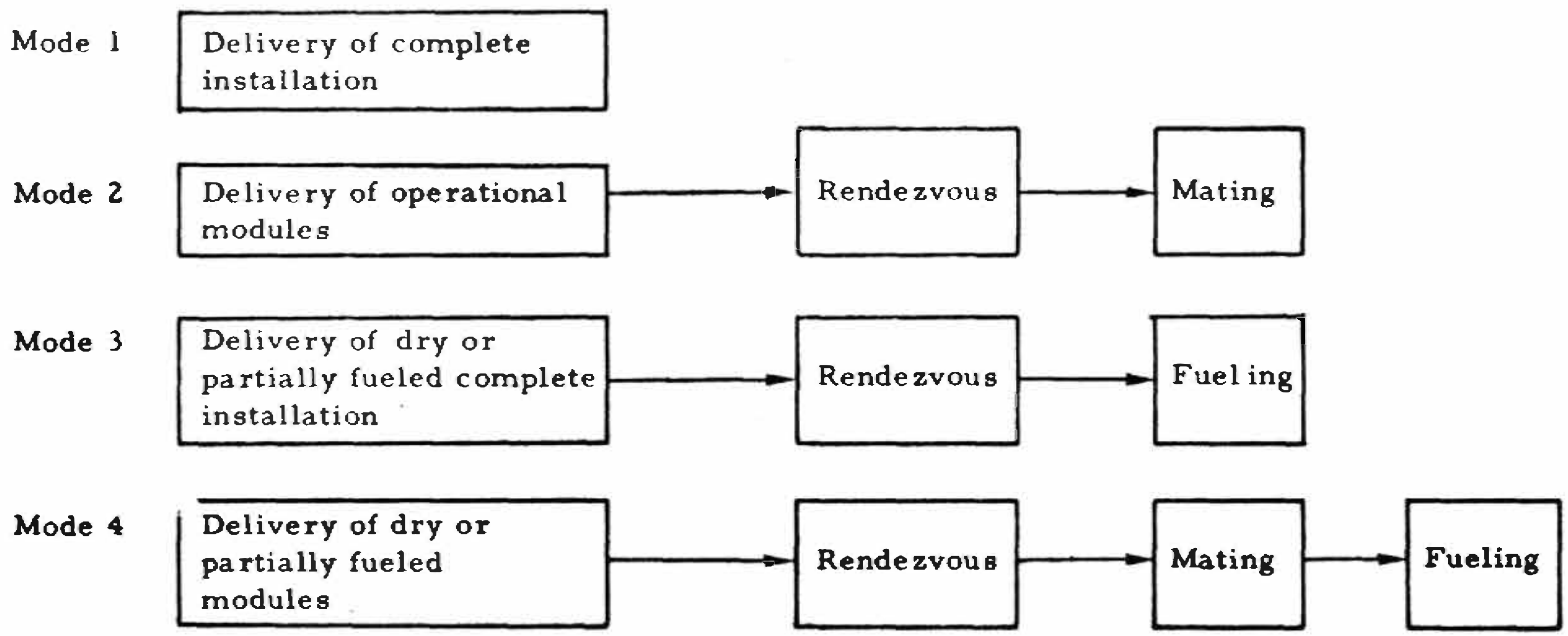
5. Noah, J. W., Identifying and Estimating R and D Costs, RAND Corp., Memorandum RM-3067-PR, May 1962

6. Ehricke, K. A., Advanced Launch and Carrier Vehicles, Chapt. 24.1, p. 24-2 of ref. 3.

7. Ehricke, K. A., A Study of Post-Nova Launch Vehicles, Intermediate Report No. 1. Prepared for the George C. Marshall Space Flight Center, Future Projects Office, Contract NAS8-5022, GD/A Rep. No. AOK62-0005, Sept. 1963

8. Douglas Aircraft Rep. SM-41719 (April, 1962)

Fig. 1 <u>ESTABLISHMENT OF ORBITAL INSTALLATIONS</u>
<u>OPERATIONAL MODES</u>



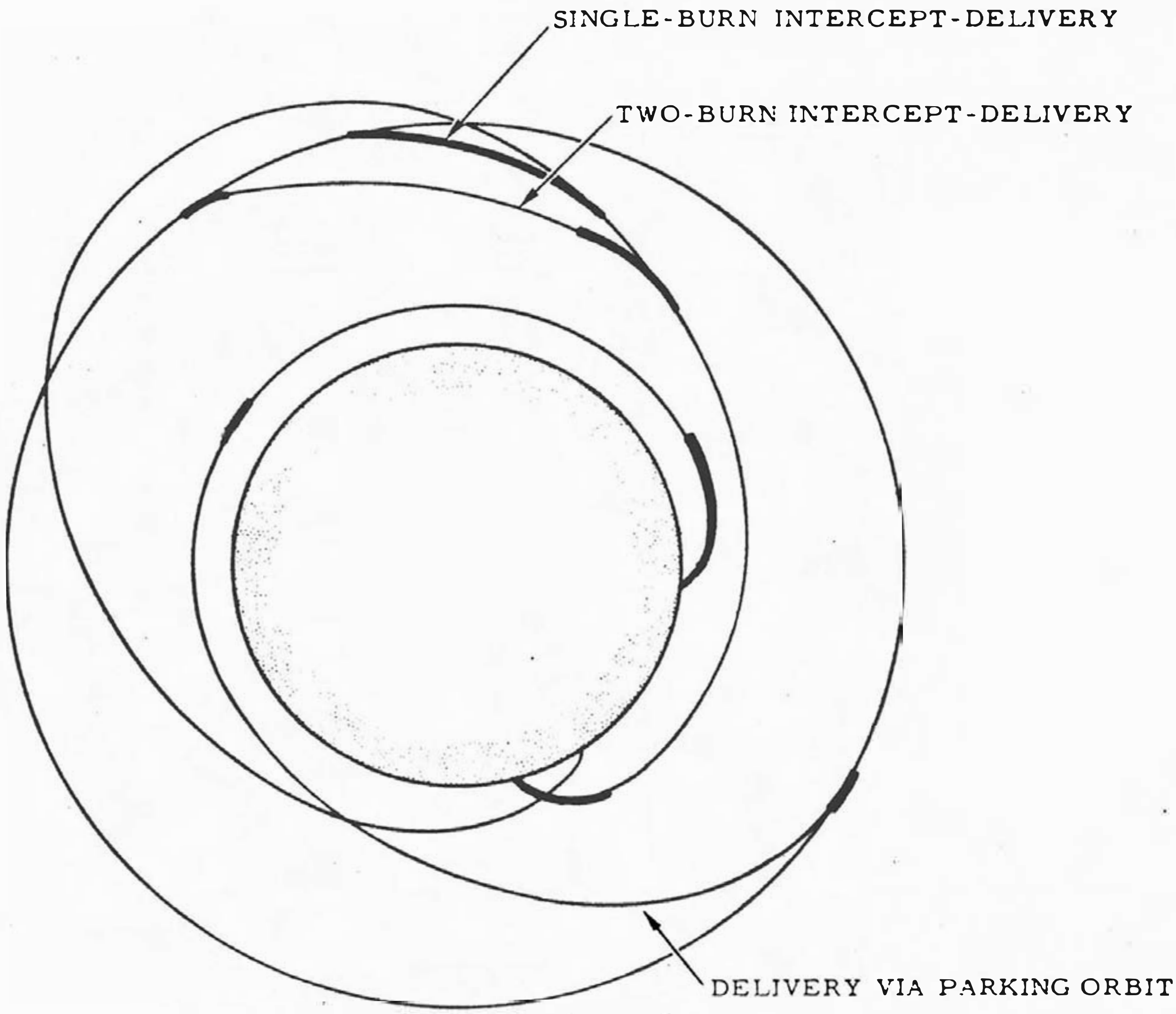


Fig. 2 Intercept-Delivery and Delivery Via Parking Orbit

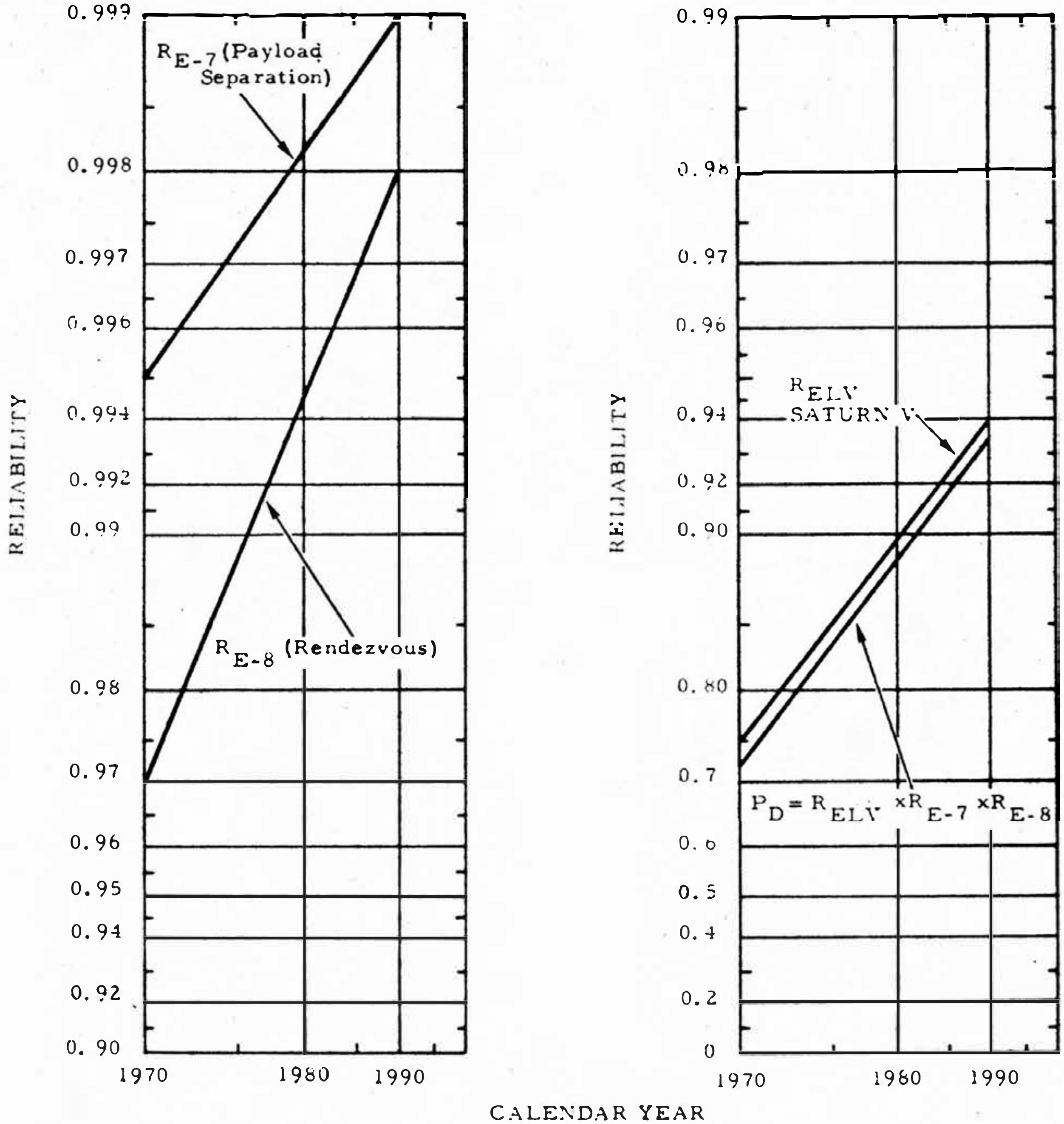


Fig. 3. Reliability of Saturn V, Payload Separation and Rendezvous.

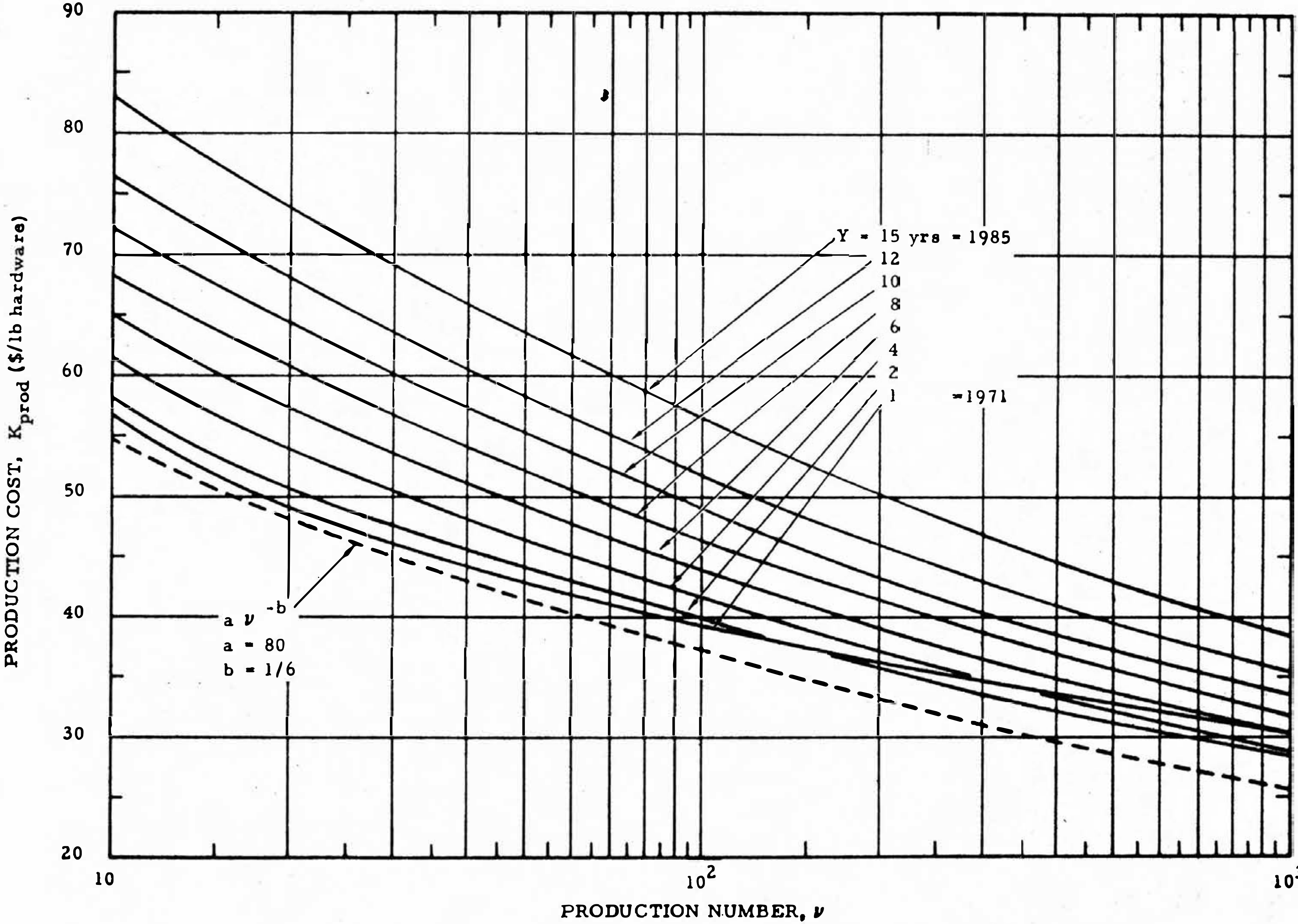


Fig. 4. Possible Variation of Specific Production Cost of Saturn V as Function of Production Number between 1971 and 1985, Based on Eq. (4b).

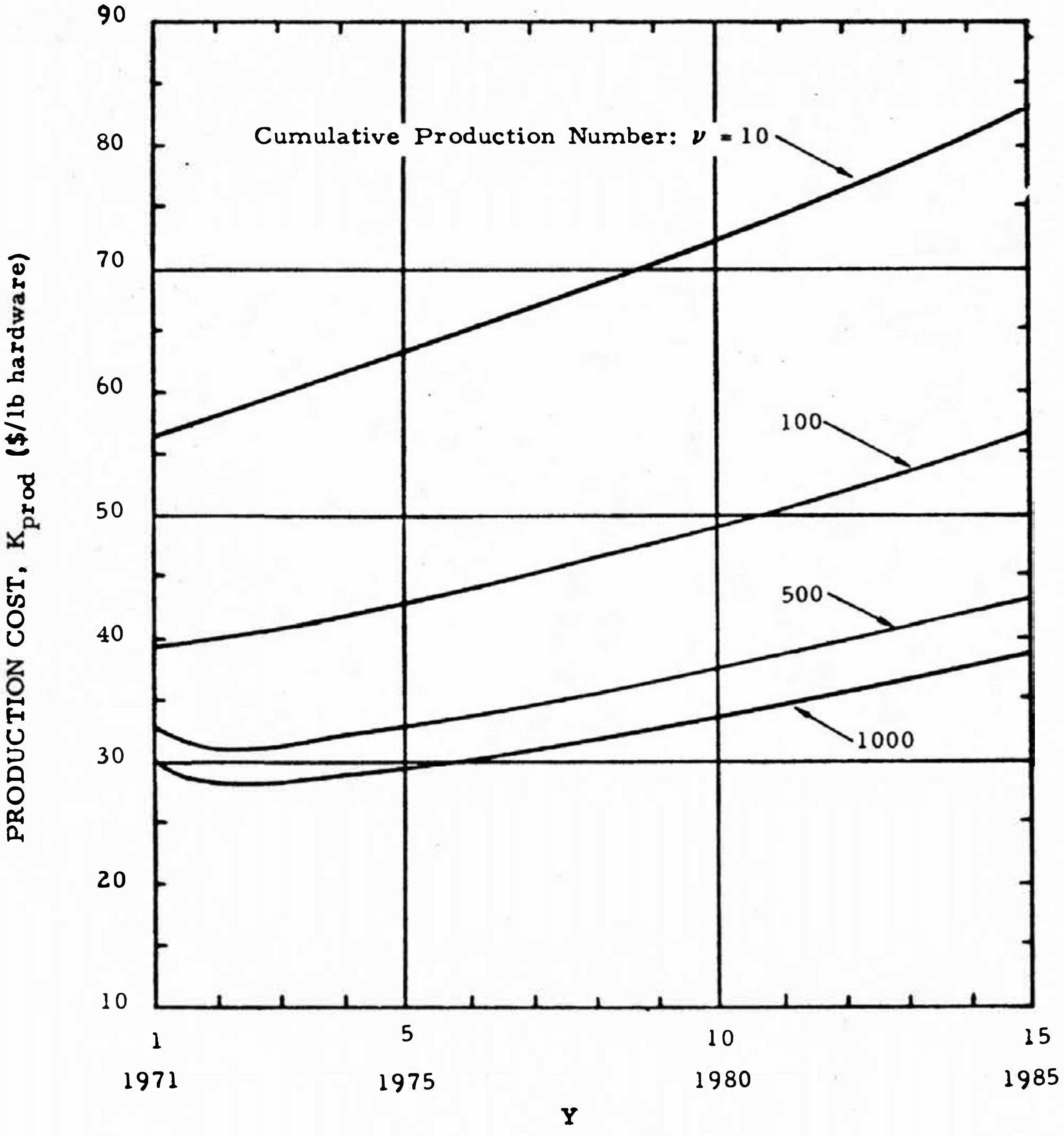


Fig. 5. Possible Variation of Specific Production Cost of Saturn V as Function of Time for Different Cumulative Production Numbers, Based on Eq. (4b).

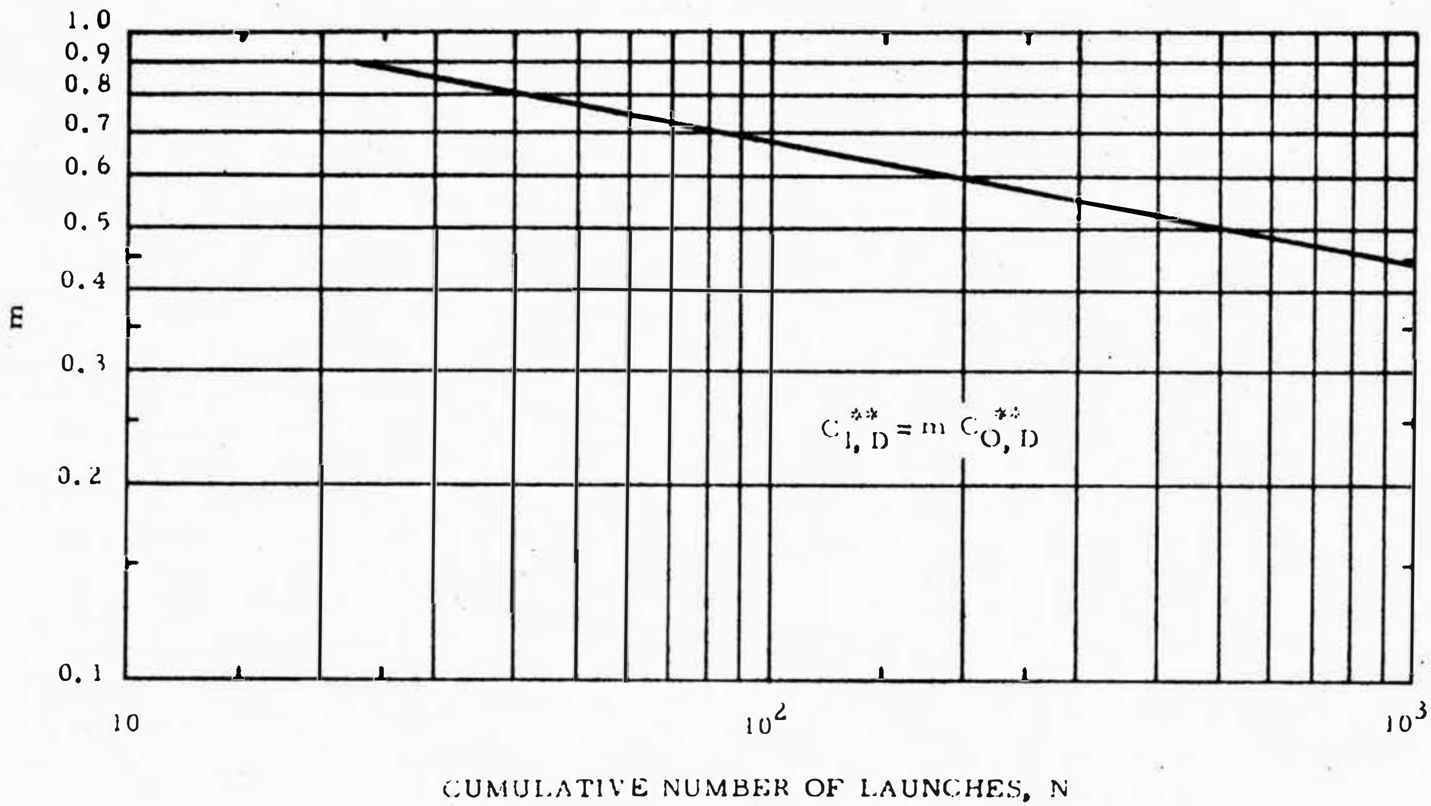


Fig. 6 Variation of Ratio of Indirect to Direct Operating Cost for Orbital Delivery Using a Land-Launched Expendable ELV of Saturn V Type.

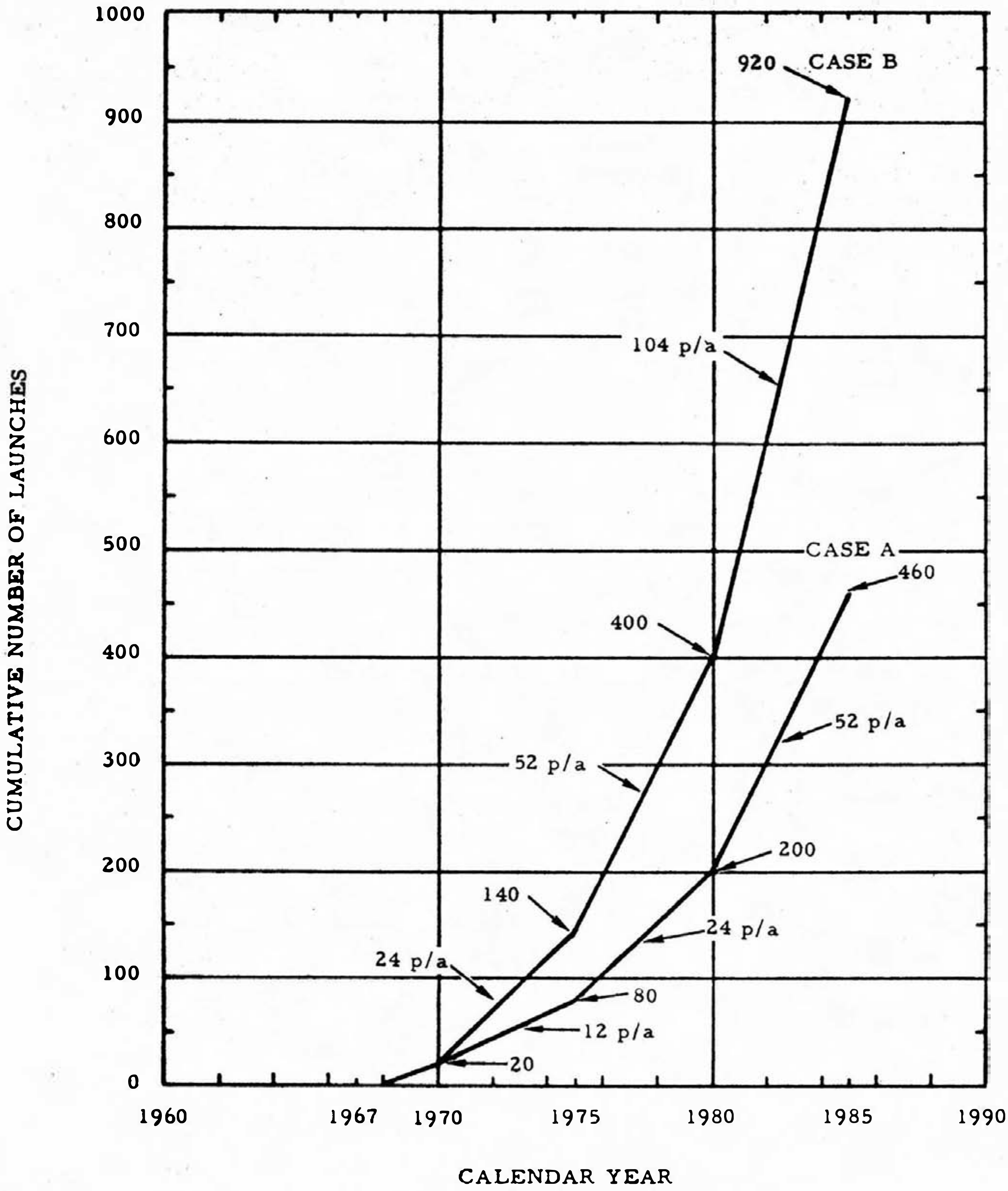


Fig 7 Two Models of Growth of Saturn V Launch Rate.

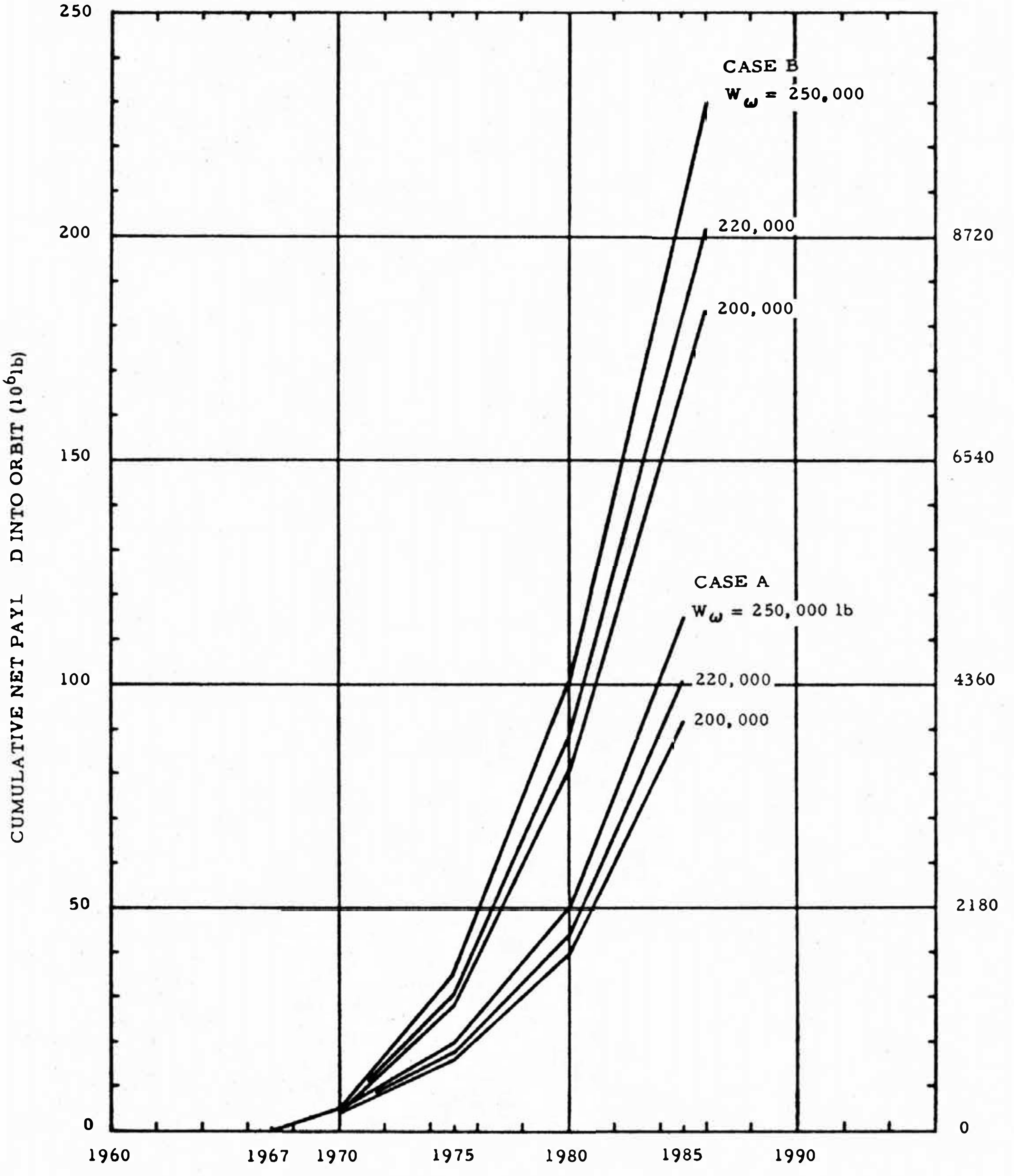


Fig. 8 Cumulative Payload into Orbit for Two Models of Saturn V Launch Rate Development.

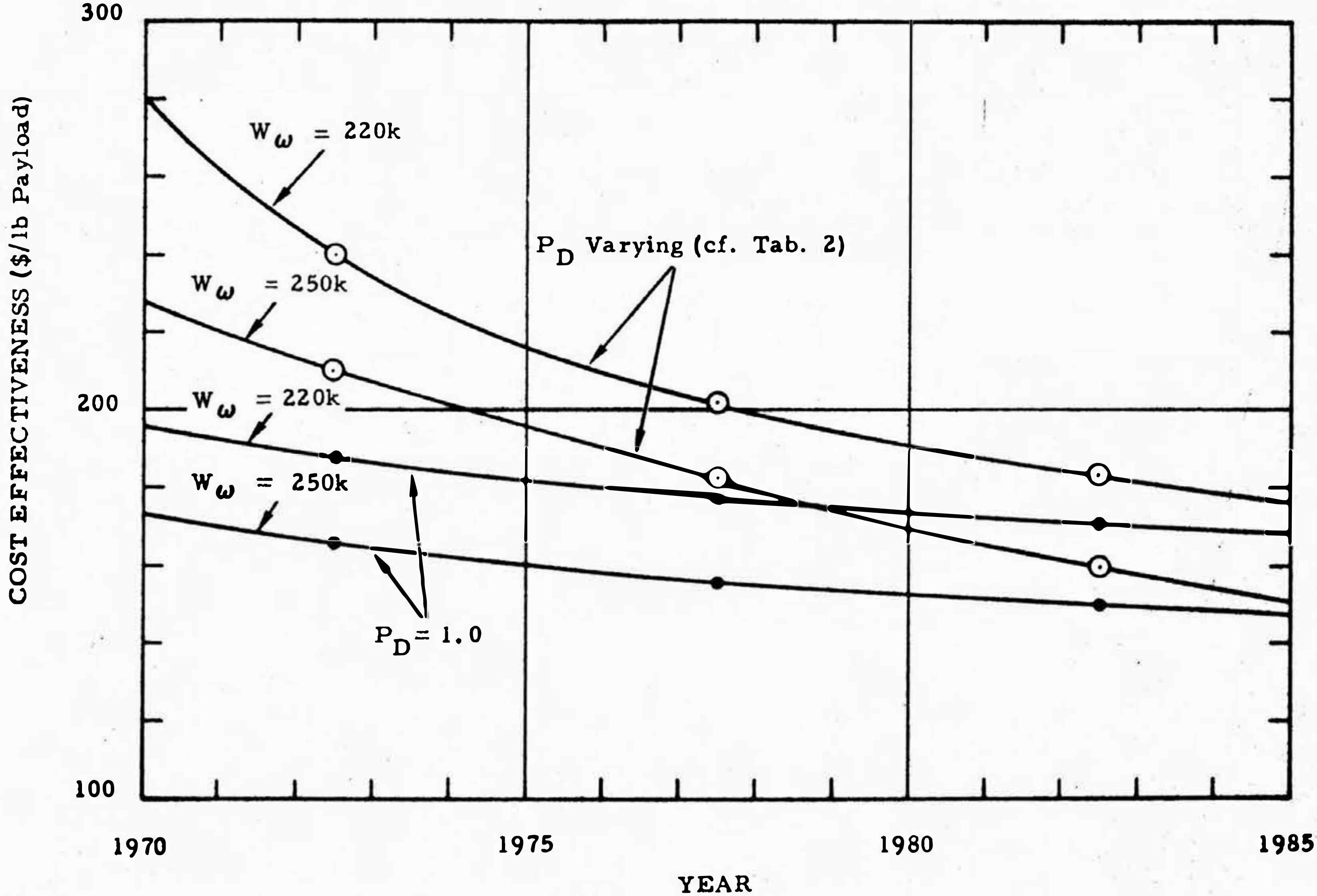


Fig. 9. Total Cost Effectiveness of Saturn V Versus Time, Based on Case A in Fig. 7 and the Mean Values Listed in Tab. 2.

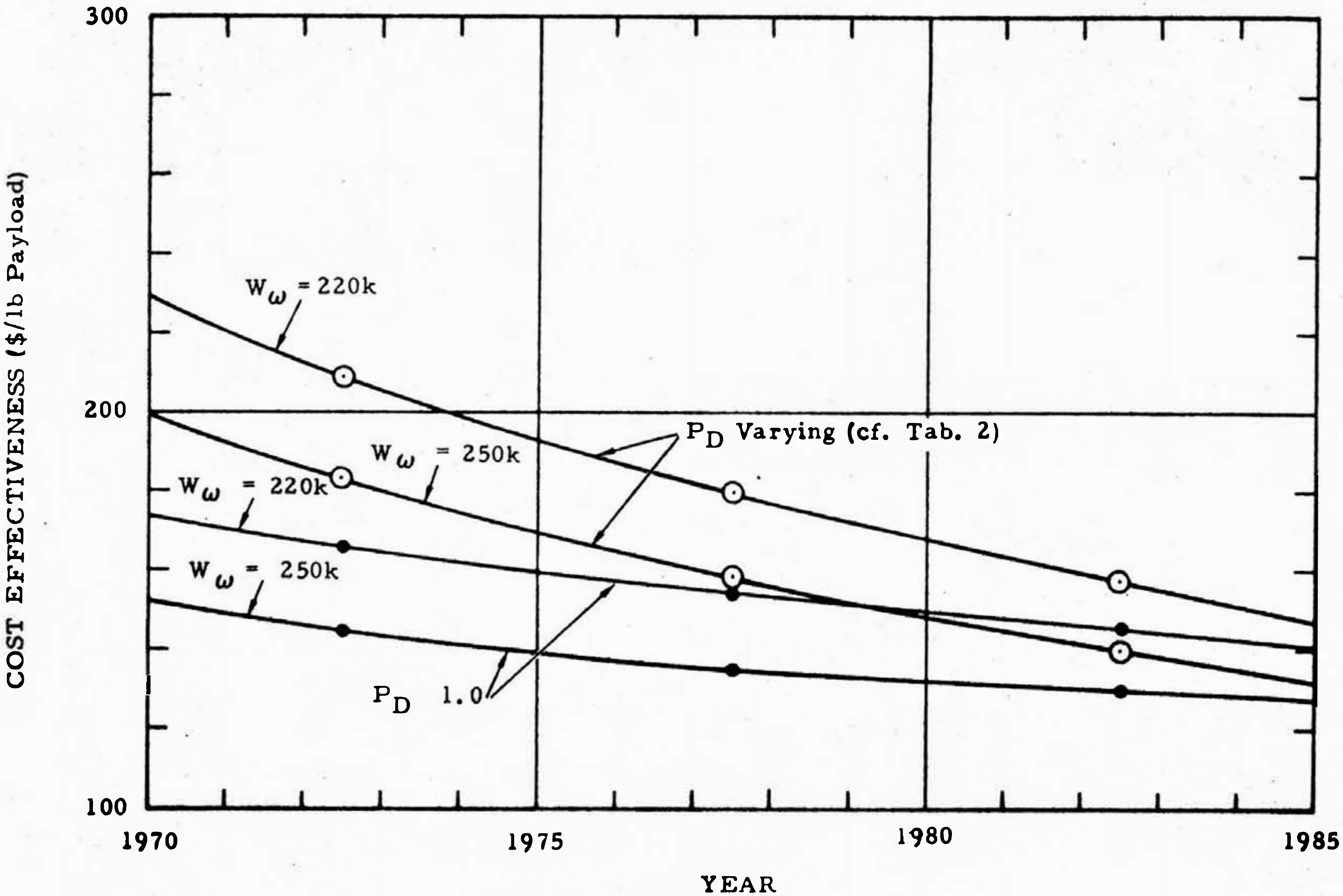


Fig. 10. Total Cost Effectiveness of Saturn V Versus Time, Based on Case B in Fig. 7 and the Mean Values Listed in Tab. 2.

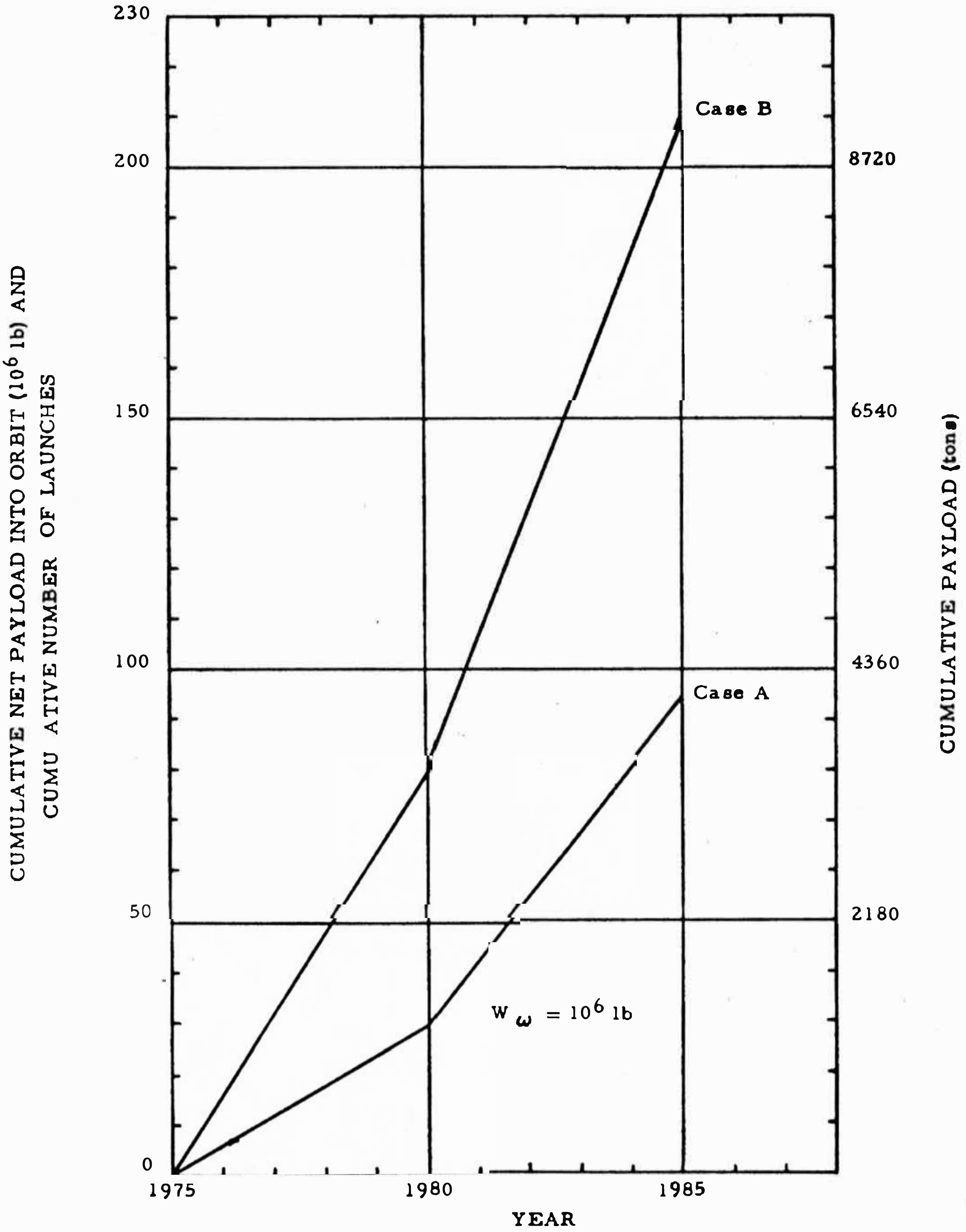


Fig. 11. Cumulative Payload into Orbit for Two Models of a Post-Saturn ELV Launch Rate Development.

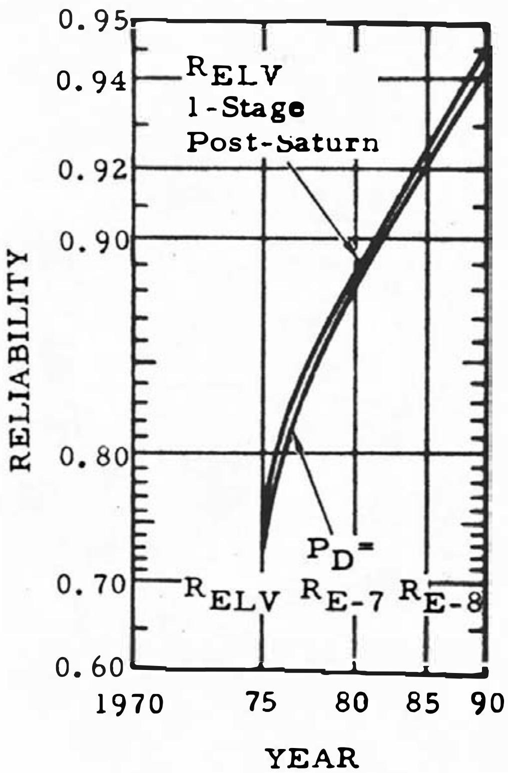


Fig. 12. Assumed Reliability Versus Time for Post-Saturn ELV.

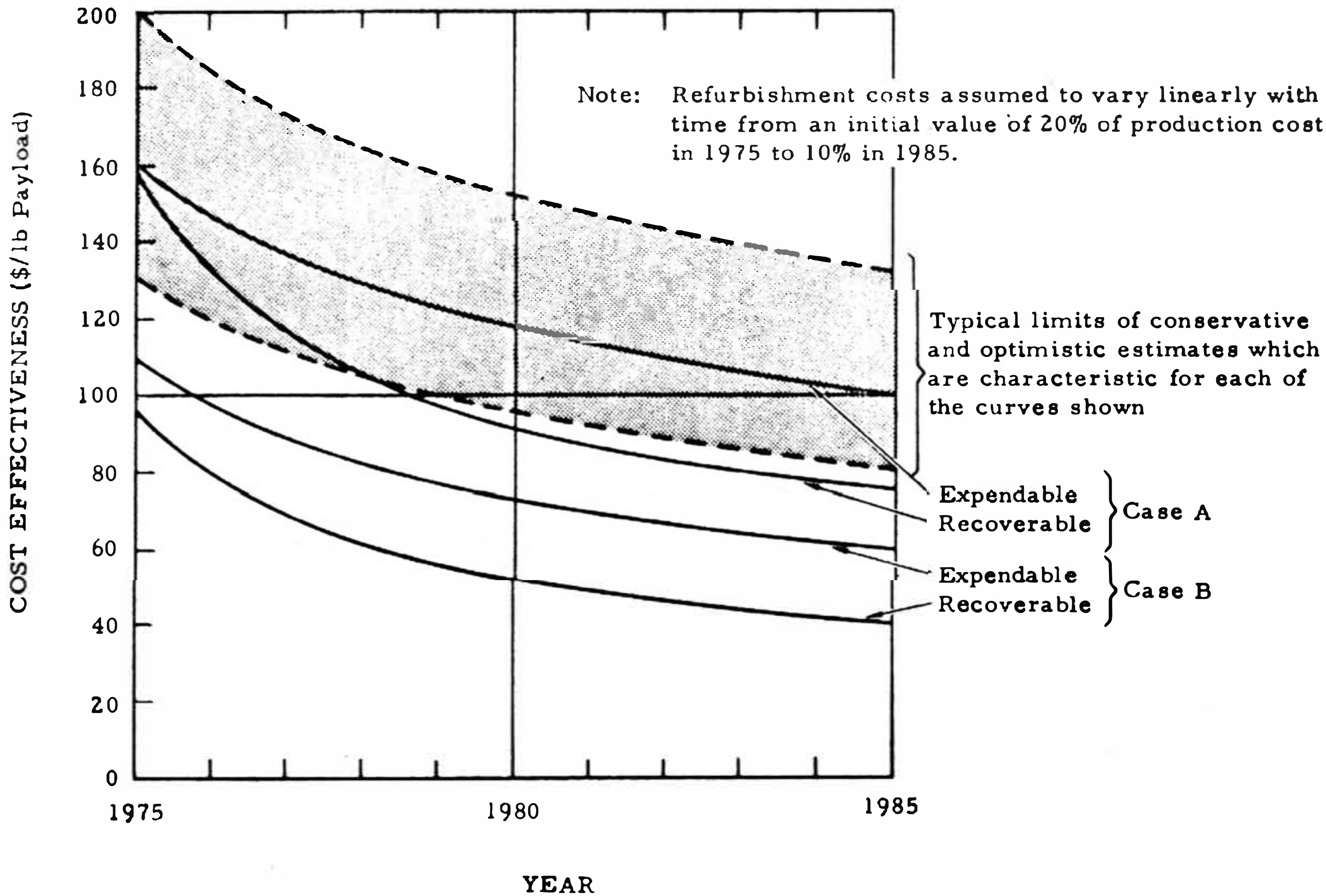


Fig. 13. Typical Total Cost Effectiveness Curves for a Single-Stage Chemical ELV of 10^6 lb Payload, for Successful Delivery of Payload Weights Specified in Fig. 11.

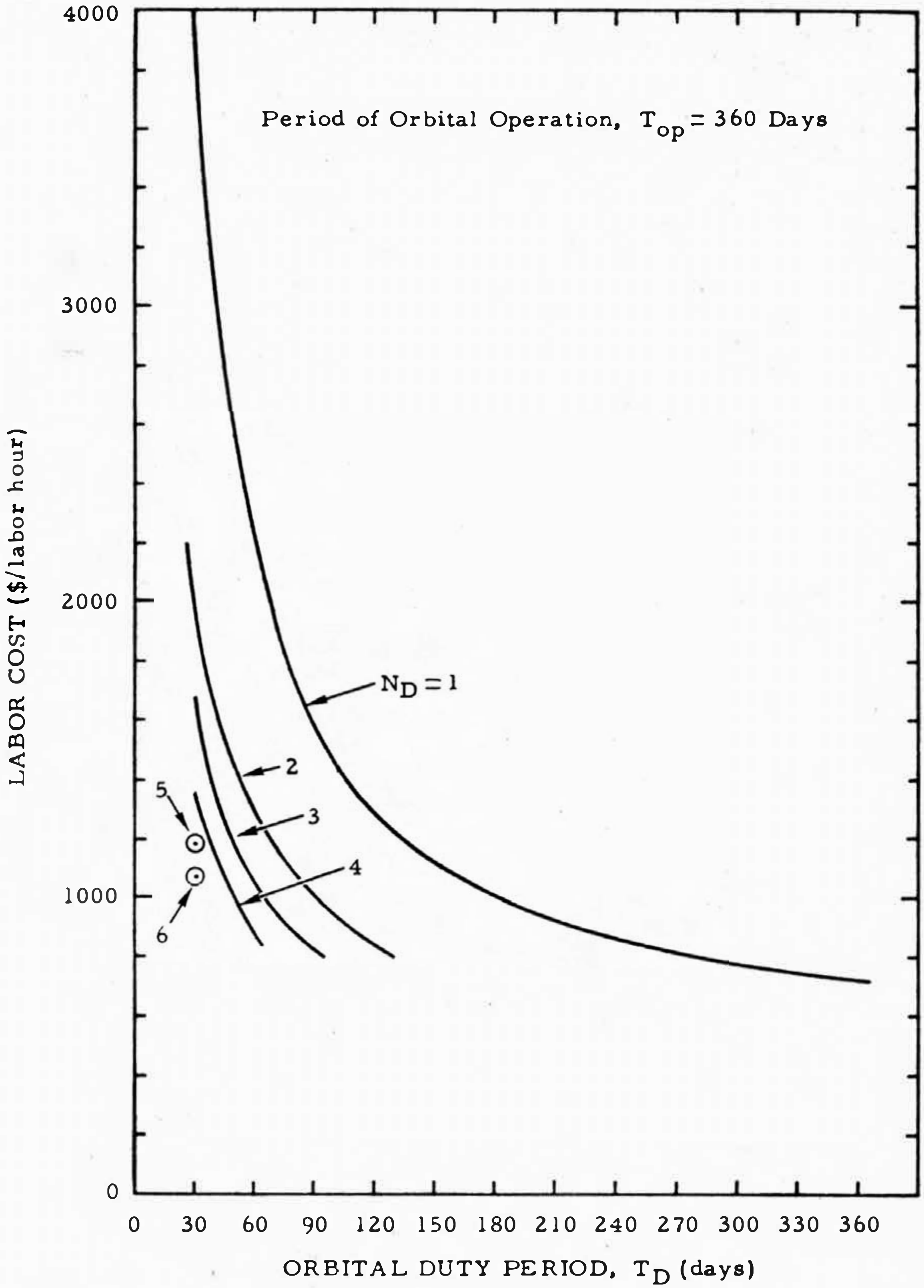


Fig. 14 Hourly Orbital Labor Cost for an Operational Period of 360 Days.

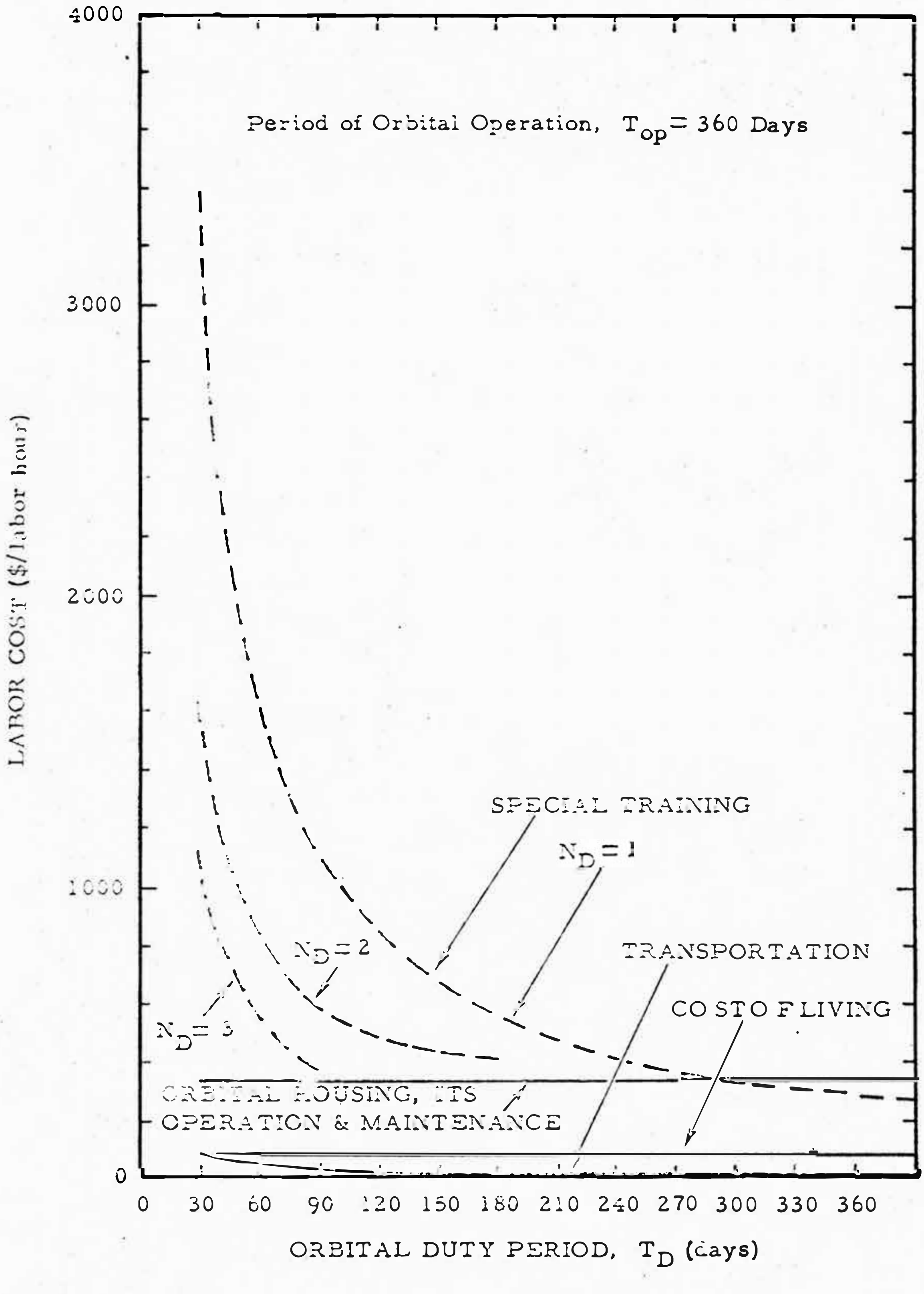


Fig. 15 Factors Determining Orbital Labor Cost.

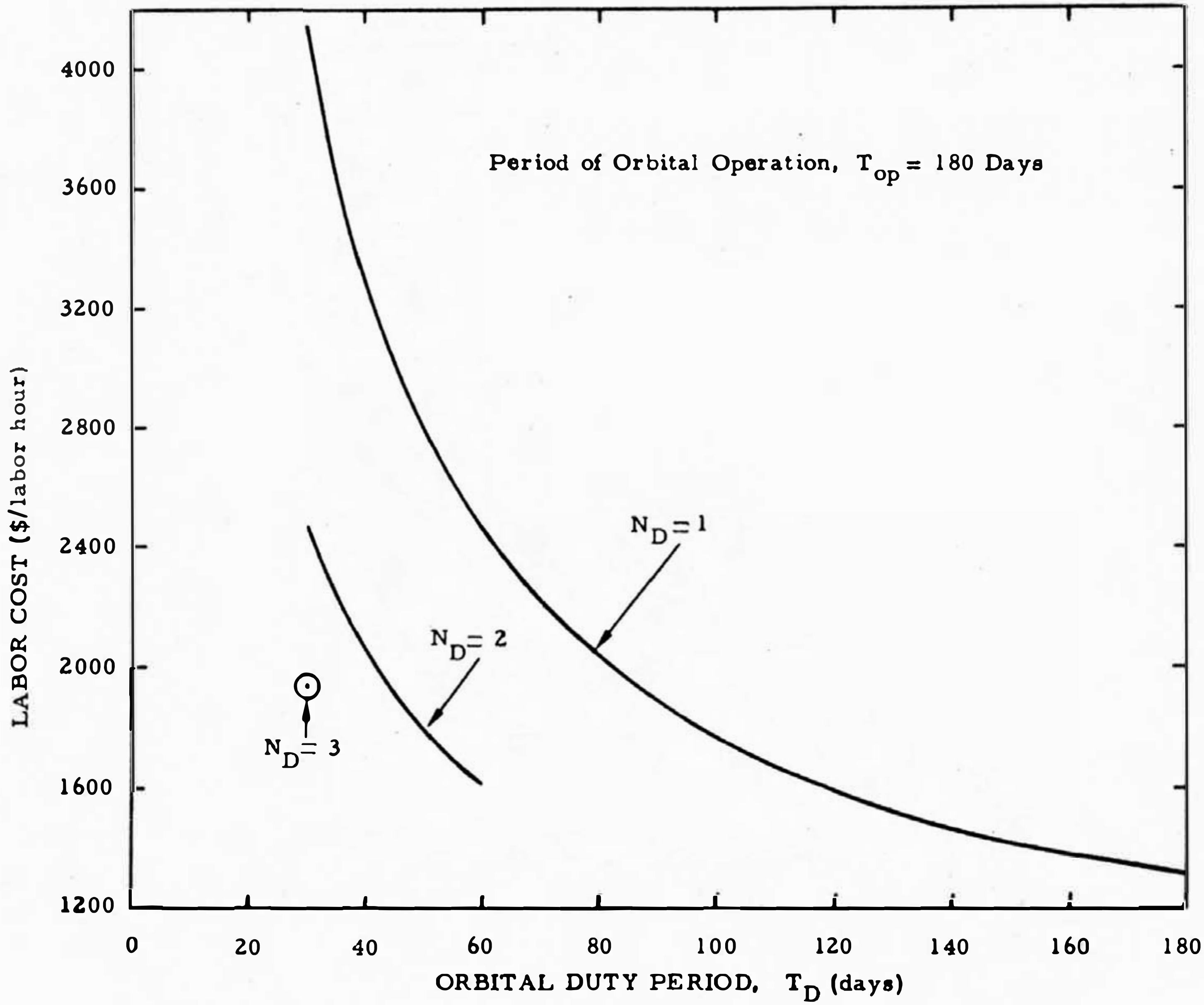


Fig. 16 Hourly Orbital Labor Cost for an Operational Period of 180 Days.

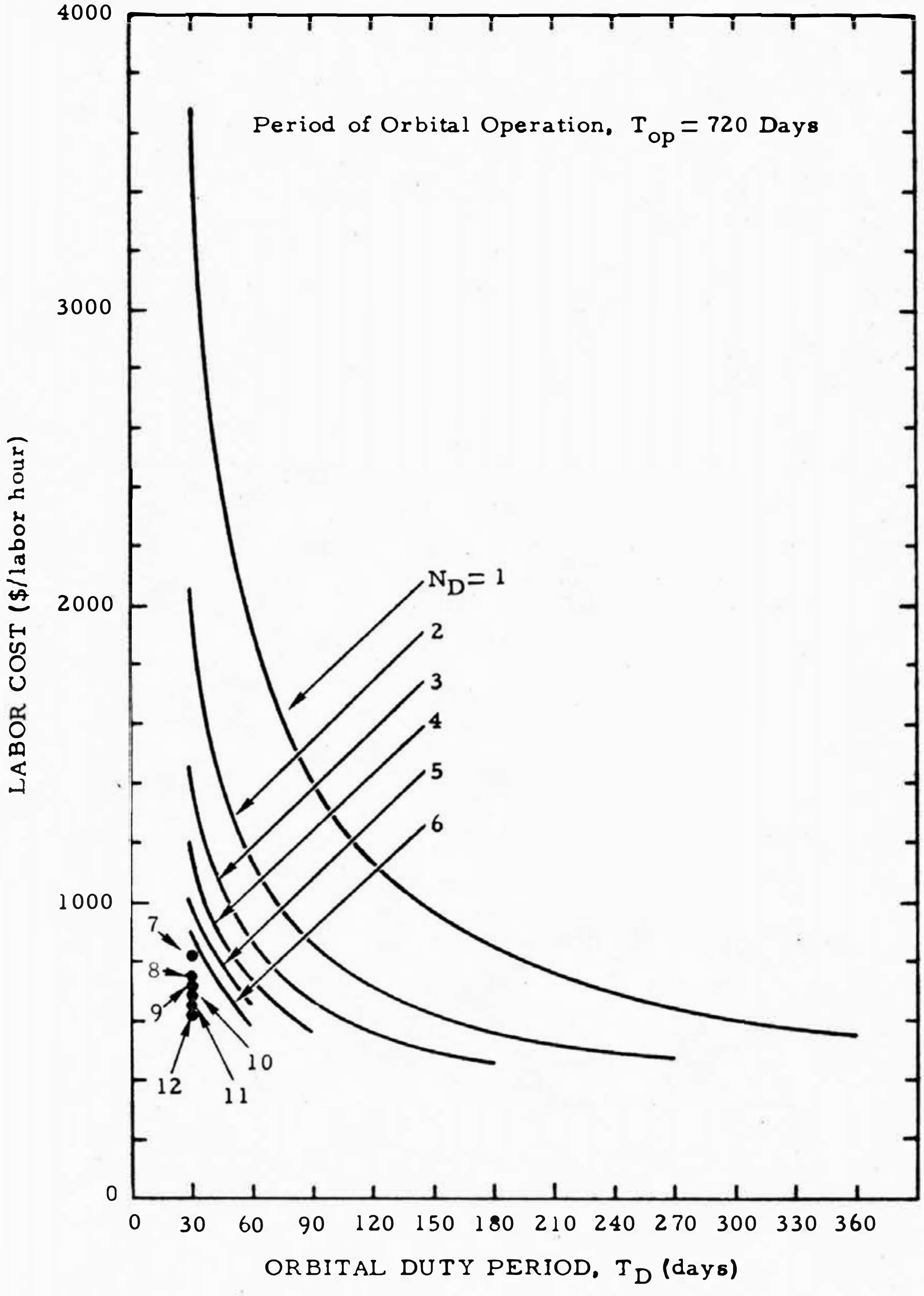


Fig. 17 Hourly Orbital Labor Cost for an Operational Period of 720 Days.

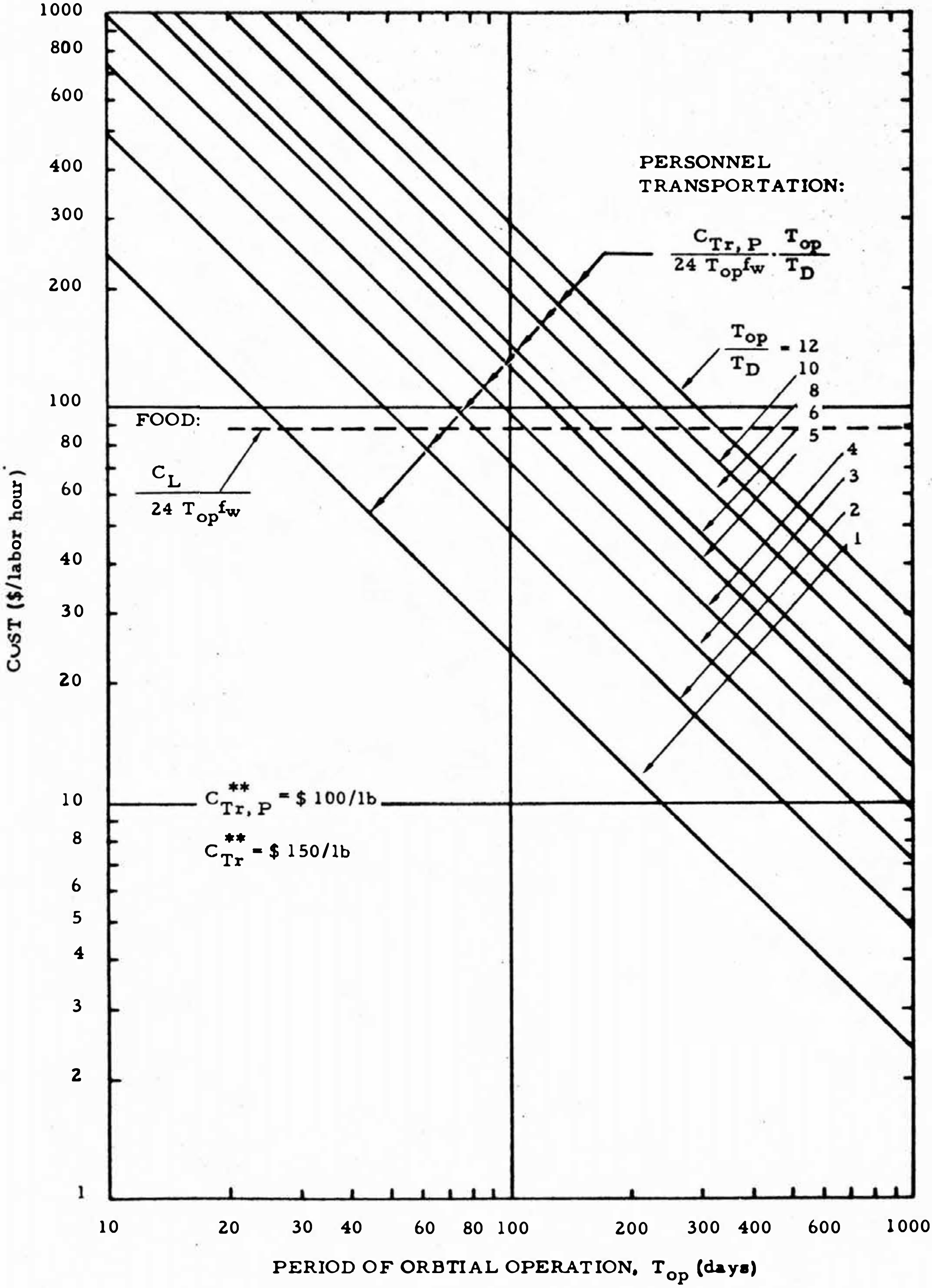


Fig. 18. Cost of Living and of Personnel Transportation.

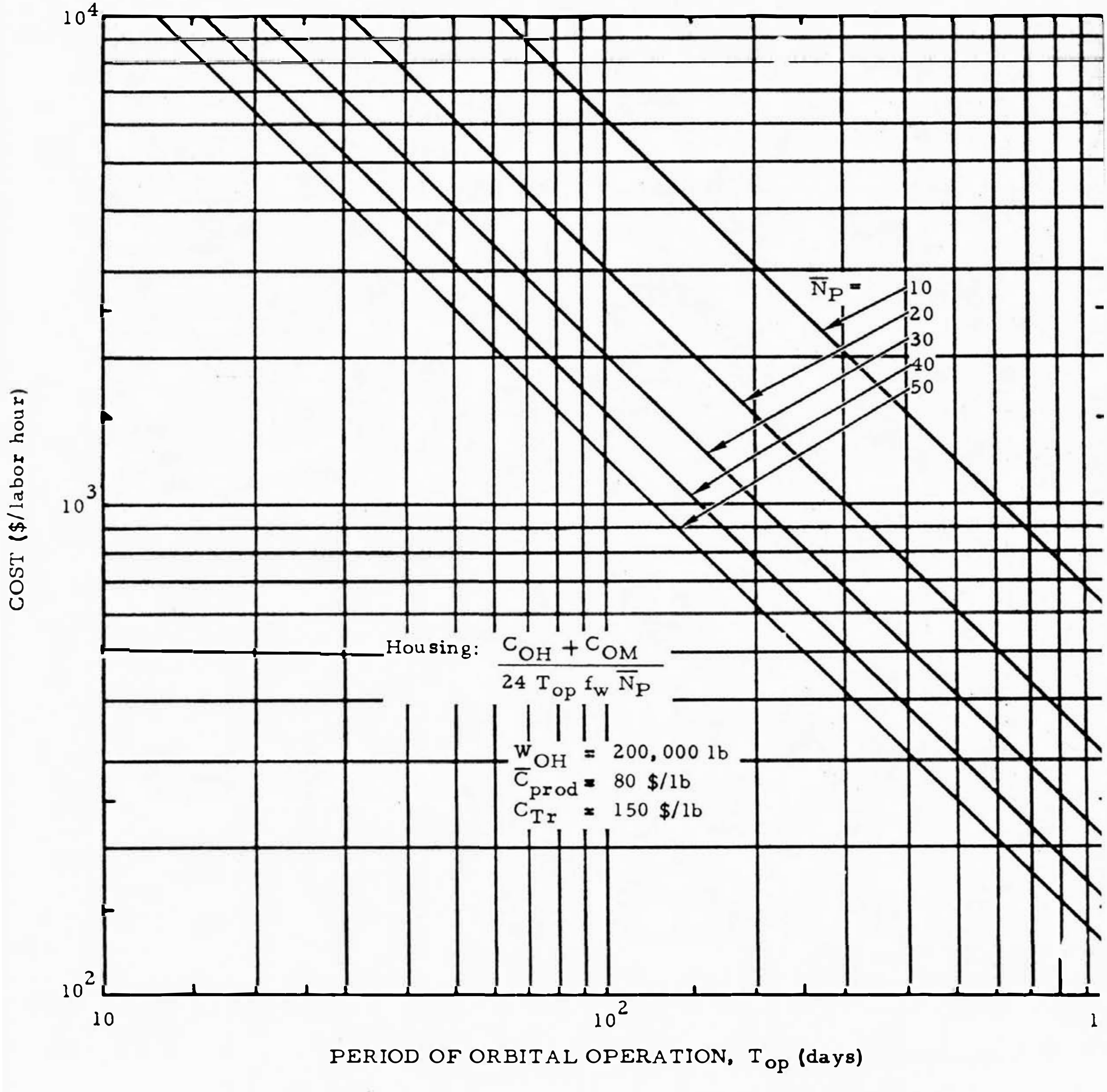


Fig. 19. Cost of Orbital Housing, its Maintenance and Associated Ground Tracking Operations.

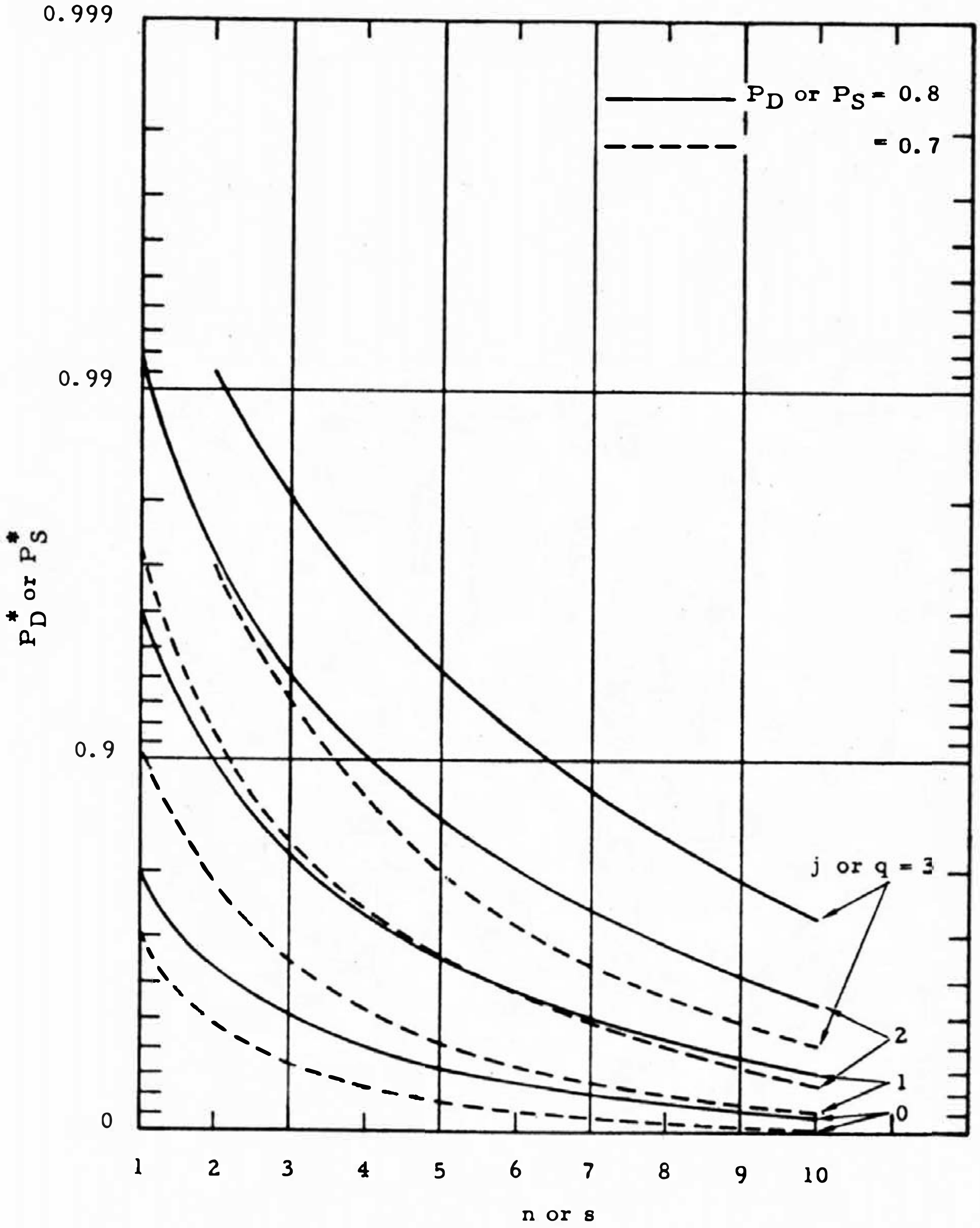


Fig. 20. Probability P_D^* or n Deliveries in n & j or Fewer Delivery Attempts and Probability P_S^* of s Successful Orbital Fuelings in s & q or Fewer Fueling Attempts.

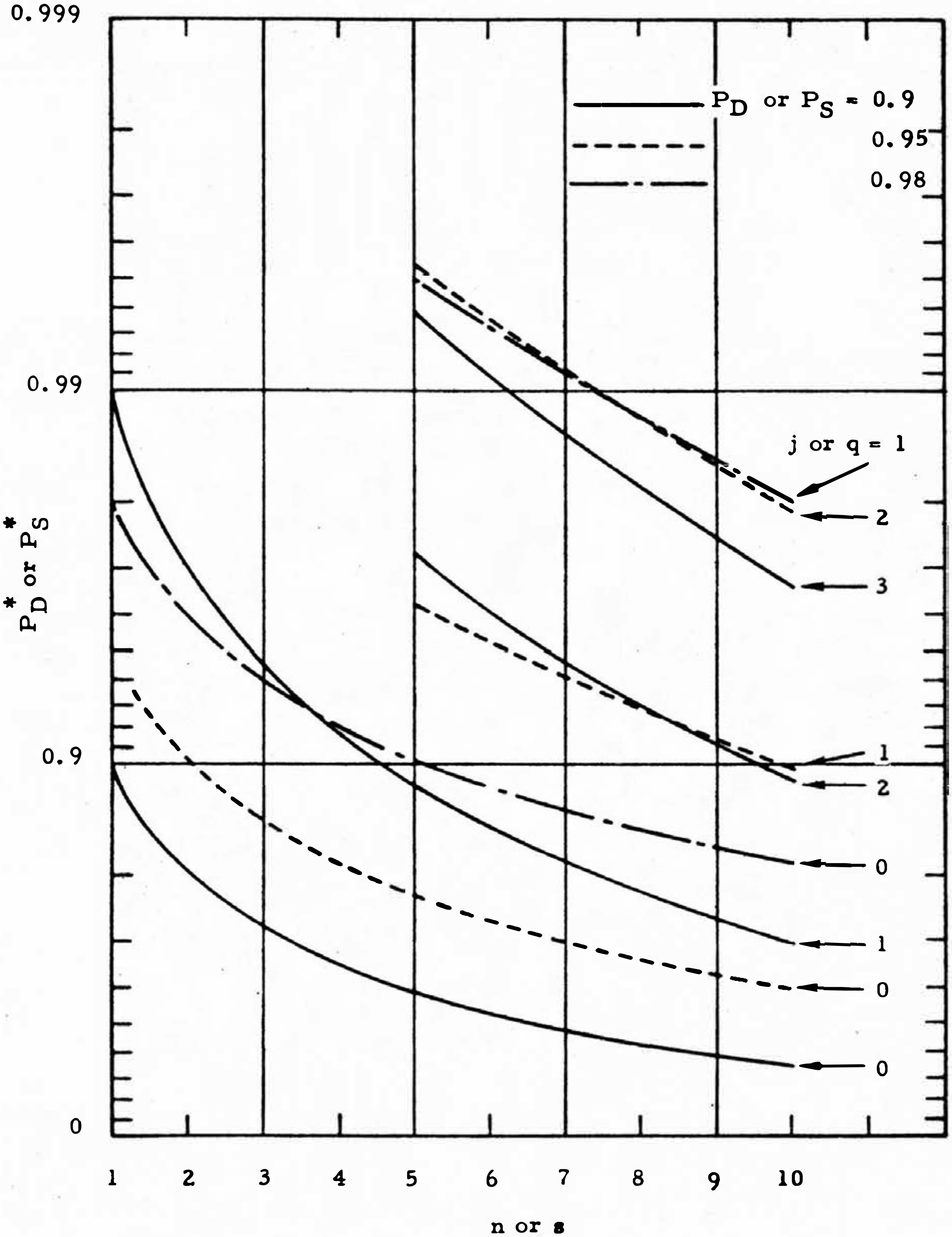


Fig. 21. Probability P_D^* of n Deliveries in n & j or Fewer Delivery Attempts and Probability P_S^* of s Successful Orbital Fuelings in s & q or Fewer Fueling Attempts.

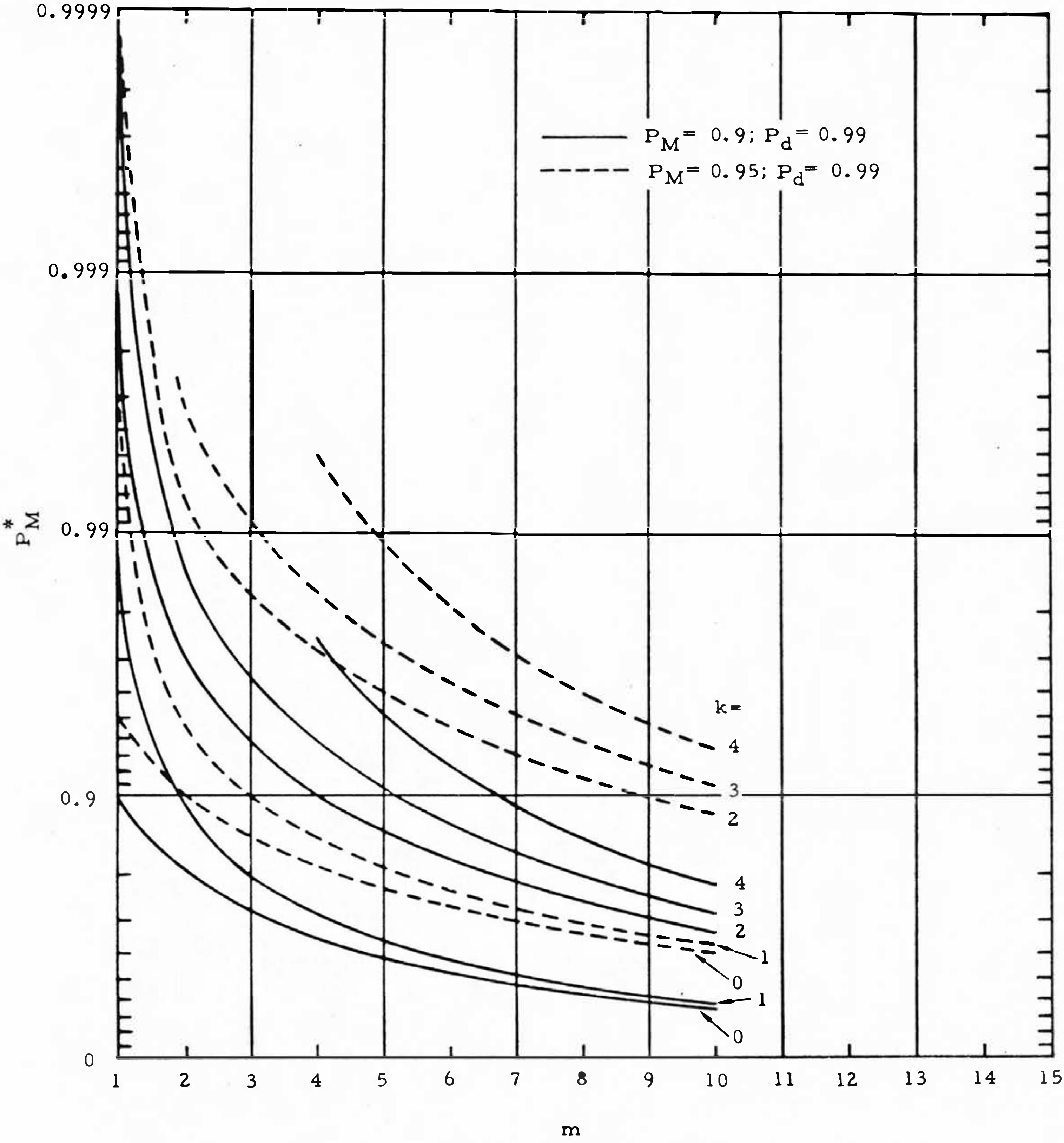


Fig. 22. Probability P_M^* of m Matings in m & k or Fewer Mating Attempts.

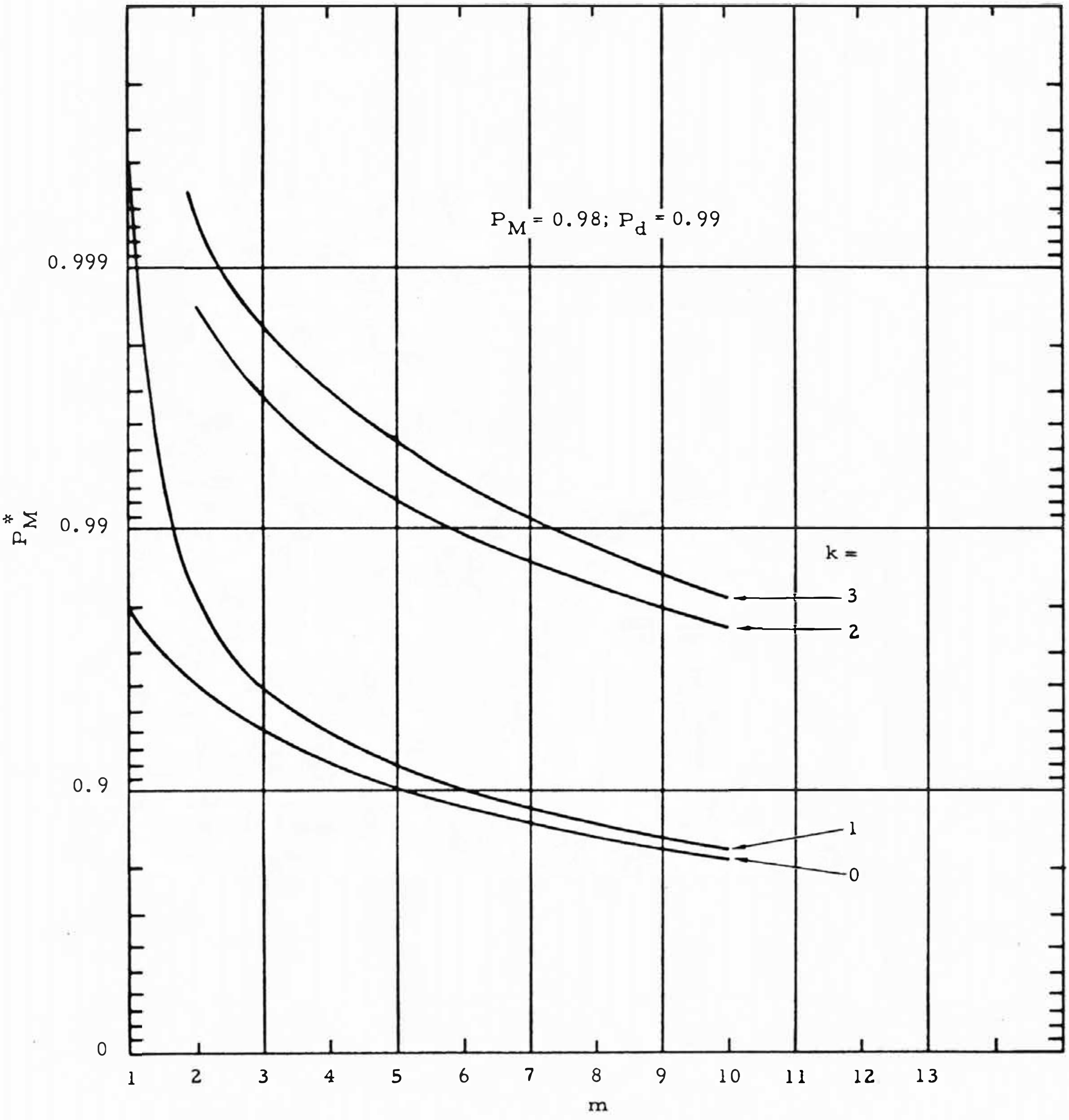


Fig. 23. Probability P_M^* of m Matings in m & k or Fewer Mating Attempts.

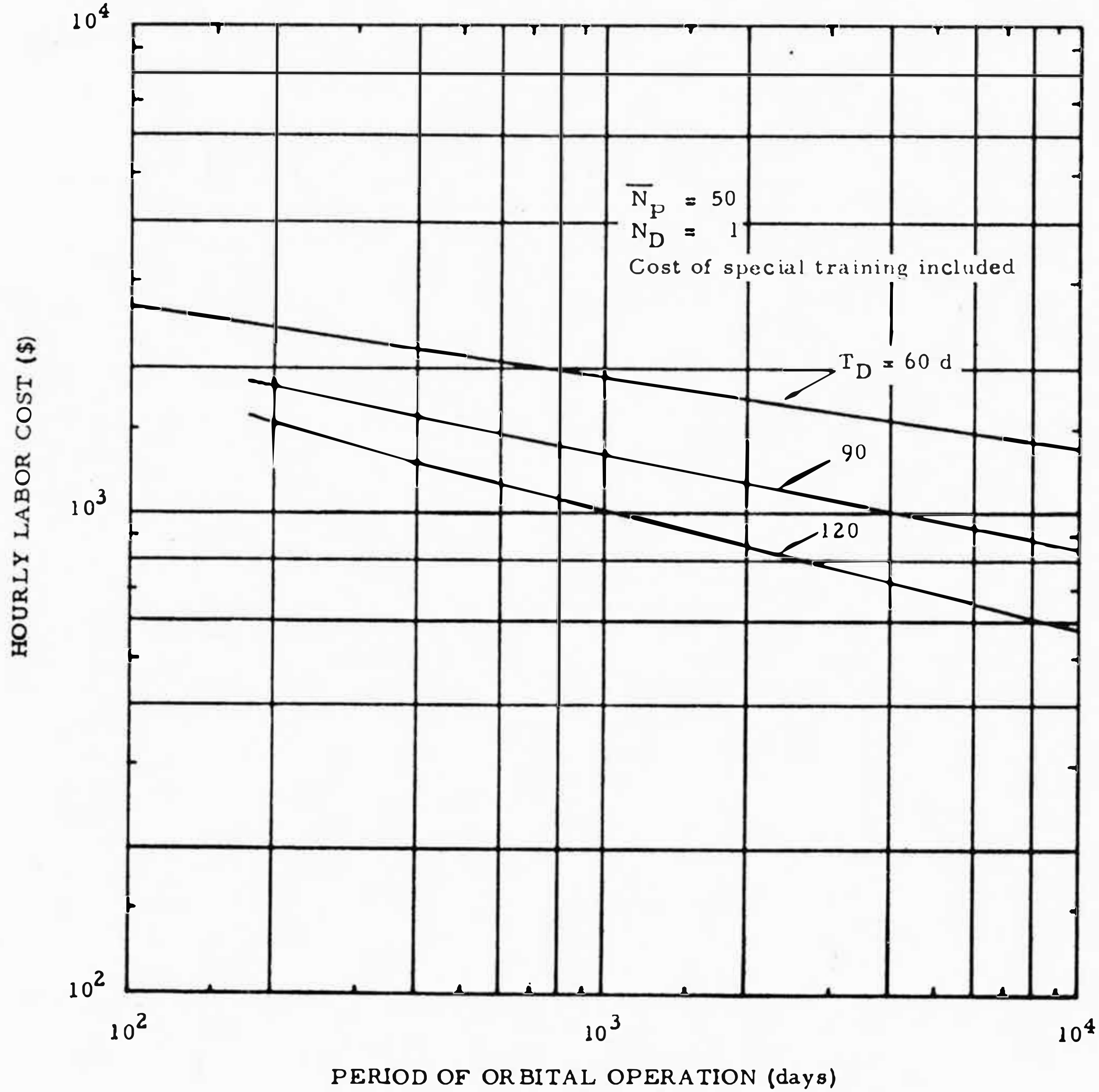


Fig. 24. Hourly Labor Rate Versus Period of Orbital Operation.

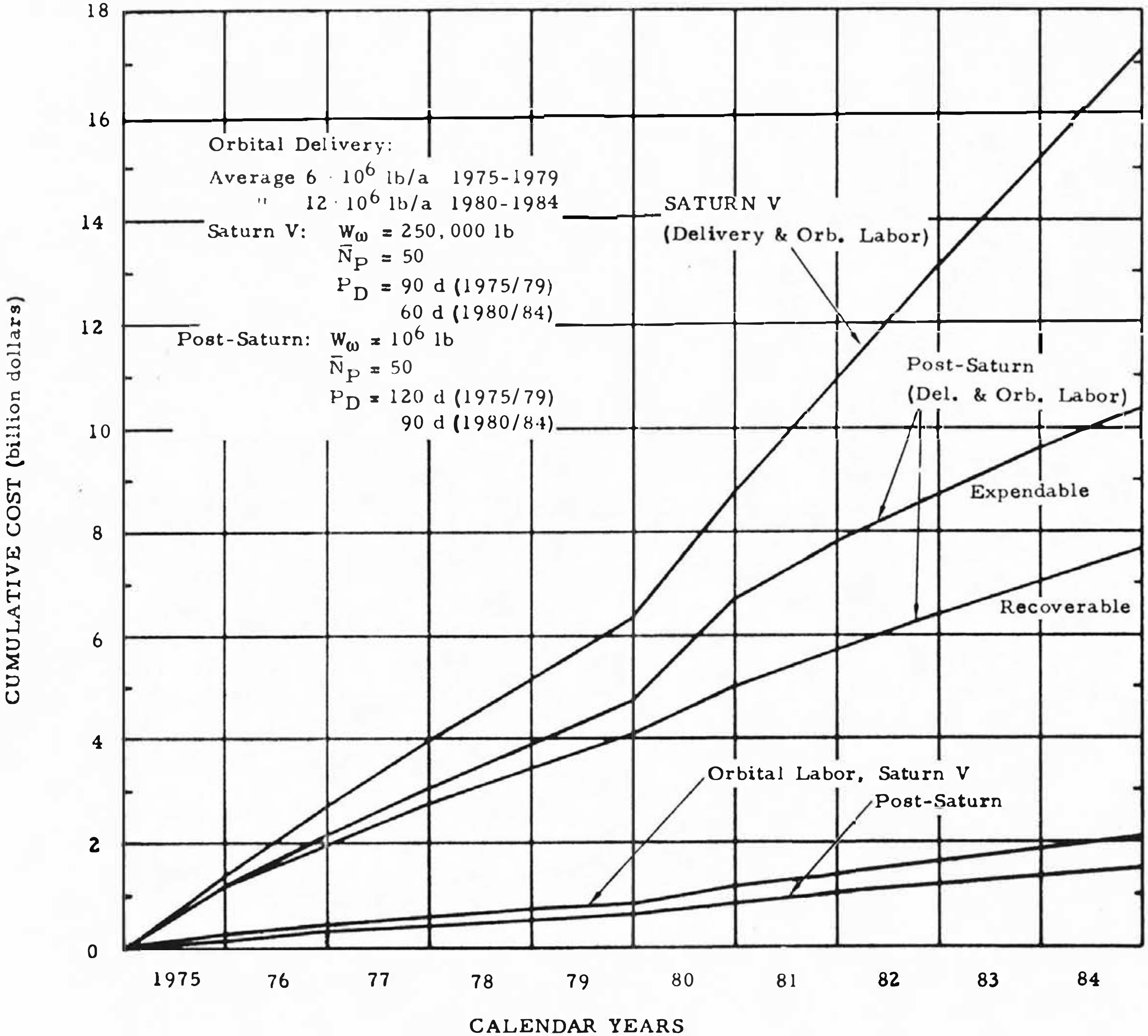


Fig. 25. Cumulative Cost of Delivery and Orbital Labor 1975-1980.

FIGURE 26 INTENTIONALLY OMITTED

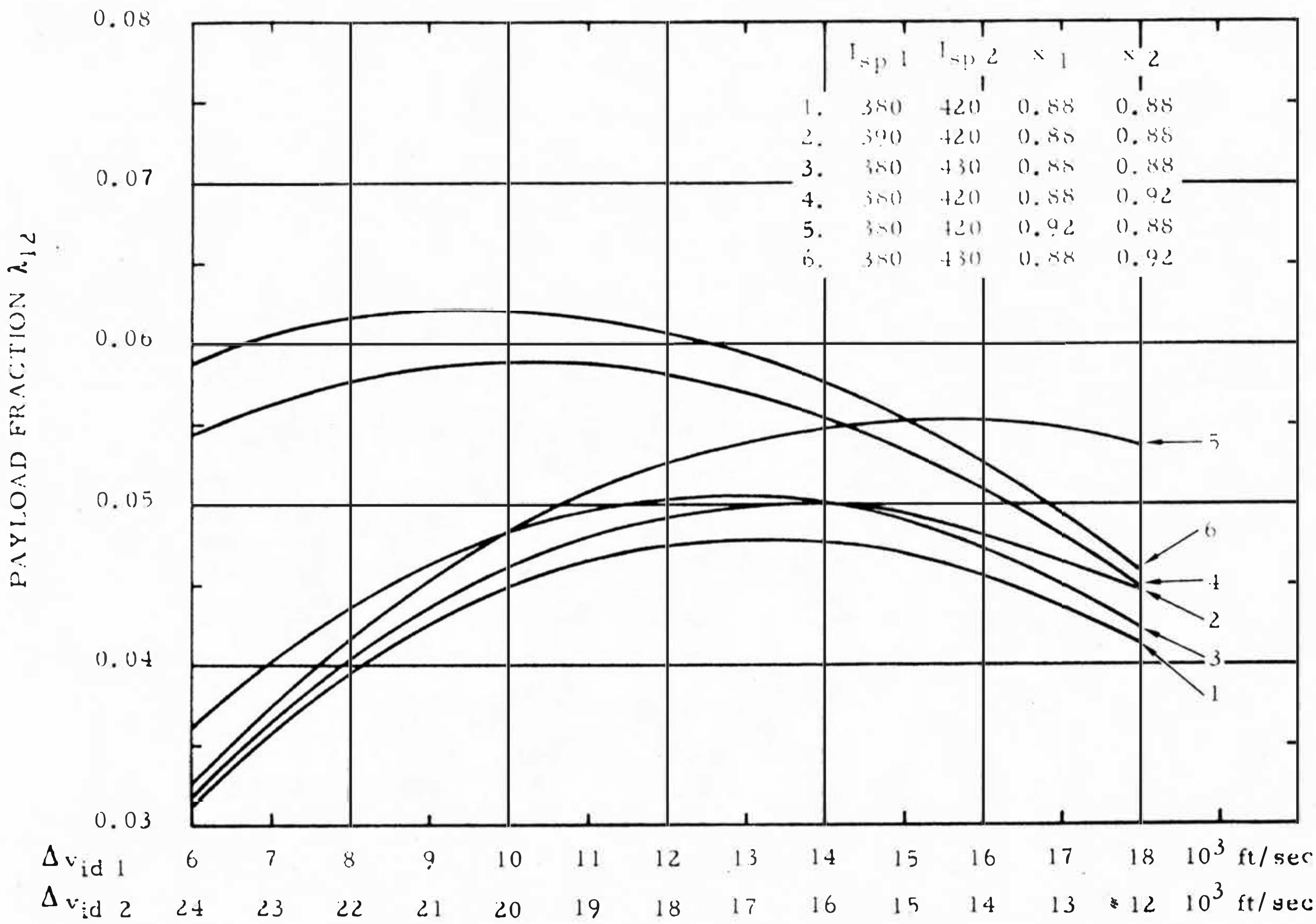


Fig. 27 2-Stage-to-Orbit. Conventional O_2/H_2.

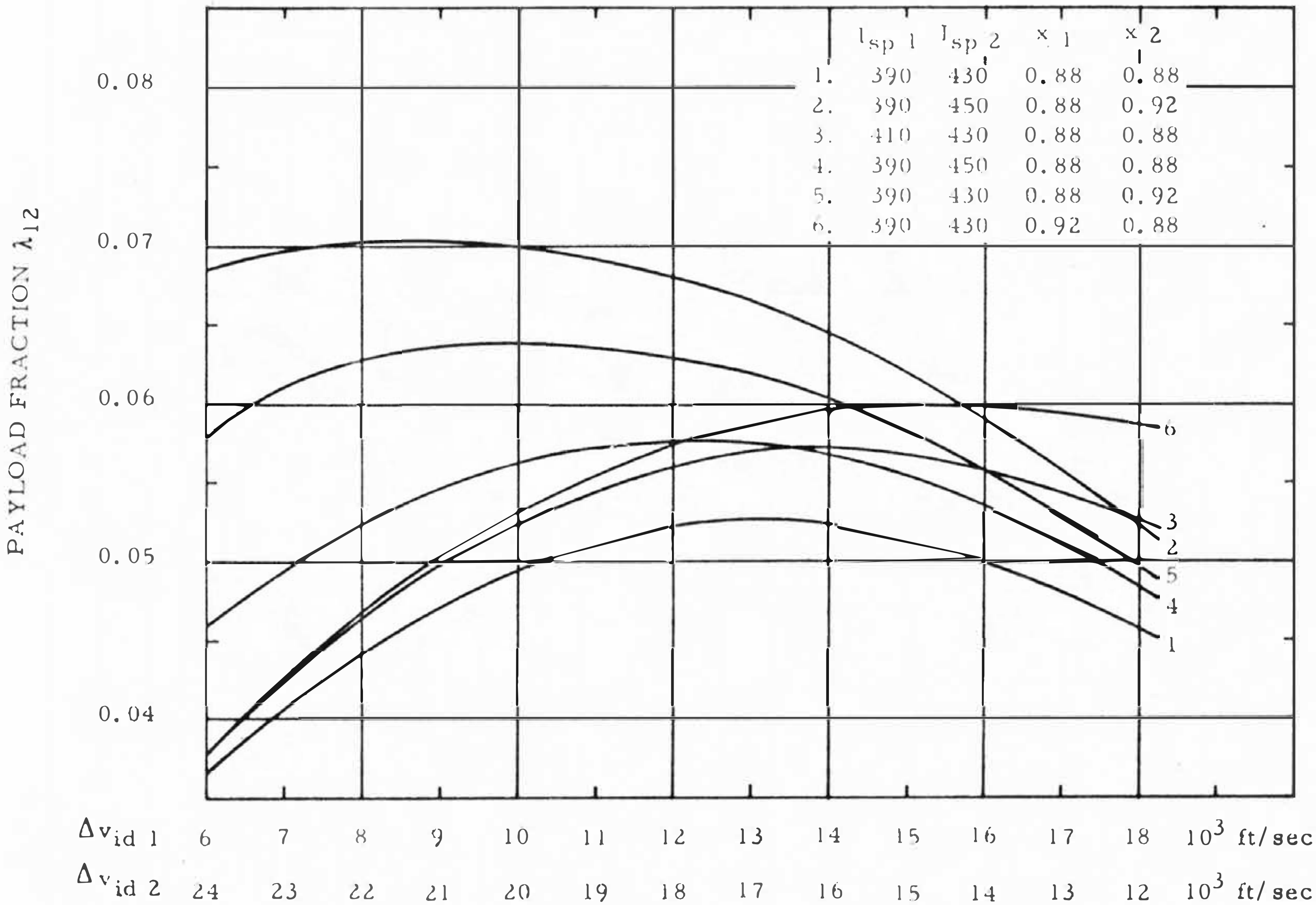


Fig. 28 Advanced O_2/H_2.

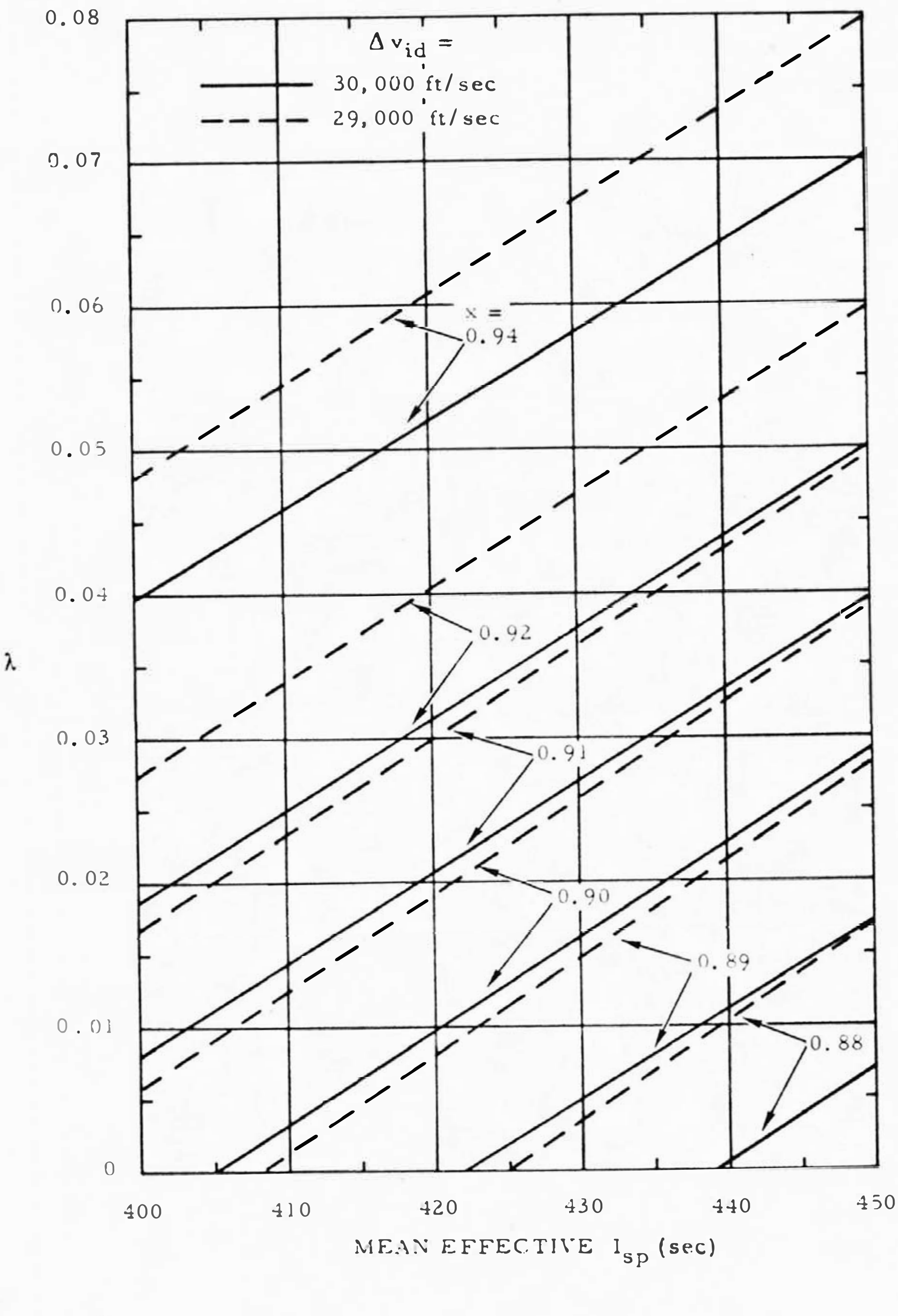


Fig. 29 Single-Stage-to-Orbit, O_2/H_2.

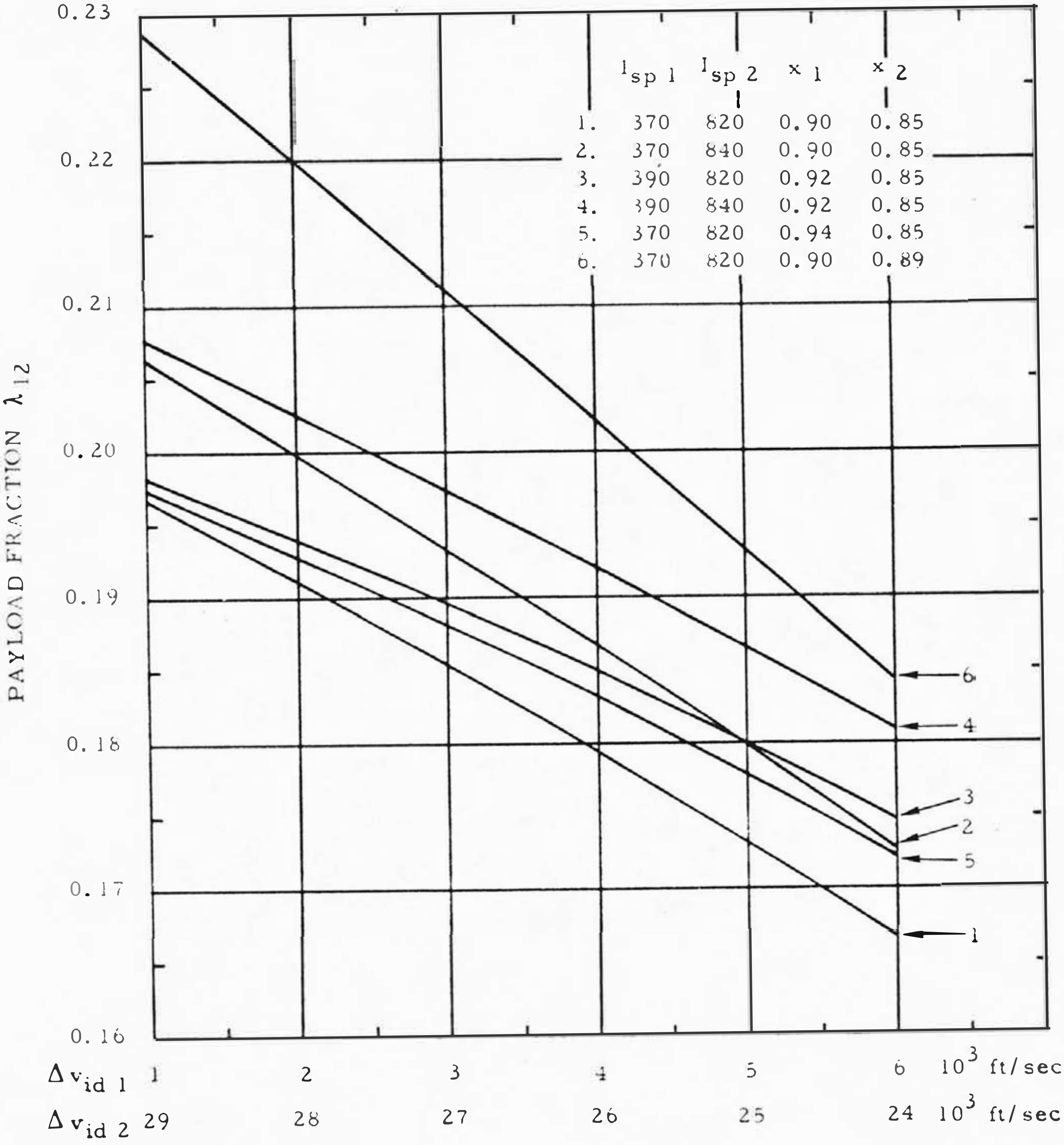


Fig. 30 Helios

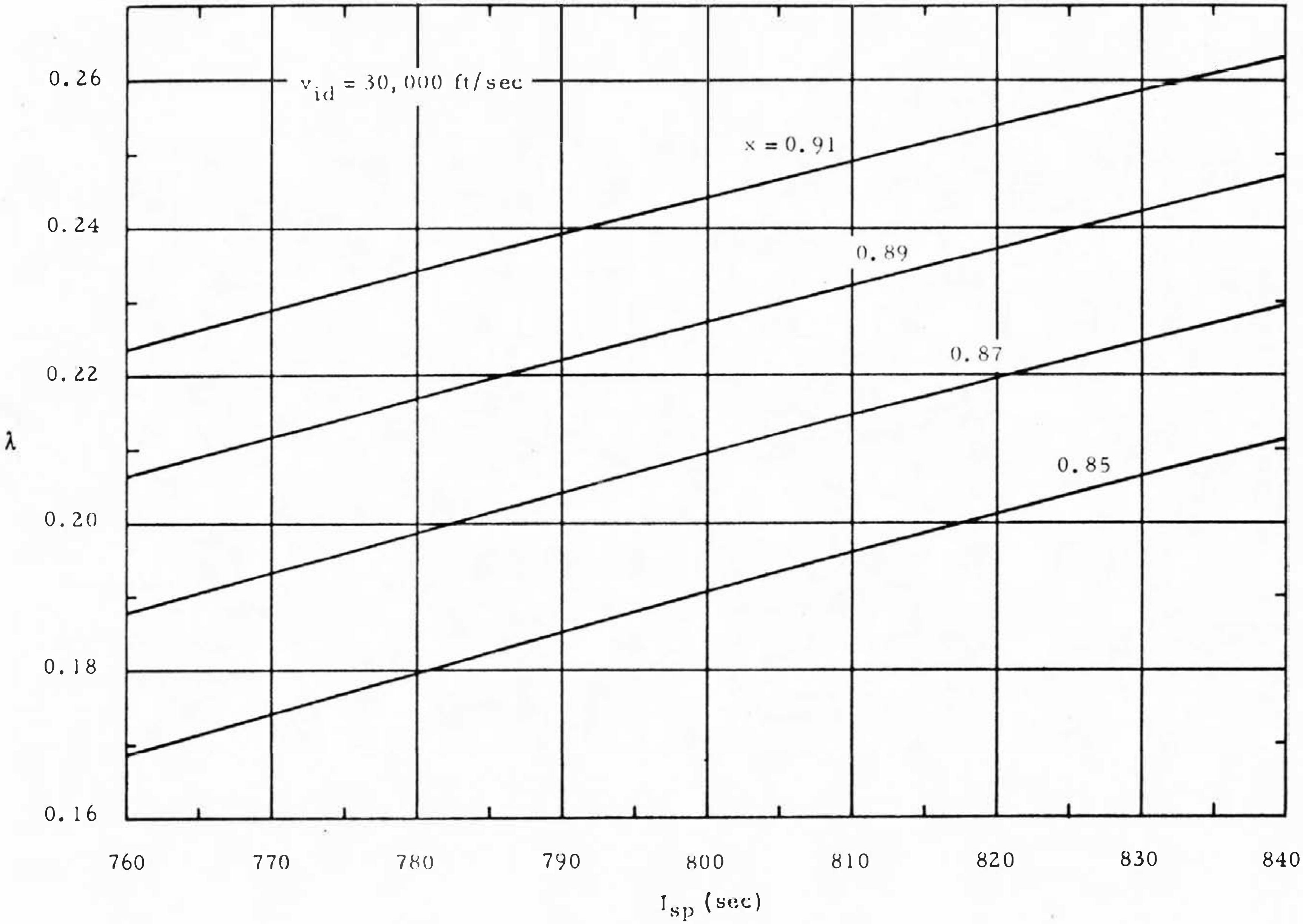


Fig. 31 Single-Stage-to-Orbit, Nuclear.

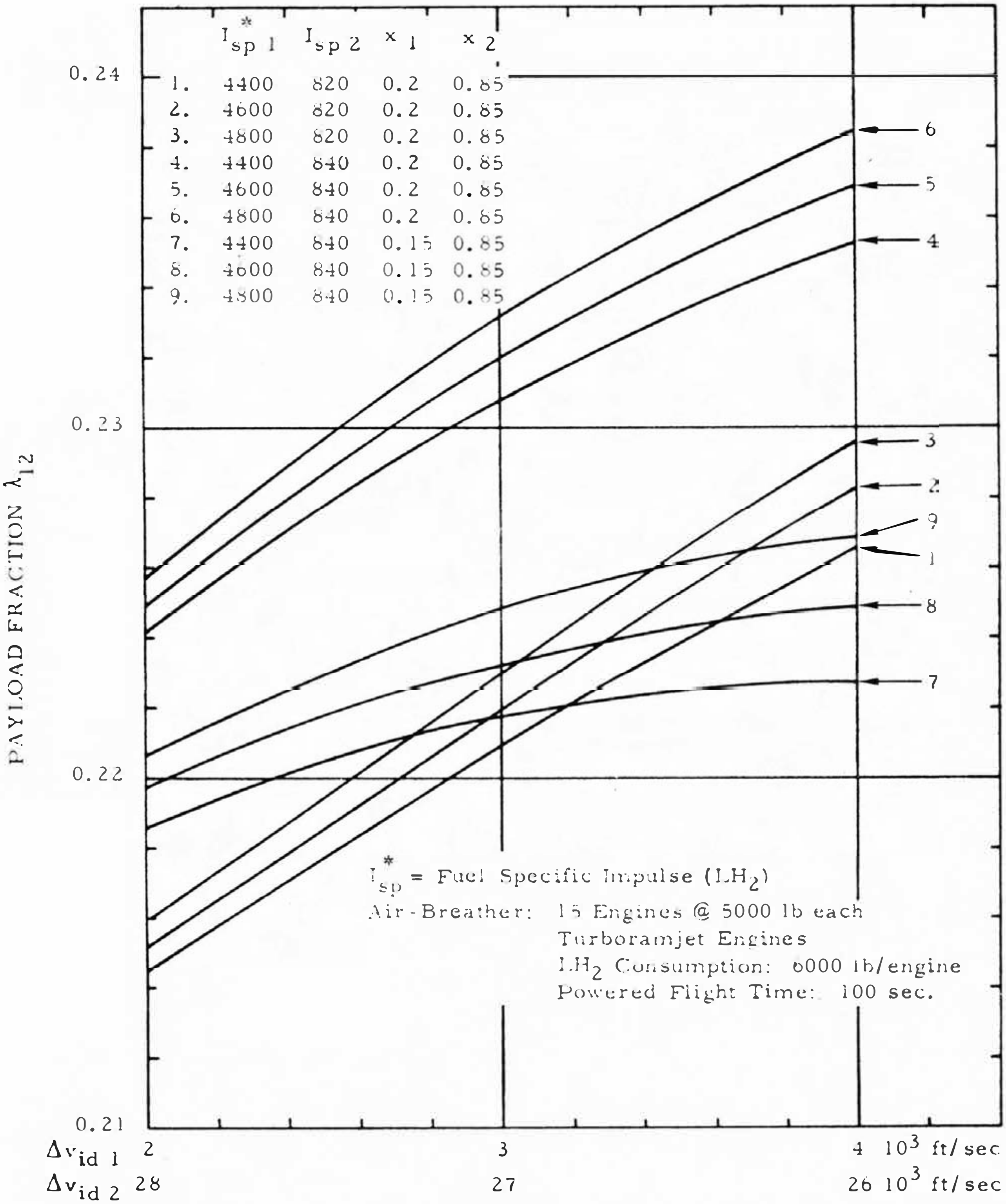


Fig. 32 Helios with Airbreathing Lift-off-Stage.

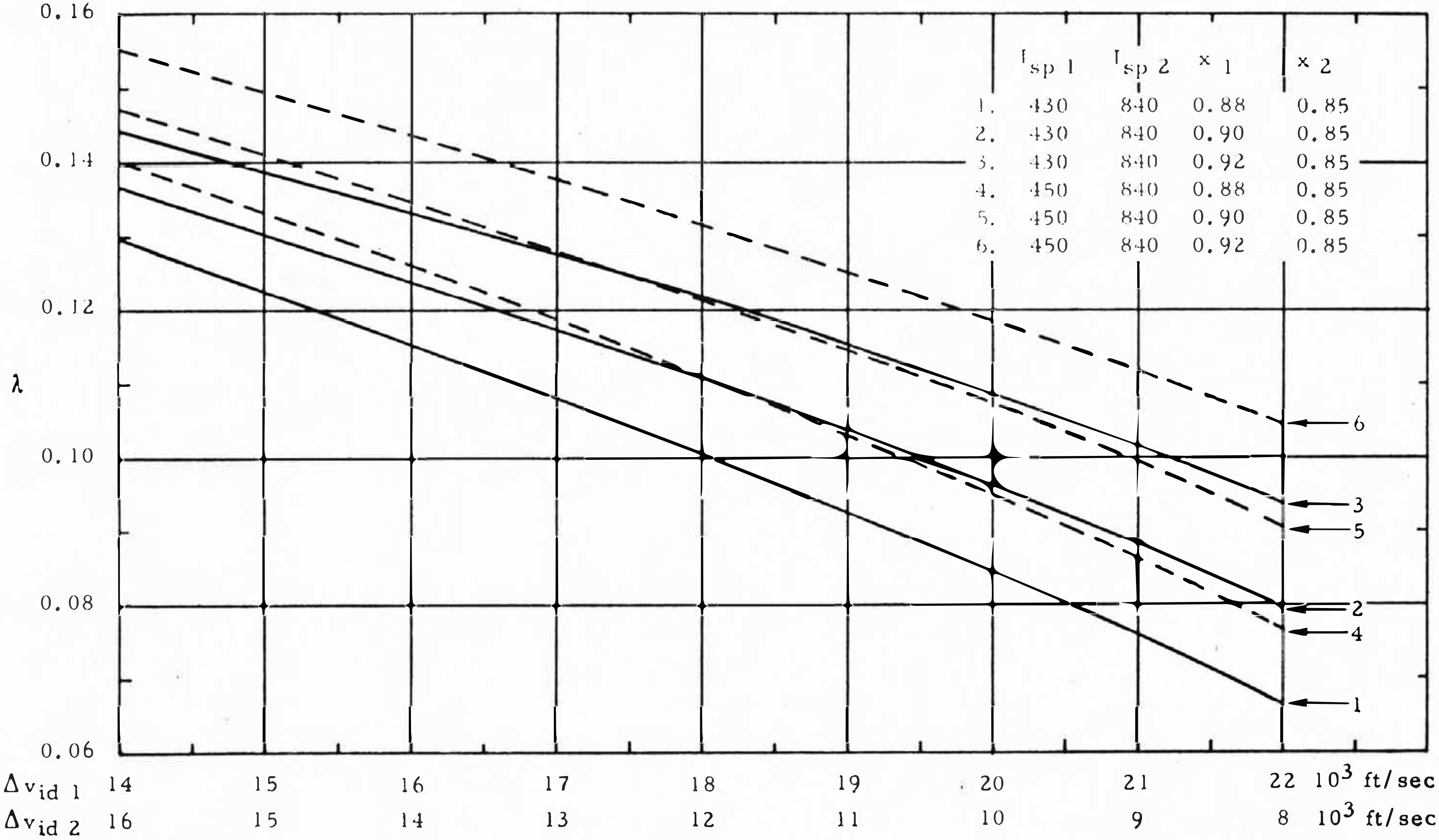


Fig. 33 Two-Stage Chemo-Nuclear ELV.

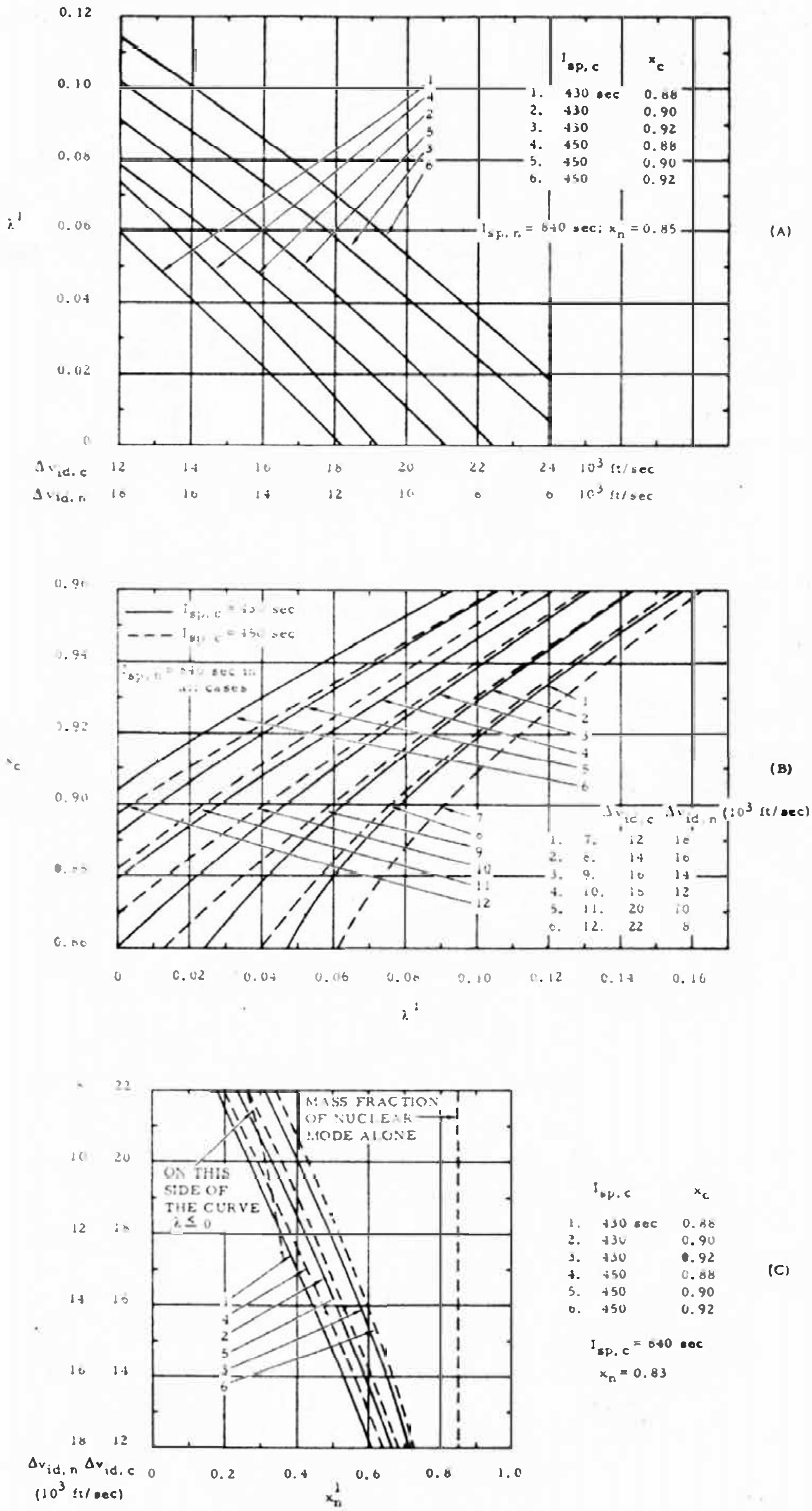


Fig. 34 Single-Stage-to-Orbit Chemonuclear ELV

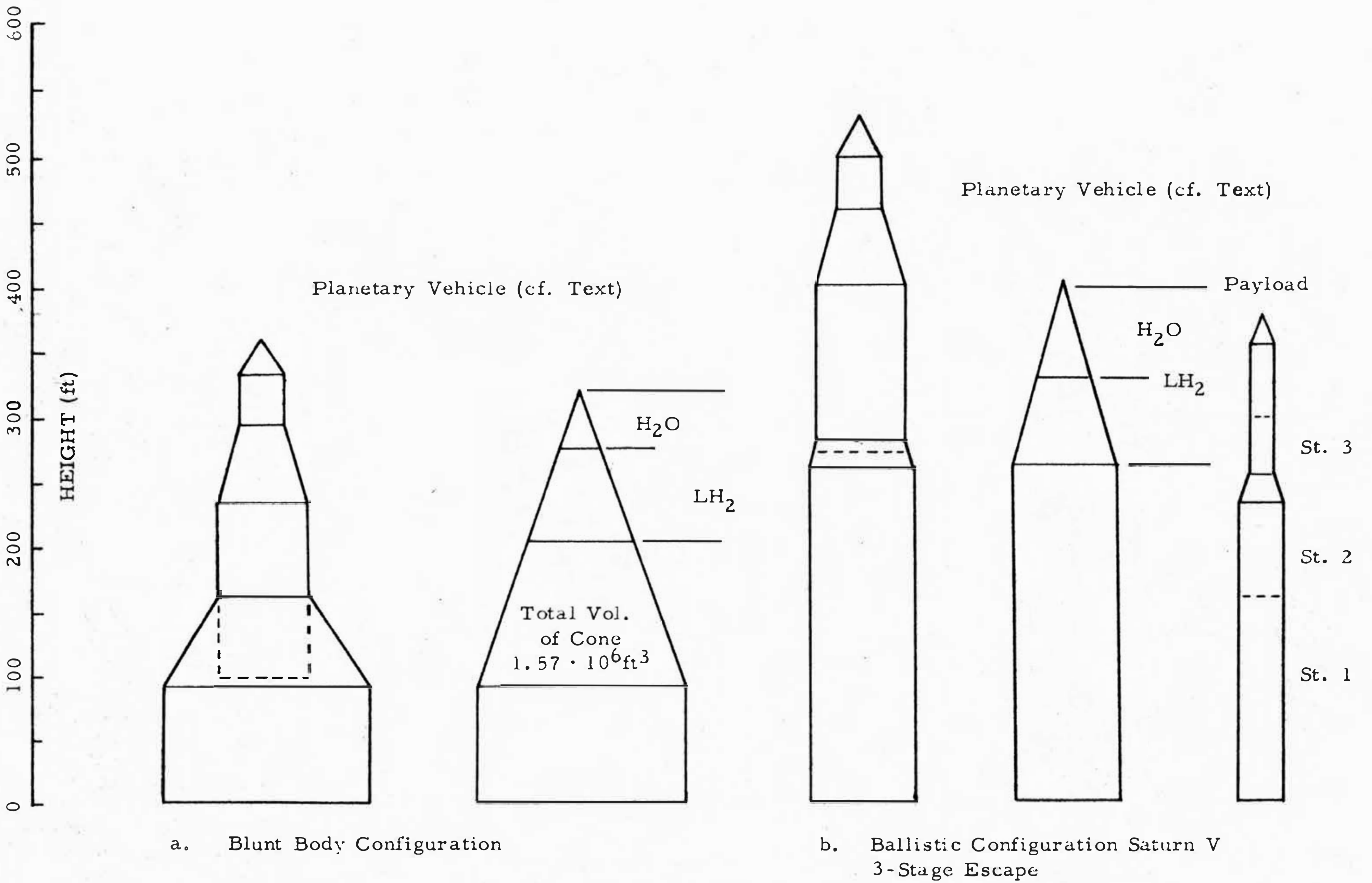


Fig. 35. Envelope of Two ELV Configurations in Relation to Various Payloads Sizes at 10^6 lbs Payload Weight. Three Stage Saturn V Configuration is shown for Comparison.

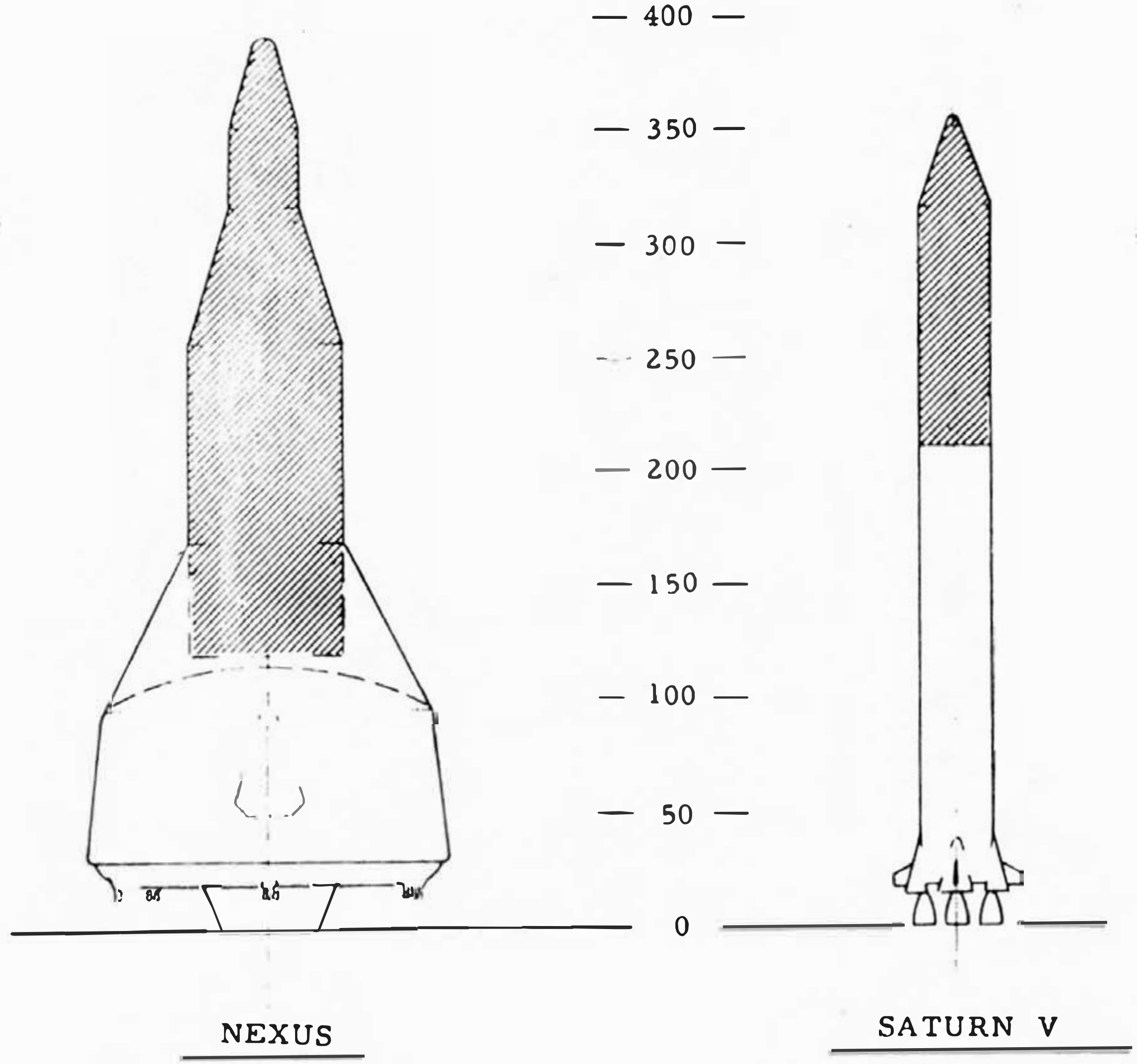


Fig. 36

VEHICLE COMPARISON

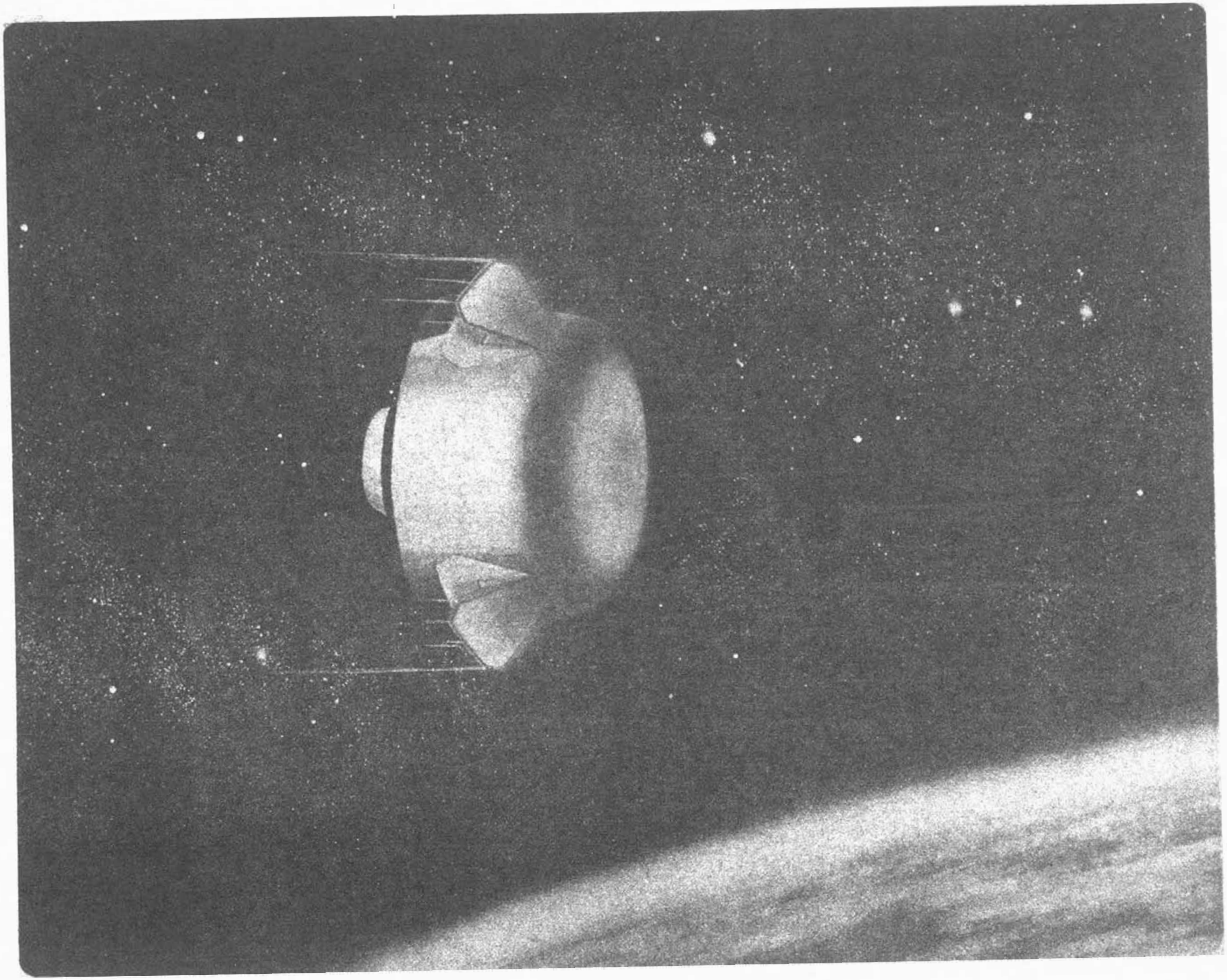
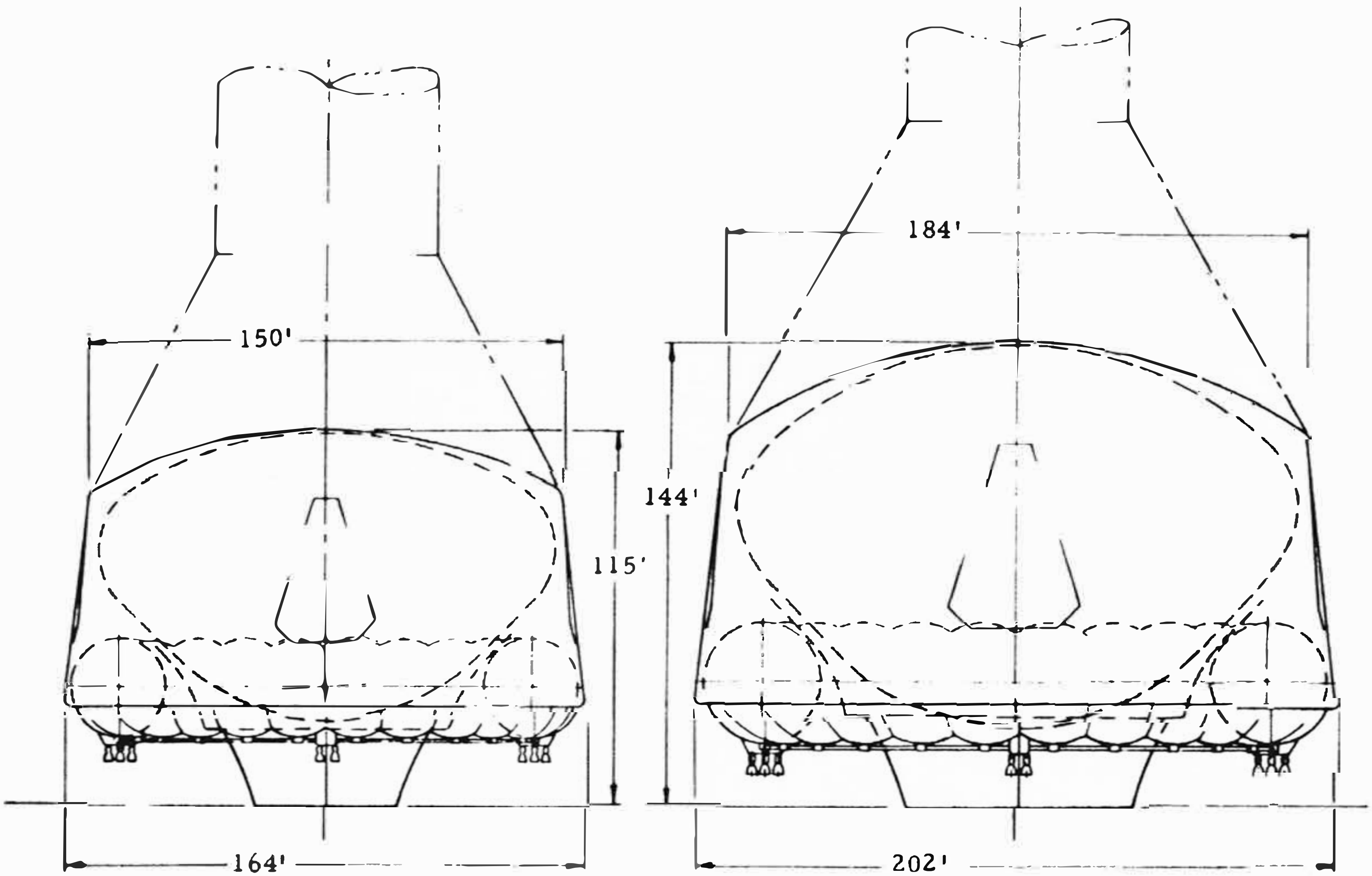


Fig. 37 NEXUS During Re-Entry Phase



DATA (million lbs)

24	GROSS WEIGHT (nominal)	48
32	TOTAL THRUST (nominal)	64
24	MAIN ENGINE (nominal)	48
8	CONTROL ENGINES (nominal)	16
1	PAYLOAD (nominal)	2

Fig. 38

NEXUS COMPARISON

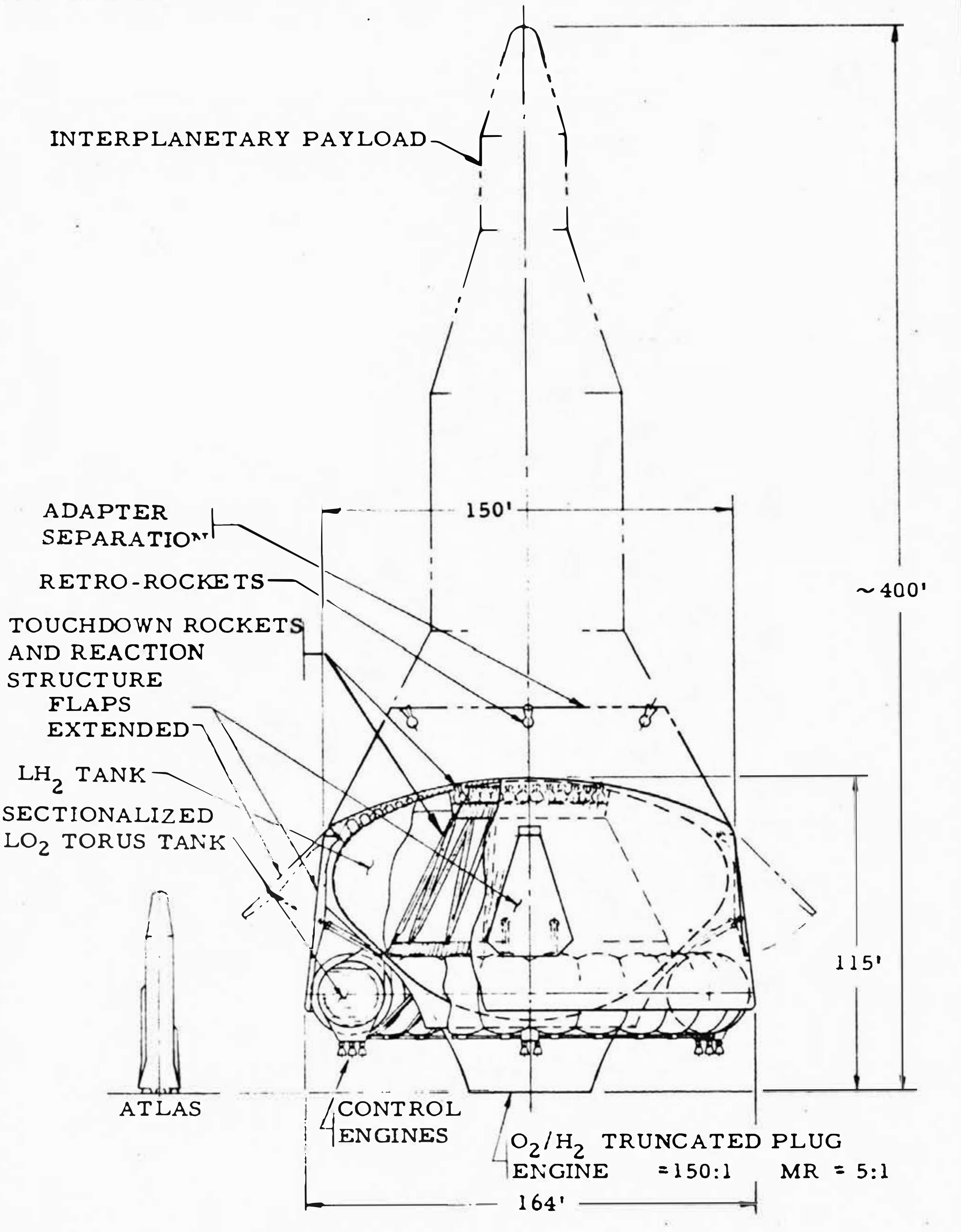


Fig. 39 — NEXUS - SINGLE STAGE BOOSTER SYSTEM —

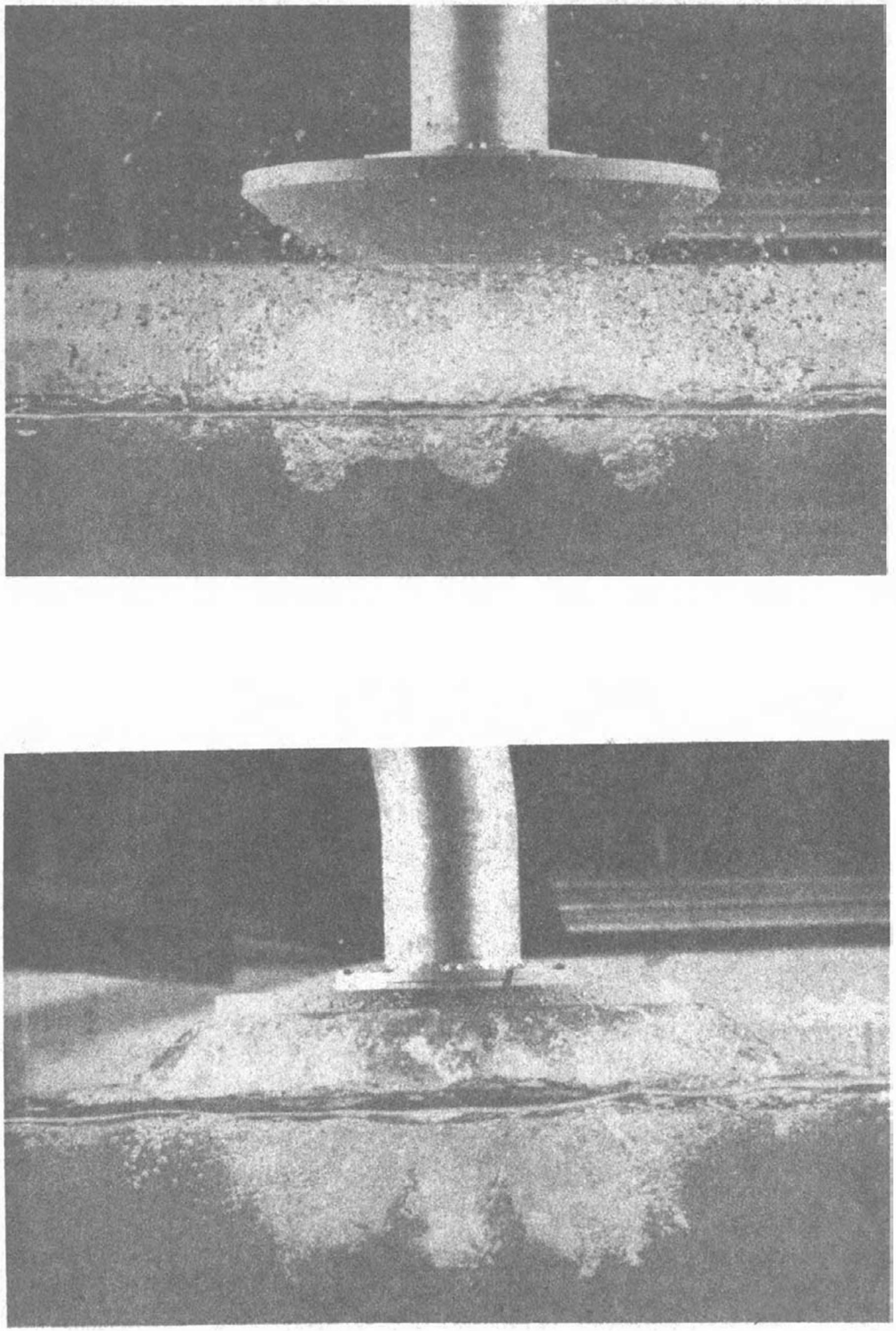


Fig. 40 Simulation of Water Landing of NEXUS with Retro-Thrust

FIG. 41

INTENTIONALLY

OMITTED

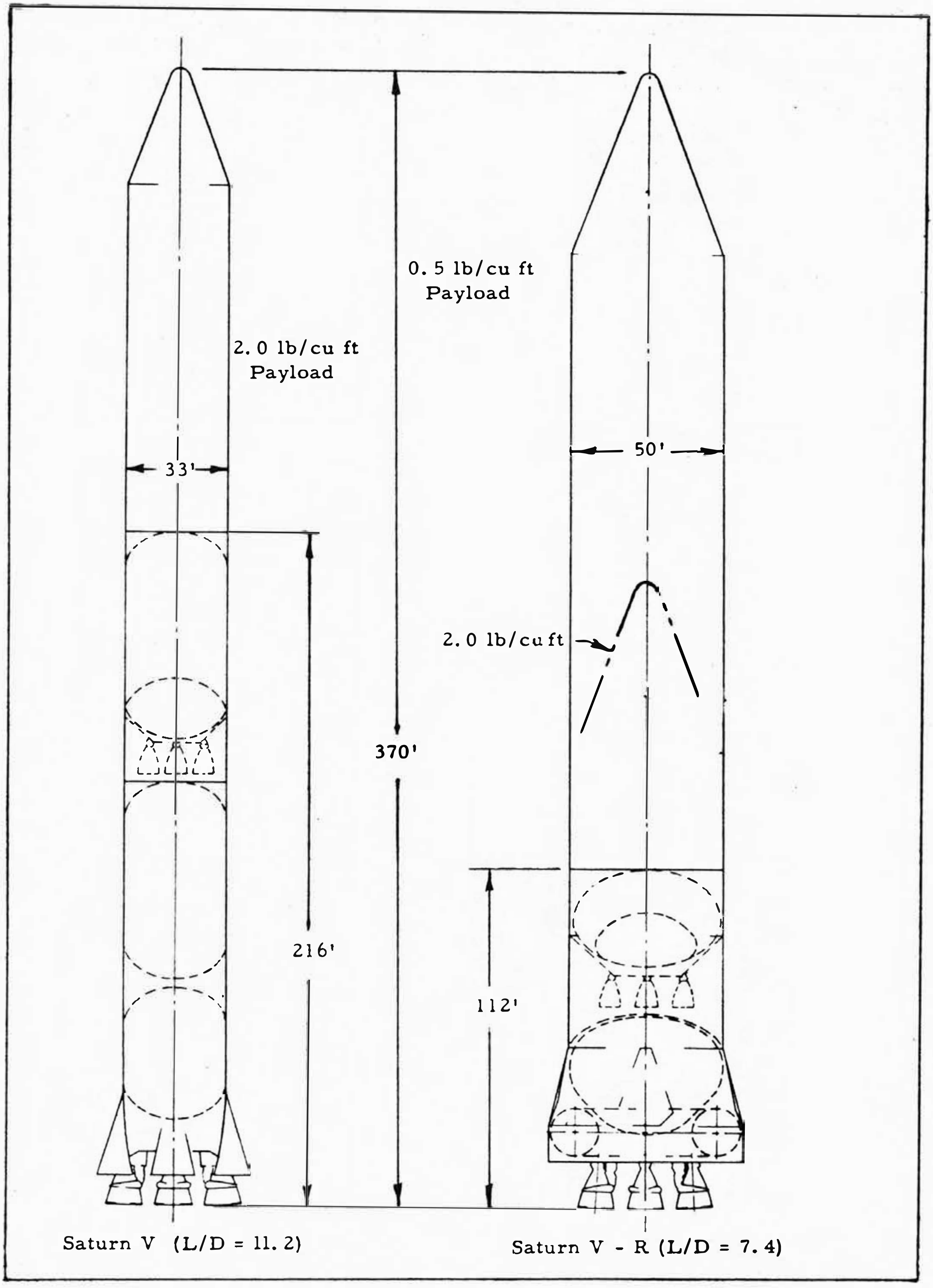


Fig. 42 Comparison of Present Saturn V With a 50 Ft Diameter Saturn V Version With Recoverable First Stage

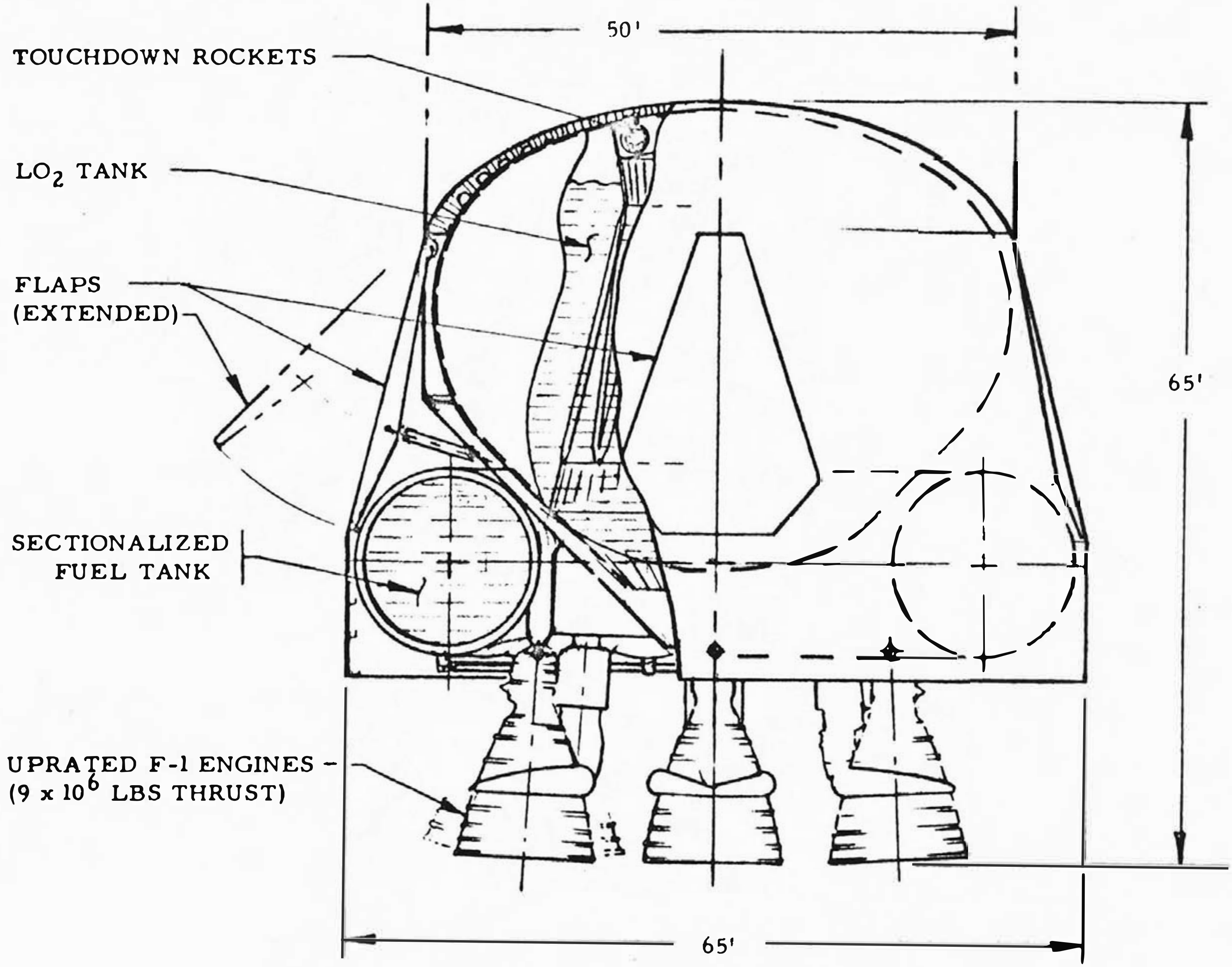


Fig. 43 SATURN V - R FIRST STAGE

GENERAL DYNAMICS

Convair Division