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SENSITIVITY OF SPACE SHUTTLE WEIGHT AND COST TO STRUCTURE SUBSYSTEM WEIGHTS

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16. Abstract

A study has been made which has established quantitative relationships between changes in space shuttle weights and costs with changes in weight of various portions of space shuttle structural sybsystems. These sensitivity relationships, as they apply at each of three points in the development program (preliminary design phase, detail design phase, and test/operational phase) have been established for five typical space shuttle designs, each of which was responsive to the missions in the NASA Shuttle RFP, and one design was that selected by NASA.

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FOREWORD

This is the final report of the "Study of Sensitivity of Space Shuttle Weight and Cost to Structure Subsystem Weights," prepared under Contract No. NAS 1-11438. Principal contributors to this study have been T. E. Wedge and R. P. Williamson with support in specific areas by D. W. Fellenz, C. C. Richie, M. C. Vaughn, S. G. Broadhead, H. F. Schmidt, H. H. Drosdat, C. J. Rudey, J. A. Dippel, and other members of the Space Shuttle team at LMSC.

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Section 1

INTRODUCTION AND SUMMARY

SCOPE AND PURPOSE

Sensitivities of Space Shuttle System weight and cost to variations in structural weights are critical parameters needed to compare design alternatives at all levels of vehicle system design. Applications vary from use in program risk evaluation, comparing major configuration alternatives, through system growth studies, needed to plan weight control programs, down to engineering tradeoffs at the most detailed level of design. The objective of this study is to evaluate system weight and cost sensitivities for the five representative Space Shuttle configurations illustrated in Fig. 1-1. A principal purpose in doing so is to enhance understanding of the range of variation of system sensitivity values and of the causes for these variations. Sensitivities vary widely from one case to another and depend significantly on the assumptions used when resizing the vehicle to retain compliance with performance requirements. Thus, it is important to the user of sensitivities to understand their derivation, and it is the purpose here to delineate all significant features of the methods of derivation used.

Each of the five configurations in Fig. 1-1 is designed to meet the performance requirements imposed by the Space Shuttle Request for Proposal (RFP), issued by NASA on 17 March 1972. Configurations C and D utilize reusable solid rocket motors (SRMs), which burn in parallel with the orbiter main engines, and an expendable external tank to carry the hydrogen/oxygen propellants for these main engines. These represent the type of configuration specified by the RFP. Configuration D employs a delta-wing orbiter, the type favored by NASA and proposed by the potential contractors in response to the RFP, while Configuration C uses a delta-body orbiter of a type studied for the past several years by LMSC. Configurations A and B use larger versions of the

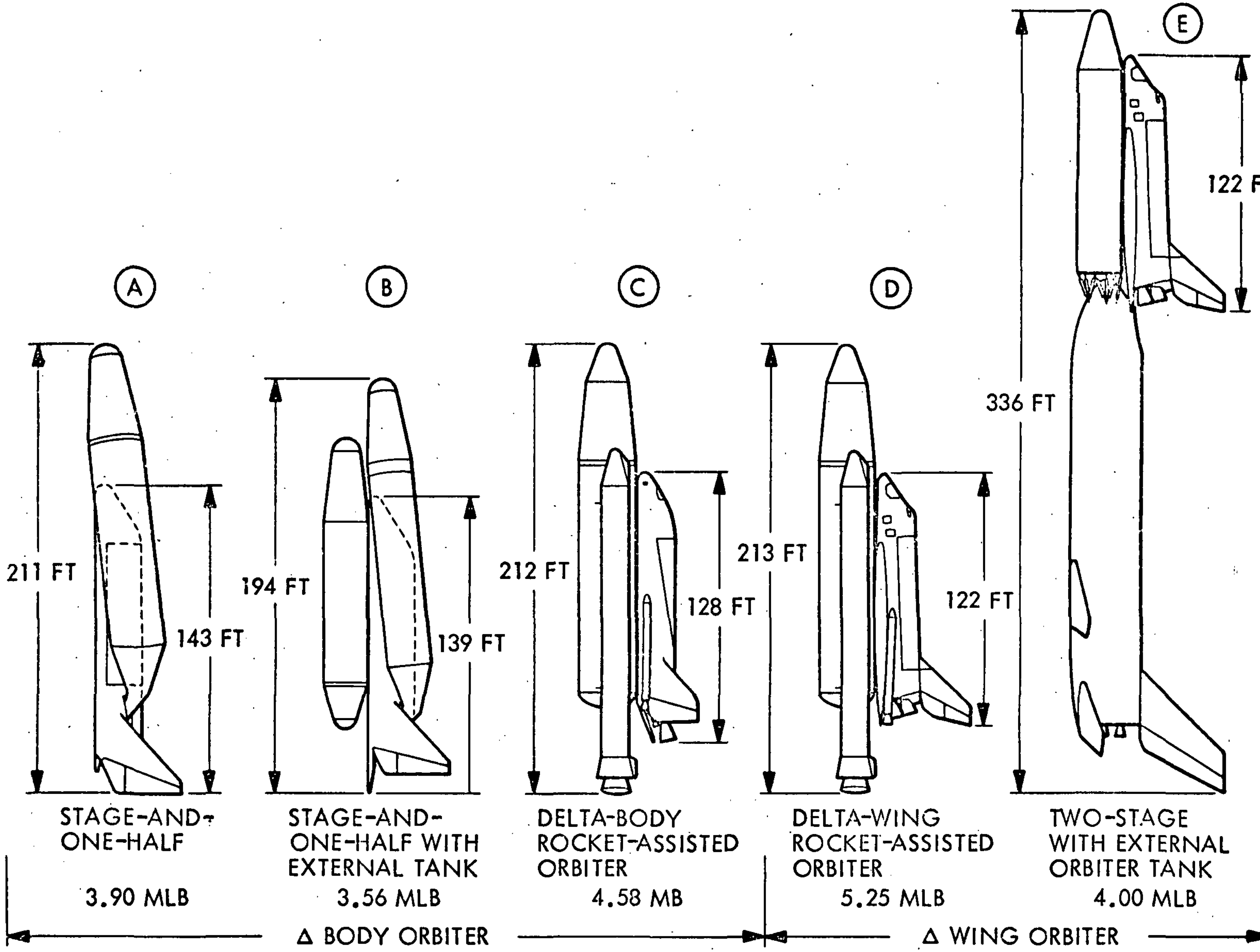

Figure 1-1 Five Configurations Same Performance

delta body configuration, incorporating a sufficient main engine complement (11 engines in A and 9 in B) to provide the total ascent propulsion thrust. Configuration A employs, in addition to droptanks, reusable internal tanks in the orbiter and is the latest stage-and-one-half version resulting from extensive LMSC study of this system. Configuration B is also a stage-and-one-half but carries propellants needed by the orbiter (after the droptanks are staged) in an external tank which is carried to orbit. As in Configuration C and D, the external tank incorporates a retro system to enable it to be deorbited on the first orbital revolution. Configuration E uses a flyback heat-sink booster, an external orbiter tank, and an orbiter essentially identical to that of Configuration D.

After definition of the baseline designs, the question is asked: "What would be the system weight and program cost effects, at various specified phases of the development program, of a change from the baseline in the estimated structural subsystem weight of any one of the major vehicle elements (orbiter, orbiter external tank, droptank, SRM booster, or flyback booster, as applicable)?" The following three phases of the development program are specified, giving consideration to constraints imposed on redesign when the weight change occurs: (1) preliminary design phase, when all system elements and systems can be resized to establish a new "optimized" configuration (2) detailed design phase, when it is assumed that, while any required changes are made to meet all performance requirements, only one vehicle element is resized to maintain ascent performance capability, and (3) test/operations phase, when no redesign is allowed but a payload capability loss results from an overweight vehicle element. Thus the problem of extimating sensitivities can be considered a problem of predicting the behavior of the Space Shuttle program organization, with respect to vehicle redesign, if weight trouble is encountered. An estimation of the cost effect of payload capability loss goes even beyond that organization, to consideration of the cost of payload system redesign.

BASELINE WEIGHTS

Weight data for major elements of each of the baseline configurations are summarized in Table 1-1. In designing each of these vehicle systems, the payload requirement of 40K to 90-degree inclination (polar) was the most severe requirement of the three missions specified in the RFP, and this mission was used to size the ascent propulsion systems and tankage.

The fact that Configuration B is somewhat lighter than Configuration A results only slightly from the better mass fraction of external tanks compared to internal tanks. Most of the effect comes from secondary weight savings in main engines, on-orbit and retro propellants, thermal protection and landing gear, and more nearly optimum staging (with less constraint on the orbiter propellant quantity). The overall effect was sufficient to allow reduction of the orbiter length by 17 feet to provide even greater savings. While Configuration B avoids the development risk of large reusable cryogenic tankage, it involves increased TPS risk because of its higher wingloading.

Configuration C is lighter than Configuration D because of the lower weight orbiter caused mostly by less weight in structure, in the on-orbit maneuvering system (OMS), and in the reaction control system (RCS). The delta body structure has less total surface area (11,210 ft^2) than deltal wing vehicle (12,968 ft^2) and the structure is distributed to cause lower line loads. A 7880-lb estimated savings in structural weight is considered conservatively low. The OMS and RCS in the delta body vehicle take advantage of the greater internal volume available to show a weight savings of about 11,000 lb by using hydrogen/oxygen propellants rather than the less efficient storable propellants used in the delta wing vehicle. An estimated increase in development cost for H_2/O_2 RCS of about $40 million is more than compensated for by the effects of a 4,000-lb RCS weight saving, which yields a program cost reduction (500 flight program) of over $100 million.

The orbiter of Configuration E is the same as that of Configuration D except that 6,000 lb of abort rocket thrust structure is removed, plus the secondary

Table 1-1

WEIGHT SUMMARY

BASELINE CONFIGURATIONS

(10^3 Pounds)

		Dry	Inert* (W 40K P/L)	Liftoff (W 40K P/L)
A	Orbiter	300	345	692
	Droptank	135	145	3,207
	Total			3,899
B	Orbiter	257	302	325
	Ext Tank	38	41	936
	Droptank	98	106	2,296
	Total			3,557
C	Orbiter	160	203	218
	Ext. Tank	67	73	1,671
	SRM Booster	319	319	2,619
	Abort Motors		68	68
	Total			4,576
D	Orbiter	171	215	241
	Ext. Tank	68	73	1,681
	SRM Booster	383	383	3,252
	Abort Motors		76	76
	Total			5,250
E	Orbiter	165	208	234
	Ext. Tank	55	59	1,170
	Booster	437	492	2,599
	Total			4,003

*Orbiter inert weight is landing weight; other element inerts are staged weight effects of this removal. The Configuration E flyback booster uses 12 liquid rocket engines with 8% overthrust capability. With the failure of any one engine, liftoff at essentially the normal thrust-to-weight would still be possible. For this reason, it is assumed that no separate abort rocket system need be incorporated.

BASELINE COSTS

Estimated program costs for the five baseline designs are illustrated in Fig. 1-2. The stage-and-one-half configurations (A and B) show slightly greater development costs and considerably less operations cost than the solid boosted configurations (C and D). The greater development cost is primarily due to the larger-sized orbiter needed to incorporate all ascent rocket engines. It may be the uncertainty in this cost estimate for development of a large orbiter which most detracts from the desirability to NASA of the stage-and-one-half approach. The lower recurring costs result mainly from the lesser cost of operational hardware and propellants.

While advantage is taken in Configurations C and D of reuse of solid rocket systems to a reasonable extent (average of 6 reuses for solid cases and subsystems), they are eventually expended and there is still a clear cost advantage of droptanks and liquid propellants as expendables over solid rocket systems.

Comparing Configurations A and B, the orbiter of Configuration B, being slightly smaller than that of A, costs a little less to develop. The added recurring costs for more expendable tankage compensates for development and production savings to make the total program cost pattern for Configuration B almost identical to that for A.

There is a saving of about $400 million in total program cost (500 flight program) for Configuration C compared to Configuration D. The lower weight delta body orbiter allows a reduction in liftoff weight from 5.25M lb to 4.58M lb and reduces SRM requirements so that the average recurring cost per flight

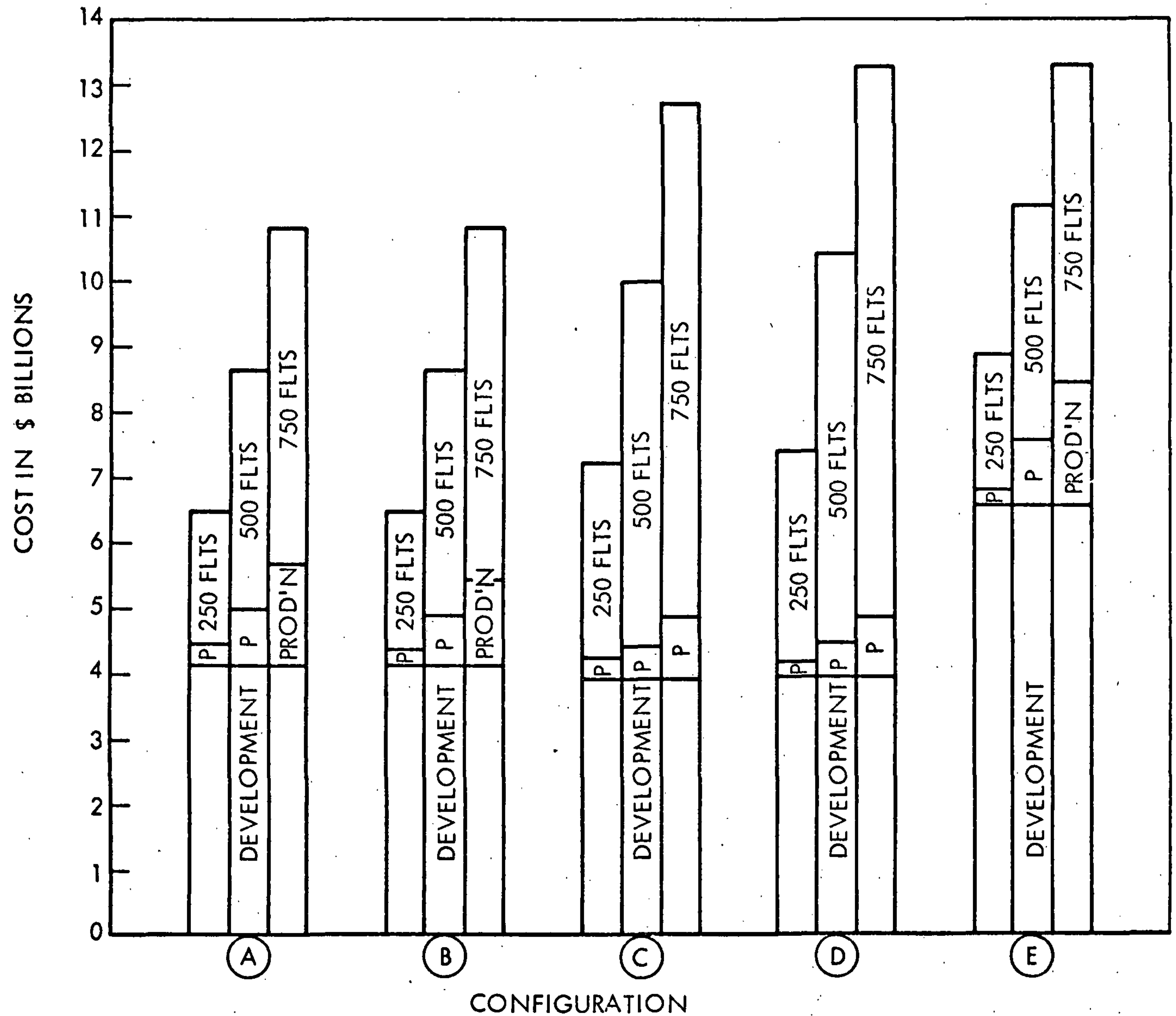

Figure 1-2 Cost Summary

drops about $750,000. These estimates have been made on a conservative basis and more detailed study should show even greater savings. An area of cost comparison between Configurations C and D which is probably not conservative is the orbiter development cost, estimated for C to be only $11 million more than for D. This is the balance of a $40M increase due to the H_2/O_2 RCS system and a savings due to lighter weight structure. Due to lack of data, no complexity factor has been introduced to account for the more-complex shape of the Configuration C delta body orbiter. If conservative factors of 1.1 for development and 1.2 for production were applied to the delta body structure costs, an increase of about $50M would accrue in the Configuration C orbiter non-recurring costs. The 500 flight program results would then show Configuration C non-recurring costs to be $20M more than for D rather than $30M less, and total program savings would be reduced from $400M to $350M. Thus, Configuration C clearly has a lower total program cost than D, but its development cost could be slightly higher.

The development cost of Configuration E, with its fully reusable booster, is sufficiently greater than that of D that it takes a full 750 flight program to reach a breakeven point.

In view of the various aspects of costs, a significant advantage of the currently proposed Configuration D over Configurations A, B, or C may be minimum uncertainty in development cost estimates. It is beyond the scope of this study to completely evaluate the magnitude of this advantage. However, as will be seen, the sensitivity values provided can contribute some insight, since they would be some of the important inputs to a risk analysis.

SENSITIVITY SUMMARY

A summary of cost sensitivity results of this study (considering "free" input weight) is depicted in Figure 1-3. Further detail is provided in extensive tables in Section 5, including the breakdowns of total program cost sensitivities for a 500 flight program into the contributions from development, production, and recurring cost changes and by the system cost changes in each

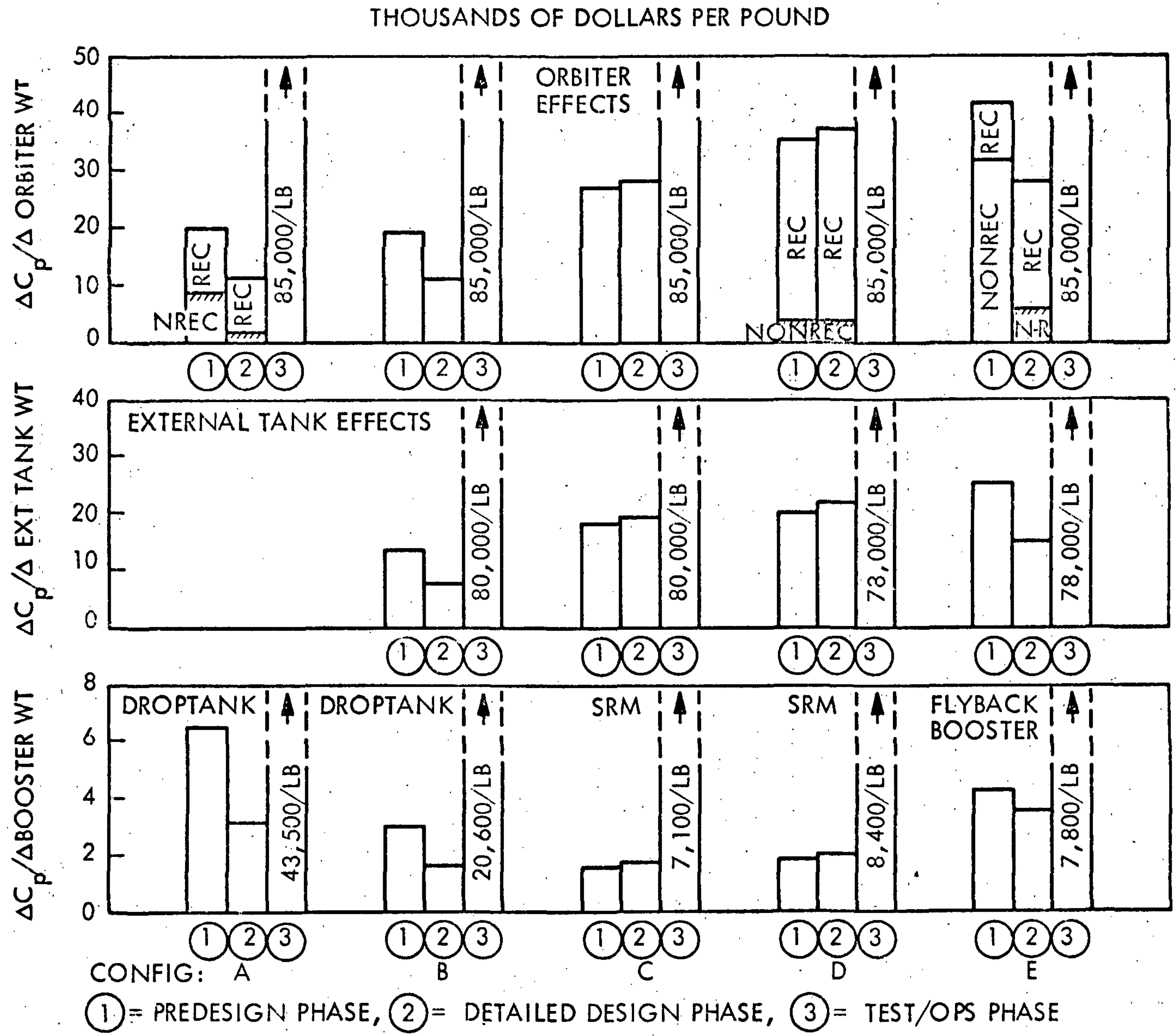

Figure 1-3 Total Program Cost (Cp) Sensitivities 500-Flight Program

major vehicle element. Also, for Configurations A and D, results for a 250 flight program and a 750 flight program are included. For all cases, values for both free and costed input weight are provided. (Free input weight results are applicable to design trade studies and are used throughout this summary.) Performance and direct cost sensitivities are also delineated in considerable detail. By providing this wide range of results and providing, in Section 4, an extensive discussion of the methods used in their derivation, it is hoped to enhance understanding of the subtleties of the variations in these important parameters.

Fixed performance sensitivities are applicable for changes during preliminary design and detailed design phases when the vehicle can be redesigned to retain compliance with performance requirements. As seen in Figure 1-3, these sensitivities show a range of total program cost effects from less than $2,000 per lb to about $42,000 per lb. Much of this spread can be naturally attributed to the rocket stage being considered (less sensitivity for booster input weight than for orbiter), but there can be a factor of three between corresponding cost sensitivities for different configurations.

The relatively high fixed performance sensitivities to orbiter weight of the solid booster vehicles (Configurations C and D) results principally from the increase in the recurring costs for SRMs of increased size. This is such a strong effect that the total program cost sensitivity of Configuration D is considerably greater than that of Configuration A even though its development cost sensitivity during preliminary design is only about 50% of that for A.

An apparent anomaly occurs in the change in sensitivity when going from the preliminary design phase to the detailed design phase. The sensitivities of the stage-and-one-half cases (and the two-stage case) decrease considerably, while that for the solid-booster vehicles increases slightly. This irregularity results from the redesign constraints appropriate for these two types of vehicle during detailed design. The main engine thrust for the stage-and-one-half system would be frozen late in the preliminary design at a liftoff thrust-to-weight ratio of 1.3, which is large enough to allow for a later increase in droptank size. The resizing assumption for the stage-and-one-half thereafter involves constant thrust engines (with decreasing thrust-to-weight ratio down to 1.15). This involves minimal redesign of the orbiter. The result is a considerably lower nonrecurring cost sensitivity. It can be said that the cost penalty had already been taken when the orbiter and its engines were oversized to provide a sufficient thrust-to-weight margin. A thrust-to-weight decrease from 1.3 to 1.15 in Configuration A provides for approximately 20,000 lb of orbiter growth.

For the solid-boosted systems, the orbiter redesign is always "minimal" since its main engines are not critical to liftoff and are assumed fixed even during preliminary design. This is the reason for low development cost sensitivities (in both preliminary and detailed design phases) for these cases.

In detailed design when the external tank size is fixed, there is a slightly higher gross-weight sensitivity and the solids (which, to their benefit, are still assumed to be resized with a constant thrust-to-weight ratio) grow more than in the preliminary design phase.

An important advantage of solids is that they can be resized in both propellant quantity and thrust level fairly easily. This means that the vehicle does not have to be oversized initially to allow for possible growth that may not occur. This advantage shows up here as an apparent disadvantage of a continued high cost sensitivity in the detailed design phase. This effect actually results from the fact that more design flexibility can be retained with a solid booster than in the case of stage-and-one-half (thrust, as well as propellant capability, can be changed).

A point to be noted about the nature of cost sensitivities is that they must be examined in context with the baseline design and baseline costs. Any vehicle can be desensitized by oversizing the baseline so that little or no redesign is ever required. Each system approach has its own peculiar characteristics, and direct sensitivity comparisons must be interpreted with care.

PERFORMANCE/COST SENSITIVITY CORRELATION

Figure 1-4 provides a comparison of performance sensitivity (that of liftoff weight with respect to orbiter weight) with total program cost sensitivity to orbiter weight. It can be seen that good correlation exists between these two sensitivities for Configurations A, B, C, and D during the preliminary design phase. However, in the detailed design phase, or when Configuration E is considered, very little correlation exists.

It appears that for Configurations with expandable booster systems, when complete design freedom is available a program cost change can be reasonably well predicted from the value of a liftoff weight change. The program cost of a system with a reusable booster is, however, much more sensitive to liftoff weight. Also when design constraints appropriate to a given system are imposed(that is, in the detailed design phase), prediction of program cost changes requires more careful analysis of each configuration separately.

CONCLUSIONS

The results presented point to these principal conclusions:

1. Space Shuttle cost sensitivities are quite high for all configurations. The following table summarizes the most important values, which are the sensitivities to orbiter weight for Configuration D, the currently planned approach.

CONFIGURATION D SENSITIVITIES TO ORBITER WEIGHT

Program Phase	Development Cost	Total Program Cost (500 flights)
Prelim. Design	$4000/lb	$34,900/lb
Detailed Design	$4400/lb	$36,900/lb
Test/Operations	-	$85,000/lb

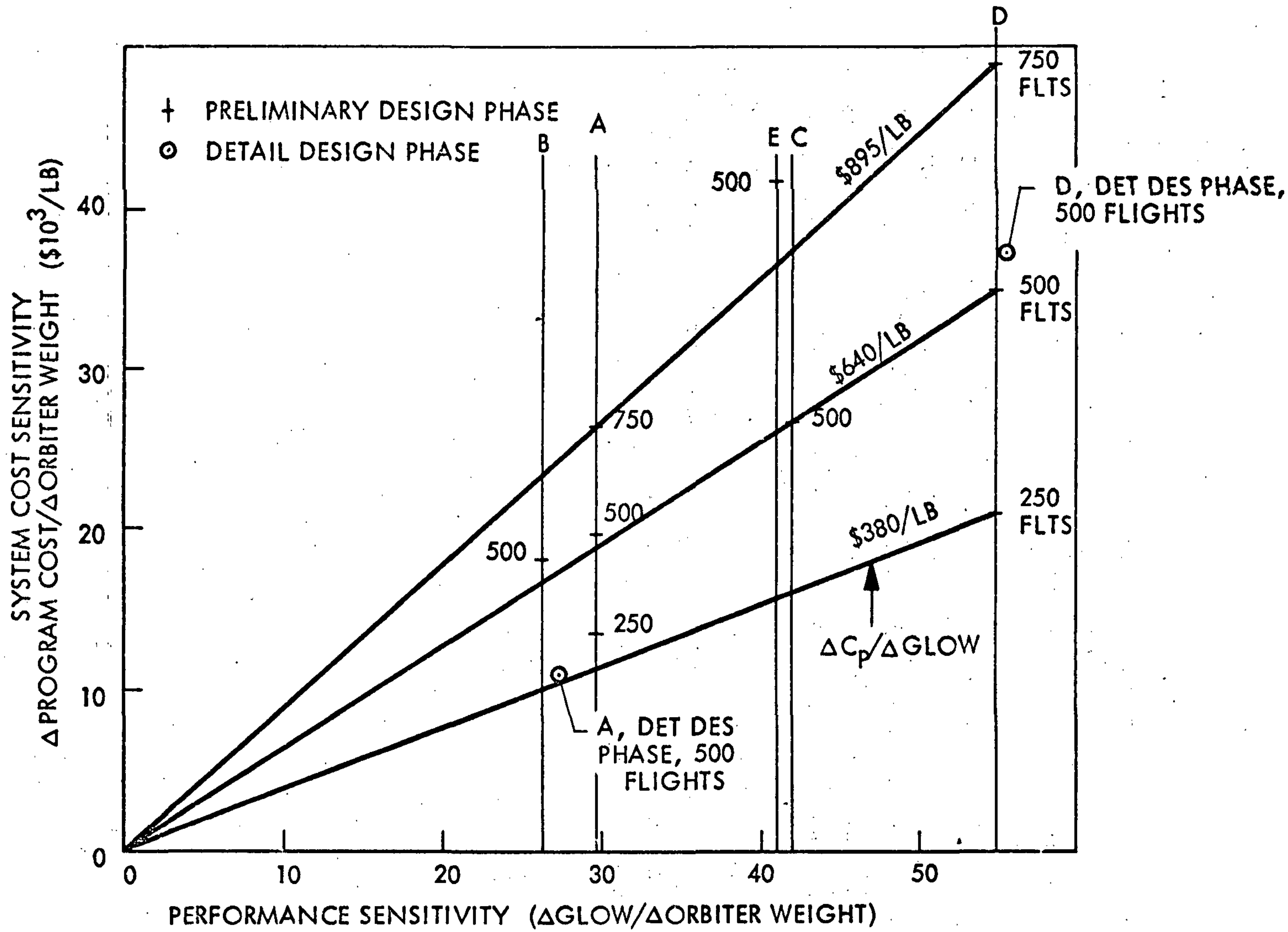

Figure 1-4 Correlation Between Performance and Cost Sensitivities

2. Sensitivity values vary widely with the various parameters of this study. Figure 1-5 depicts the trends for the four principle parameters. It also indicates the significance of each parameter by the ratio between the maximum and minimum cost sensitivity over the range of the parameter (for the worst case with other parameters held fixed).

Review of the methods used to obtain the results shows that to utilize these methods requires extensive computerized system design and costing capabilities. Since care has been taken to avoid overlooking any significant cost effects, the sensitivity results are believed reliable within perhaps ±25%. This is sufficient accuracy for most applications of system sensitivities.

It is likely that simpler methods for deriving sensitivities with comparable accuracy can be developed. The extensive results of this study could be used as a data bank of accurate results (based on the assumptions made) for testing the validity of simpler analysis techniques.

	INCREASING COST SENSITIVITIES →			MAX/MIN*
BOOSTER CONFIGURATION				
TOTAL PROGRAM COSTS	LIQUID WITH EXPENDABLE TANK (CONFIGURATION A AND B)	SOLID WITH SOME REUSE (CONFIGURATION C AND D)	FLYBACK REUSABLE (CONFIGURATION E)	2.2
DEVELOPMENT COSTS	SOLID	LIQUID (EXP. TANKS)	FLYBACK REUSABLE	9.6
ORBITER CONFIGURATION	DELTA-BODY	DELTA-WING		1.33
PROGRAM PHASE				
LIQUID BOOSTERS	DETAILED DESIGN	PRELIMINARY DESIGN	TEST/OPS	7.5
SOLID BOOSTERS	PRELIMINARY DESIGN	DETAILED DESIGN	TEST/OPS	3.2
PROGRAM TRAFFIC	250 FLIGHTS	500 FLIGHTS	750 FLIGHTS	2.5

*ALL RATIOS ARE FOR TOTAL PROGRAM COST SENSITIVITIES EXCEPT THE 9.6 DEVELOPMENT COST RATIO FOR BOOSTER CONFIGURATION.

Figure 1-5 Sensitivity Trends

Section 2

GROUNDRULES AND ASSUMPTIONS

2.1 PERFORMANCE REQUIREMENTS

The performance requirements to be met by all configurations are those defined in "Space Shuttle Program Request for Proposal No. 9-BC421-67-2-40P", issued by NASA on 17 March 1972. This document contains approximately 40 pages of technical requirements, ranging from the General System Requirements, in which capabilities required for three missions are defined, through quite detailed performance requirements for each of the vehicle subsystems. By drawing on the results of the LMSC proposal effort, which defined Configuration D, each of five configurations are designed to meet all of these performance requirements.

The Space Shuttle RFP specifies a solid rocket booster system such as Configuration C or D. These two configurations are designed to meet all RFP requirements and have nearly identical capabilities. As shown in Figure 2-1, mission performance capability of the other configurations varies somewhat even though all configurations meet the same mission requirements. Other specifics of performance, such as abort capability, will also vary with configuration but designs are as comparable as possible.

2.2 DESIGN GROUNDRULES

Groundrules for the design of each configuration are summarized as follows:

Configuration A, Stage-and-One-Half

- Body shape and length the same as Model LS-200-11, as defined in LMSC/A995931, Vol. II, Part 2, of Alternate Concepts Study Extension, Final Report, dated 15 November 1971.

		REQUIREMENTS			
MISSION NUMBER		1	2		3
ORBIT INCLINATION		28.5 DEG	55 DEG		90 DEG
V ON ORBIT (OMS + RCS)		1070 FT/SEC	1520 FT/SEC		650 FT/SEC
			W/O ABES	W ABES	
PAYLOAD REQUIREMENT*		65K	25K	NONE	40K
		CAPABILITIES			
A (1-1/2 STAGE)	△BODY ORBITER	67.2K	38.9K	7.4K	$\boxed{40K}$
B (1-1/2 STAGE EXT. TANK)	△BODY ORBITER	67.6K	39.7K	9.2K	$\boxed{40K}$
C (△BODY RAD)	△BODY ORBITER	66.7K	41.5K	20.9K	$\boxed{40K}$
D (△WING RAD)	△WING ORBITER	67.0K	39.7K	19.0K	$\boxed{40K}$
E (2 STAG EXT. TANK)	△WING ORBITER	65.8K	41.1K	20.5K	$\boxed{40K}$

*MAXIMUM RETURN PAYLOAD REQUIREMENT = 40K

VEHICLE SIZING CASE

Figure 2-1 Payload Requirements and Capabilities

- Fin size to be increased as needed to meet 150 knots landing speed requirement
- Structure weight to be modified to be consistent with titanium/aluminum honeycomb used on Configuration D
- Weights of all subsystems to be updated to be consistent with capabilities of Configuration D
- H_2/O_2 on-orbit maneuvering propulsion system (OMPS) and reaction control system (RCS)
- OMPS and RCS tankage to be common and to incorporate cross-feed capability with main engine system for maximum mission flexibility (similar to Model LS-200-11)
- Addition of two airbreathing engines (to Model LS-200-11), for a total of six, to meet loiter capability at 10,000 ft altitude.

<u>Configuration B, Stage-and-One-Half with External Orbiter Tank</u>

- Delta-body orbiter shortened from LS-200-11 by approximately 17 feet and internal ascent tanks removed
- Three tanks of same diameter, with twin droptanks containing added cylindrical sections as needed to establish a reasonable design considering:
    (1) Near optimum staging
    (2) Convenient attachment
    (3) Center of gravity travel producing less than 10 degrees of engine gimbal
- Nine main engines and a liftoff thrust-to-weight ratio of 1.25
- H_2/O_2 OMPS and RCS

Configuration C, Delta-Body Rocket-Assisted Orbiter

- Similar to Configuration D (LMSC proposal) but with delta body orbiter
- Orbiter of minimum length to contain payload bay and subsystems (Ref. length = 120 feet)
- Three main engines of same size as Configuration D - approximately 470K vacuum thrust in accordance with interface control document requirements
- H_2/O_2 OMPS and RCS
- Staging velocity for minimum GLOW

Configuration D, Delta-Wing Rocket-Assisted Orbiter

- Design as defined in detail in LMSC proposal in response to Space Shuttle RFP (has staging velocity for minimum GLOW)

Configuration E, Two-Stage with Flyback, Heatsink Booster and External Tank Delta-Wing Orbiter

- Orbiter same as Configuration D, except as stated below
- Orbiter engines started after staging (series burn)
- Removal of abort rocket system (including abort thrust structure removal from orbiter)
- Orbiter engines of same size as used in booster, except larger expansion ratio (90:1 rather than 35:1)
- Smaller, series-burn type external tank
- Booster to have 12 H_2/O_2 main engines of sufficient thrust to provide 1.25 liftoff thrust-to-weight ratio
- Staging velocity for minimum gross liftoff weight but less than 6,000 ft/sec to allow heatsink type thermal control on reentry

2.3 RESIZING GROUNDRULES

Resizing groundrules for determining system sensitivities to weight changes are of course different for the preliminary design and detailed design phases. In addition, specific ground rules apply uniquely to each configuration. In general, for the preliminary design phase all system elements can be redesigned and resized as needed to determine a new "optimized" configuration. In the detailed design phase, it is assumed that only one vehicle element is resized to meet ascent performance.

For both design phases, orbiter redesign is also needed in cases of orbiter weight changes to (1) retain crossrange capability (affects the thermal protection system, TPS), (2) retain landing speed (affects wing or fin size), and (3) meet flying and landing load requirements (affects structure and landing gear). For Configuration E, these same factors apply to the flyback booster when booster weights change, and to resizing of the airbreathing engine system (ABES) as needed. These considerations are summarized in Fig. 2-2 as they apply to each of the five configurations.

2.4 COATING GROUNDRULES

The principal groundrules for costing are listed in Fig. 2-3. More detail on the traffic model, as well as data on the cost estimating relationships (CERs), is included is Section 4.3.

	BASELINE CRITERIA	RESIZING CONSTRAINTS	PROGRAM PHASE PRELIM DES OR DET DES	INPUT WEIGHT	SYSTEMS RESIZED: ORBITER: MAIN ENGINE THURST	WING OR FIN SIZE	STRUCT BEEF-UP	TPS THICKNESS	LDG GEAR	EXT TANK CAPACITY	BOOSTER OR D/T CAPACITY
A	VOLUME LIMITED W_P IN ORBITER	RETAIN ORBITER BODY LENGTH	P-D	ANY	●	●	●	●	●		●
			D-D	ORB		●	●	●	●		●
				D/T							●
B	ALL TANKS SAME DIA.	TANK DIA GROWS IN P-D	P-D	ANY	●	●	●	●	●	●	●
		DROP TANK FIXED IN D-D	D-D	ORB		●	●	●	●	●	
				EXT TANK						●	
				DROP TK						●	
C	MINIMUM GLOW	NEW MIN GLOW IN P-D	P-D	ORB		●	●	●	●	●	●
				ET OR SRM						●	●
		EXT TANK FIXED IN D-D	D-D	ORB		●	●	●	●		●
				ET OR SRM							●
D	MINIMUM GLOW	NEW MIN GLOW IN P-D	P-D	ORB		●	●	●	●	●	●
				ET OR SRM						●	●
		EXT TANK FIXED IN D-D	D-D	ORB		●	●	●	●		●
				ET OR SRM							●
E	MINIMUM GLOW	NEW MIN GLOW IN P-D	P-D	ANY	●	●	●	●	●	●	●
		BOOSTER SIZE FIXED IN D-D	D-D	ORB		●	●	●	●	●	
				EXT TK						●	
				BOOSTER						●	●*

*STRUCT, LDG GEAR, ABES CHANGED BUT NO CHANGE IN ROCKET PROPELLANT CAPACITY IN BOOSTER

Figure 2-2 Resizing Groundrules

- COSTS IN 1971 DOLLARS
- NO PRIME CONTRACTOR FEE INCLUDED
- NO GOVERNMENT INSTITUTIONAL COSTS INCLUDED
- OPERATIONAL FLEET CONSISTS OF:
  - 3 ORBITERS FOR 250 FLTS (2 BOOSTERS, CONF. E)
  - 5 ORBITERS FOR 500 FLTS (4 BOOSTERS, CONF. E)
  - 8 ORBITERS FOR 750 FLTS (6 BOOSTERS, CONF. E)
- FIRST TWO ORBITERS ARE PROVIDED FROM DEVELOPMENT PROGRAM (2 BOOSTERS, CONF. E)
- OPERATIONAL PROGRAM CONSISTS OF 500 FLIGHTS* IN TEN YEARS IN ADDITION TO SIX FLIGHTS CHARGED TO DEVELOPMENT
- ONE OPERATIONAL LAUNCH FACILITY (KSC)

*SELECTED RESULTS TO BE COMPUTED FOR 250 AND 750 FLIGHTS IN TEN YEARS

Figure 2-3 Costing Groundrules and Assumptions

Section 3

CONFIGURATION BASELINES

Before system sensitivities can be computed, fairly complete baseline designs must be defined. The design choices made for the baseline can have significant effects on sensitivities. Conservative designs are generally less sensitive since less redesign is required to retain compliance with requirements. The five baseline configurations, whose characteristics are summarized in this section, have been designed to a common set of requirements and, to the extent possible, with the same degree of conservatism. They are each distinctive approaches, however, and each must be considered in light of its own peculiar characteristics.

3.1 CONFIGURATION A: STAGE-AND-ONE-HALF

The stage-and-one-half concept (Orbiter Configuration depicted in Figure 3.1-1)* employs a fully reusable orbiter vehicle in combination with a single set of expendable droptanks the stage-and-one-half system shown is basically a derivative of an existing Lockheed design (Model LS-200-11)**. Primary differences are subsystem weight increases (comparable to configuration D), increased fin area to provide capability for a landing speed of 150 knots, (the effects of which are included in the weight data) and the use of LO_2/LH_2 for the RCS subsystem.

All rocket and airbreathing engines and attitude control thrusters are assembled in the orbiter. The droptank assembly contains only the elements necessary for storing the propellants during ascent, and the plumbing and

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* Configuration drawings are at the end of the section.

** See LMSC-A995931, Final Report, Alternate Concepts Study Extention, Vol II, Part 2, One-And-One-Half Stage System, dated 15 November 1971.

instrumentation required for propellant transfer and pressurization, and pressure, temperature, and fluid level control. Thus, by reducing the function of the droptanks to that of a tank proper, and by selecting a configuration permitting a highly weight-effective design, a very high propellant fraction is obtained for this element of the vehicle system. Consequently, optimum staging is achieved at a high velocity, exceeding 18,000 ft/sec.

Using eleven 460,000 lb sea-level thrust main rocket engines and with all engines operating at liftoff, a payload capability of 40,000 lb is achieved with a nominal gross liftoff weight of 3.9×10^6 lb, corresponding to a nominal liftoff thrust-to-weight ratio of 1.30. Weight and cost summaries are shown in Tables 3.1-1 and 3.1-2.

A typical development schedule showing key milestones related to orbiter and tank development is shown in Figure 3.1-2. The relationship of performance and vehicle sensitivities to the program schedule is illustrated.

Orbital maneuvering capability is provided by an orbital maneuvering propulsion system (OMPS) consisting of two RL10 engines in combination with LO_2/LH_2 propellant tanks and a feed system designed for the long storage time requirement. The main engines may also use some of these propellants when not needed on orbit. Only one RL10 engine is used for normal operation, the second being a standby providing engine-out capability. Full attitude control capability is obtained during the ascent phase by gimbaling 5 of the 11 main engines, and pitch and yaw control during orbit maneuvers by gimbaling the operating RL10 engine.

A reaction control propulsion system (RCS) using LO_2/LH_2 propellant provides roll control during RL10 engine operation and full attitude control capability on orbit and during the initial reentry phase. It is also used for supplying the three-axis translatory impulses for docking and similar maneuvers. The LO_2/LH_2 propellants were chosen because of low contamination,

TABLE 3.1-1

DELTA-BODY STAGE-AND-ONE-HALF
MISSION WEIGHT SUMMARY
CONFIGURATION A

System	Orbiter	Droptank	GLOW
Wing Group	N/A		
Tail Group	21,644		
Body Group	62,602		
Induced Envir. Protection	36,183		
Landing, Recovery, Docking	16,017		
Propulsion — Ascent	108,230		
Propulsion — Auxiliary	7,942		
Prime Power	4,123		
Elect. Conver. and Distr.	2,915		
Hydra Conver. and Distr.	2,091		
Surface Controls	4,293		
Avionics	7,344		
Environmental Control	4,456		
Personnel Provisions	1,269		
Growth/Uncertainty	20,780		
Subtotal (Dry Weight)	299,889	134,726	
Personnel	1,621		
Cargo	40,000		
Residual Fluids	3,453	10,512	
Subtotal (Inert Weight)	344,963	145,238	
Reserve Fluids	1,953	8,080	
In-Flight Losses	5,850	7,170	
Propellant — Ascent	319,919	3,046,469	
Propellant — Maneuv/ACS	19,351		
Total Orbiter	692,036		
Total Droptank		3,206,957	
GLOW			3,898,993

TABLE 3.1-2

CONFIGURATION A COSTS

(Millions of Dollars)

Flights in Program	System Element	Nonrec		Rec	Total	Av Rec Cost Per Flt
		Dev	Prod			
250	Orbiter	3,181	292	944	4,417	
	Booster (or Droptank)	252	–	916	1,168	
	System	698	30	191	919	
	Total	4,131	322	2,051	6,504	8.20
	Tot. Nonrec/Peak	4,453/1,073				
500	Orbiter	3,181	761	1,647	5,589	
	Booster (or Droptank)	252	–	1,641	1,893	
	System	698	78	326	1,102	
	Total	4,131	839	3,614	8,584	7.23
	Tot. Nonrec/Peak	4,970/1,073				
750	Orbiter	3,181	1,409	2,335	6,925	
	Booster (or Droptank)	252	–	2,305	2,557	
	System	698	144	475	1,317	
	Total	4,131	1,553	5,115	10,799	6.82
	Tot. Nonrec/Peak	5,684/1,073				

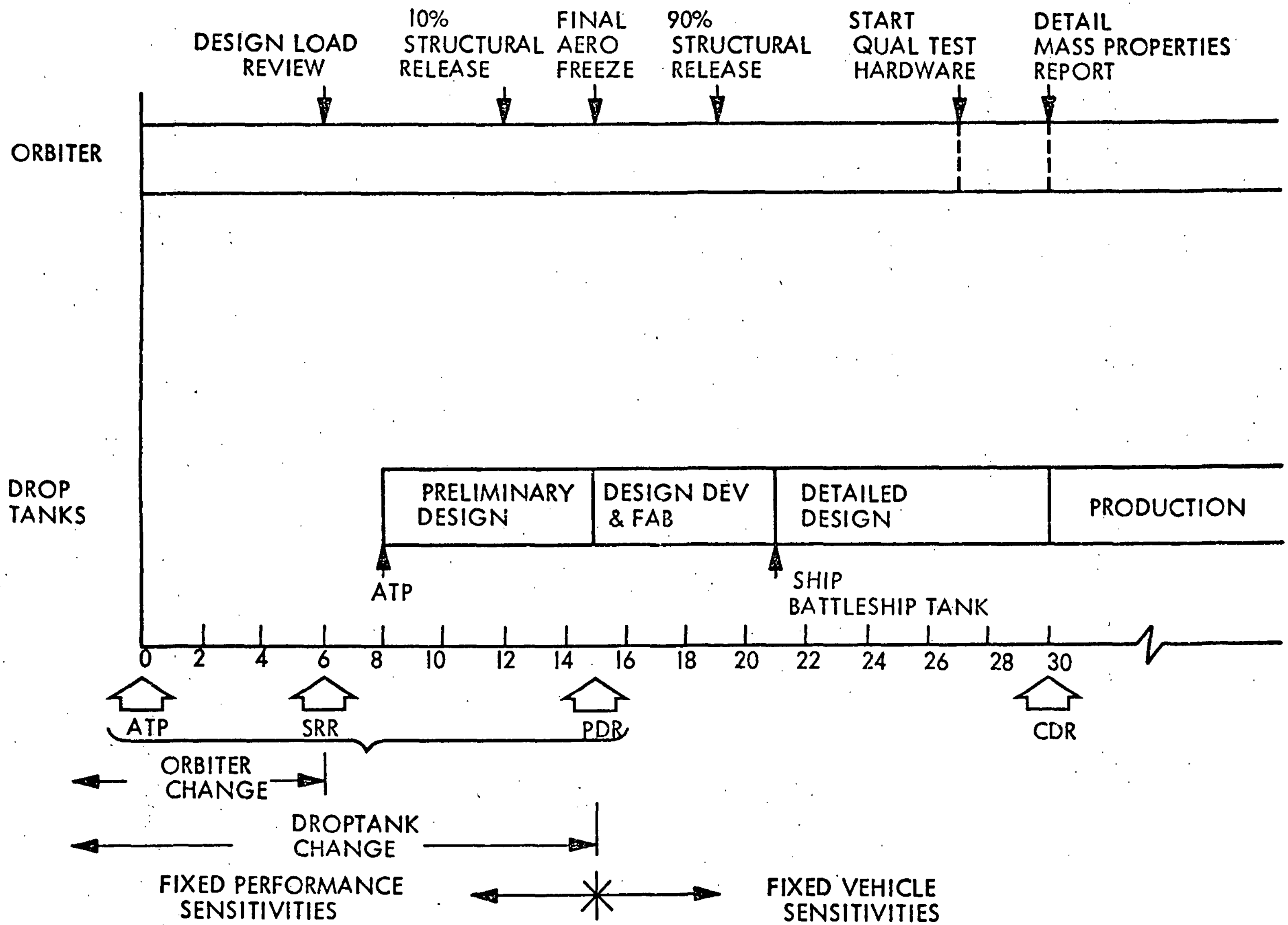

Figure 3.1-2 Typical Development Schedule - Configuration A

acceptable hazard and cost effectiveness relative to other alternatives (for delta-body vehicles when sufficient volume is available).

3.2 CONFIGURATION B: STAGE-AND-ONE-HALF WITH EXTERNAL TANK

Configuration B (Figure 3.2-1) employs a delta-body orbiter similar to that used in Configuration A with the reference length reduced from 150 ft to 133 ft. The Configuration B composite launch vehicle is composed of this orbiter in conjunction with vee-type droptanks and an external orbiter tank. The droptanks are staged during ascent prior to injection; the external HO tank is carried into orbit after propellant depletion. For this configuration, the LO_2/LH_2 propellant required for the OMPS and RCS subsystems is carried internally within the orbiter. Weight and cost summaries are shown in Tables 3.2-1 and 3.2-2.

The nine orbiter main engines have 495,000 lb sea-level thrust each to provide a liftoff thrust to weight ratio of 1.25. This value of thrust-to-weight was used, rather than the 1.3 chosen for configuration A, in the intesest of providing a small competitive configuration B orbiter. This choice results in a slightly greater risk since the lesser sensitivity of GLOW to orbiter weight only partially compensates for the smaller thrust-to-weight margin. The configuration A orbit could grow over a 6% of its dry weight before the thrust-to-weight ratio drops to 1.15. The corresponding value for configuration B is less than 5% of orbiter dry weight growth potential.

During ascent, propellants are initially fed from the twin droptanks. Slightly prior to their depletion (perhaps 3 sec), 2 engines will have been shut down to stay under a 3g load factor and 5 of the remaining engines are then throttled to 50% thrust and switched to the external orbiter tank. The other 2 engines will continue to be fed from the droptanks until depletion (soft shutdown) so that unused propellants are minimal. After droptank staging, the 5 engines are brought back to full thrust, to be appropriately throttled and/or shut down later as needed to avoid exceeding the 3g limit. Since the engines to be switched are

TABLE 3.2-1
STAGE AND ONE HALF WITH EXT. TANK
MISSION WEIGHT SUMMARY
CONFIGURATION B

System	Orbiter	External Tank	Drop-tank	GLOW
Wing Group	N/A			
Tail Group	18,393			
Body Group	56,264			
Induced Envir. Protection	33,065			
Landing, Recovery, Docking	14,412			
Propulsion — Ascent	85,306			
Propulsion — Auxiliary	7,376			
Prime Power	4,123			
Elect. Conver. and Distr.	2,616			
Hydra Conver. and Distr.	1,943			
Surface Controls	3,165			
Avionics	7,344			
Environmental Control	4,456			
Personnel Provisions	1,269			
Growth/Uncertainty	17,633			
Subtotal (Dry Weight)	257,365	37,670	98,014	
Personnel	1,621			
Cargo	40,000			
Residual Fluids	2,797	3,049	7,539	
Subtotal (Inert Weight)	301,783	40,719	105,553	
Reserve Fluids	1,757	8,106	0	
In-Flight Losses	4,844	2,821	11,645	
Propellant — Ascent		885,694	2,178,365	
Propellant — Maneuv/ACS	17,003			
Total Orbiter	325,387			
Total Orbit Tank		936,340		
Total Droptank			2,295,563	
GLOW				3,557,290

TABLE 3.2-2

CONFIGURATION B COSTS

(Millions of Dollars)

Flights in Program	System Element	Nonrecur		Rec	Total	Av Rec Cost Per Flt
		Dev	Prod			
250	Orbiter	3,116	270	923	4,309	
	External Tank Booster (or Droptank)	250	–	1,022	1,272	
	System	696	28	199	923	
	Total	4,062	298	2,144	6,504	8.58
		4,360				
500	Orbiter	3,116	700	1,609	5,425	
	External Tank Booster (or Droptank)	250	–	1,830	2,080	
	System	696	72	352	1,120	
	Total	4,062	772	3,791	8,625	7.58
		4,834				
750	Orbiter	3,116	1,292	2,280	6,688	
	External Tank Booster (or Droptank)	250	–	2,571	2,821	
	System	696	132	497	1,325	
	Total	4,062	1,424	5,348	10,834	7.13
		5,486				

the bottom row engines, less than 10 deg. of gimballing is need to track the c.g. This leaves a reasonable margin for control within the ± 7 deg gimbal limits.

The relationship between the development schedule and resizing constraints for computing sensitivities is shown in Figure 3.2-2.

3.3 CONFIGURATION C: DELTA-BODY ROCKET-ASSISTED ORBITER (RAO)

The configuration C launch vehicle (Figure 3.3-1) consists of a delta-body orbiter, an external propellant tank, two recoverable 156-in. solid-rocket motors, and two abort solid-rocket motors. The orbiter engines and main solid-rocket motors burn in parallel at liftoff; the abort motors are not burned during normal operations. Liftoff thrust-to-weight ratio is 1.45.

Configuration C utilizes a delta-body orbiter aerodynamic configuration similar to Configurations A and B reduced to 120 ft referenced length. The orbiter main propulsion consists of three engines of 470,000 lb vacuum thrust. The OMPS and RCS subsystems for the orbiter use O_2/H_2 propellants. Weight and cost summaries are shown in Tables 3.3-1 and 3.3-2. A typical development schedule showing key milestones relating to sizing assumptions for sensitivity analysis is shown in Figure 3.3-2.

3.4 CONFIGURATION D: DELTA-WING ROCKET-ASSISTED ORBITER

Configuration D vehicle (Figure 3.4-1) is of the type currently planned for development. It is the same approach as Configuration C except that a delta-wing orbiter is used instead of a delta-body orbiter. Parallel burn of orbiter main engines and main solid-rocket motors is used for ascent. System weight and cost summaries are shown in Tables 3.4-1 and 3.4-2. The weight and cost increases over Configuration C are due primarily to a less efficient structural shape and the use of storable propellants for the OMPS

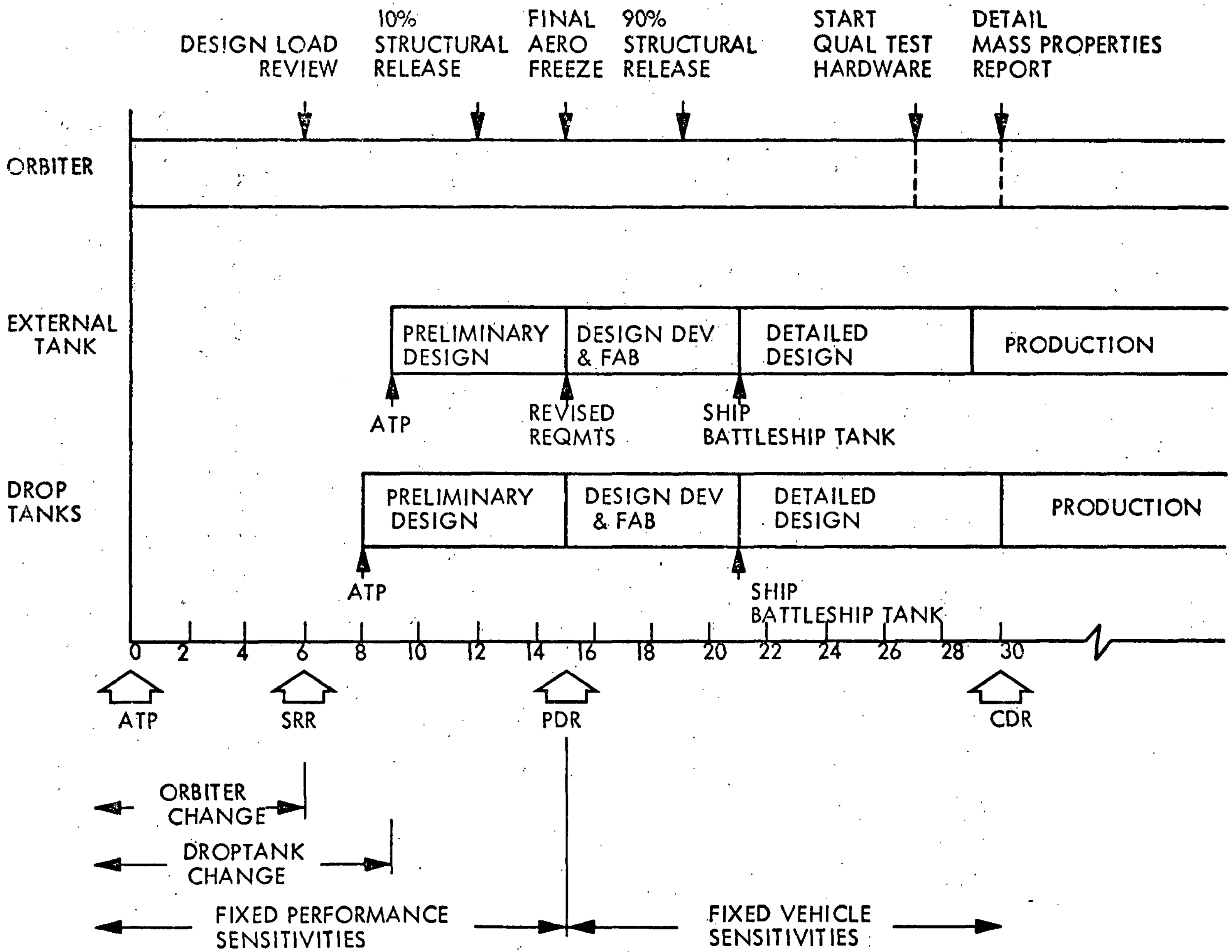

Figure 3.2-2 Typical Development Schedule - Configuration B

TABLE 3.3-1
DELTA BODY ROCKET ASSISTED ORBITER
MISSION WEIGHT SUMMARY
CONFIGURATION C BASELINE

System	Orbiter	External Tank	SRM	ARM	GLOW
Wing Group	N/A				
Tail Group	12,353				
Body Group	46,091				
Induced Envir. Protection	23,200				
Landing, Recovery, Docking	10,746				
Propulsion – Ascent	22,881				
Propulsion – Auxiliary	6,924				
Prime Power	4,123				
Elect. Conver. and Distr.	2,810				
Hydra, Conver. and Distr.	1,773				
Surface Controls	2,620				
Avionics	7,344				
Envir. Control	4,456				
Personnel Provisions	1,269				
Growth/ Uncertainty	12,946				
Subtotal (Dry Weight)	159,536	67,354			
Personnel	1,621				
Cargo	40,000				
Residual Fluids	1,962	5,698			
Subtotal (Inert Weight)	203,119	73,052	318,721		(594,892)
Reserve Fluids	1,624	6,485			
In-Flight Losses	4,705	9,011			
Propellant – Ascent		1,582,845	2,300,000		
Propellant – Maneu/ACS	11,364				
Orbiter Total	220,812				(220,812)
Ext. Tank Total		1,671,393			(1,671,393)
SRM Total			2,618,721		(2,618,721)
ARM Total				68,500	(68,500)
GLOW					(4,579,426)

TABLE 3.3-2

CONFIGURATION C COSTS

(Millions of Dollars)

Flights in Program	System Element	Nonrecur		Rec	Total	Av Rec Cost Per Flt
		Dev	Prod			
250	Orbiter	2,789	184	822	3,795	
	External Tank	211	—	519	730	
	Booster (or Droptank)	188	—	1,454	1,647	
	System	771	19	286	1,076	
	Total	3,959	203	3,081	7,243	12.32
		4,162				
500	Orbiter	2,789	454	1,415	4,658	
	External Tank	211	—	925	1,136	
	Booster (or Droptank)	188	—	2,664	2,852	
	System	771	47	513	1,331	
	Total	3,959	501	5,517	9,977	11.03
		4,460				
750	Orbiter	2,789	823	1,995	5,607	
	External Tank	211	—	1,298	1,509	
	Booster (or Droptank)	188	—	3,819	4,007	
	System	771	84	729	1,584	
	Total	3,959	907	7,841	12,707	10.45
		4,866				

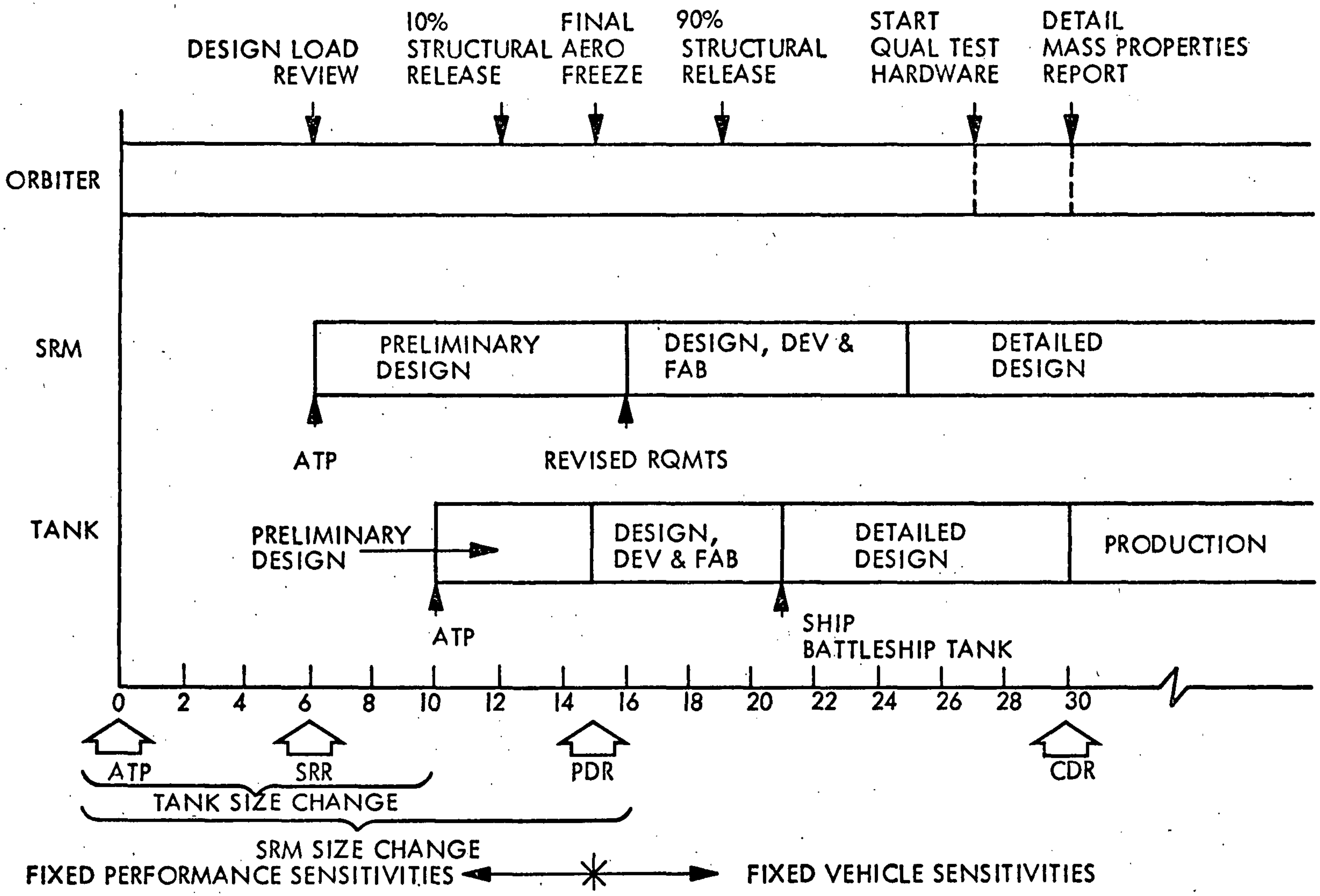

Figure 3.3-2 Typical Development Schedule, Configuration C and D

TABLE 3.4-1
DELTA WING ROCKET ASSISTED ORBITER
MISSION WEIGHT SUMMARY
CONFIGURATION D

System	Orbiter	External Tank	SRM	ARM	GLOW
Wing Group	16,974				
Tail Group	4,345				
Body Group	45,005	49,878	309,467	13,796	
Induced Envir. Protection	20,946	6,760	25,620	1,402	
Landing, Recovery, Docking	11,733		19,518		
Propulsion – Ascent	22,880	6,690	14,730	3,884	
Propulsion – Auxiliary	9,783	2,665			
Prime Power	4,123				
Elect. Conver. and Distr.	2,914	300			
Hydra Conver. and Distr.	1,417				
Surface Controls	3,995				
Avionics	7,344	120	942	170	
Envir. Control	4,456				
Personnel Provisions	1,269				
Growth/ Uncertainty	14,005	1,327	12,780		
Subtotal (Dry Weight)	171,189	67,740	383,057	19,252	
Personnel	1,621				
Cargo	40,000				
Residual Fluids	2,051	5,698			
Subtotal (Inert Weight)	214,861	73,438	383,057	19,252	
Reserve Fluids	2,445	6,889			
In-Flight Losses	4,705	4,868			
Propellant – Ascent		1,595,781	2,869,338	56,248	
Propellant – Maneuv/ACS	19,429				
Total	241,440	1,680,976	3,252,395	75,500	5,250,311

TABLE 3.4-2

CONFIGURATION D COSTS

(Millions of Dollars)

Flights in Program	System Element	Nonrecur		Rec	Total	Av Rec Cost Per Flt
		Dev	Prod			
250	Orbiter	2,778	188	820	3,786	
	External Tank	216	–	520	736	
	Booster (or Droptank)	202	–	1,639	1,841	
	System	781	19	305	1,105	
	Total	3,977	207	3,284	7,468	13.13
	Tot Nonrec/Peak	4,184/1,033				
500	Orbiter	2,778	465	1,412	4,655	
	External Tank	216	–	928	1,144	
	Booster (or Droptank)	202	–	3,004	3,206	
	System	781	48	547	1,376	
	Total	3,977	513	5,891	10,381	11.78
	Tot Nonrec/Peak	4,490/1,033				
750	Orbiter	2,778	842	1,990	5,610	
	External Tank	216	–	1,303	1,519	
	Booster (or Droptank)	202	–	4,311	4,513	
	System	781	86	779	1,646	
	Total	3,977	928	8,383	13,288	11.18
	Tot Nonrec/Peak	4,905/1,033				

and RCS systems. The volume requirements of LO_2/LH_2 systems would necessitate an increase in fusalage dimensions.

The orbiter vehicle is sized to carry a four-man flight crew and provides facilities for a 6-passenger complement. Airlock/docking facilities are provided between the crew cabin and payload bay. A pair of structural doors protect the 15 ft by ft 60 payload bay. Orbiter-mounted abort rockets are located on each side of the orbiter at the aft area of the wing-to-fuselage intersection. These solid-propellant rockets are sized to permit off-the-pad abort and to permit escape from the vehicle tank and SRM.

An airbreathing propulsion system of 4 engines is provided as mission equipment. The ABPS is installed in the payload bay for mission operations during approach and landing.

An all-aluminum external tank is used to carry the main impulse propellants. It consists of two tanks (hydrogen and oxygen) connected by an intertank section. It is attached to the orbiter at three separable points in a tripod arrangement.

Orbiter ascent boost is provided by two parallel-burn 3.52 MLB thrust each, solid propellant, 156 in diameter rocket motors.

A set of lateral-firing separation solid rockets are installed at the forward and aft ends of each SRM to provide for direct translation of the boost rockets away from the orbiter at staging. A parachute recovery system is installed in the nosecone to decelerate and position the spent rockets for aft-end water impact and recovery.

The development schedule is the same as for configuration C and is shown in Figure 3.3-2.

3.5 CONFIGURATION E: TWO STAGE TANDEM (FLYBACK BOOSTER AND DELTA-WING ORBITER)

Configuration E is a two-stage tandem launch system which employs a fully reusable O_2/H_2 booster in conjunction with a delta-wing orbiter/external tank second stage. The orbiter vehicle is essentially that of Configuration D, but uses a smaller external tank, since it burns in series with the booster. The booster vehicle is of the heat-sink type, fully reusable, with a ratio of fuel weight to total weight, λ, at liftoff of about 0.805. The booster contains sufficient JP-4 fuel for powered flyback after staging. Orbiter main propulsion consists of three 480,000 lbf vacuum thrust engines. Since the launch vehicle has tandem staging, the abort rocket motors are removed, giving an orbiter dry weight approximately 6000 lb lower than the configuration D orbiter by removal of thrust structure. Liftoff thrust-to-weight ratio for configuration E is 1.25, thrust being supplied by 12 booster main O_2/H_2 engines of 418,000 lbf sea level thrust each (same power lead as orbiter engines). The launch vehicle configuration is shown in Figure 3.5-1, weight and cost summaries are shown in Tables 3.5-1 and 3.5-2. A typical development schedule is shown in Figure 3.5-2.

TABLE 3.5-1
TWO STAGE WITH EXTERNAL TANK
MISSION WEIGHT SUMMARY
CONFIGURATION E

System	Orbiter	External Tank	Booster	GLOW
Wing Group	16,974		65,815	
Tail Group	4,345		27,187	
Body Group	39,005		121,018	
Induced Envir. Protection	20,946		6,207	
Landing, Recovery, Docking	11,733		22,097	
Propulsion – Ascent	22,880		86,300	
Propulsion – Cruise			29,740	
Propulsion – Auxiliary	9,783		12,118	
Prime Power	4,123		11,603	
Elect. Conver. and Distr.	2,914			
Hydra Conver. and Distr.	1,417			
Surface Controls	3,995			
Avionics	7,344			
Environmental Control	4,456		2,693	
Personnel Provisions	1,269			
Ballast Other System			19,319	
Growth/Uncertainty	13,406		33,210	
Subtotal (Dry Weight)	164,590	55,000	437,307	
Personnel	1,621		1,621	
Cargo	40,000			
Residual Fluids	2,044	3,960	52,796*	
Subtotal (Inert Weight)	208,255	58,960	491,724	
Reserve Fluids	2,393	6,532		
In-Flight Losses	4,705	4,868	18,046	
Propellant – Ascent		1,100,000	2,088,942	
Propellant – Maneuv/ACS	18,838			
Total	234,191	1,170,360	2,598,712	4,003,263

*Including 30,916 lb JP-4 fuel

TABLE 3.5-2

CONFIGURATION E COSTS

(Millions of Dollars)

Flights in Program	System Element	Nonrecur		Rec	Total	Av Rec Cost Per Flt
		Dev	Prod			
250	Orbiter	2,712	189	805	3,706	
	External Tank	202	0	469	671	
	Booster (or Droptank)	6,634	30	526	3,190	
	System	1,052	23	184	1,259	
	Total	6,600	242	1,984	8,826	7.94
		6,842				
500	Orbiter	2,712	470	1,384	4,566	
	External Tank	202	0	836	1,038	
	Booster (or Droptank)	2,634	423	970	4,027	
	System	1,052	91	327	1,470	
	Total	6,600	984	3,517	11,101	7.03
		7,584				
750	Orbiter	2,712	851	1,949	5,512	
	External Tank	202	0	1,173	1,375	
	Booster (or Droptank)	2,634	773	1,407	4,814	
	System	1,052	166	464	1,682	
	Total	6,600	1,790	4,993	13,383	6.65
		8,390				

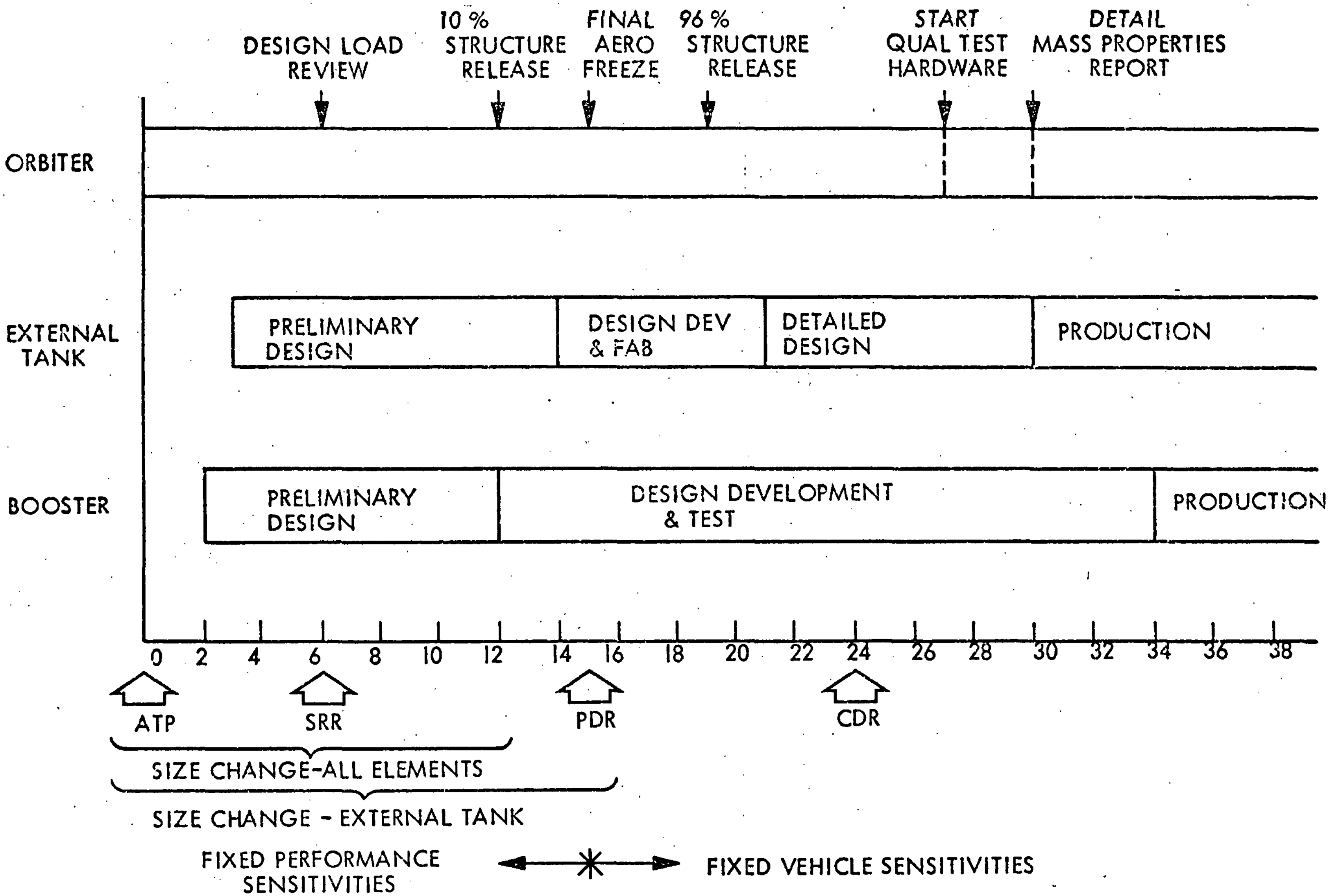

Figure 3.5-2 Typical Development Schedule - Configuration E

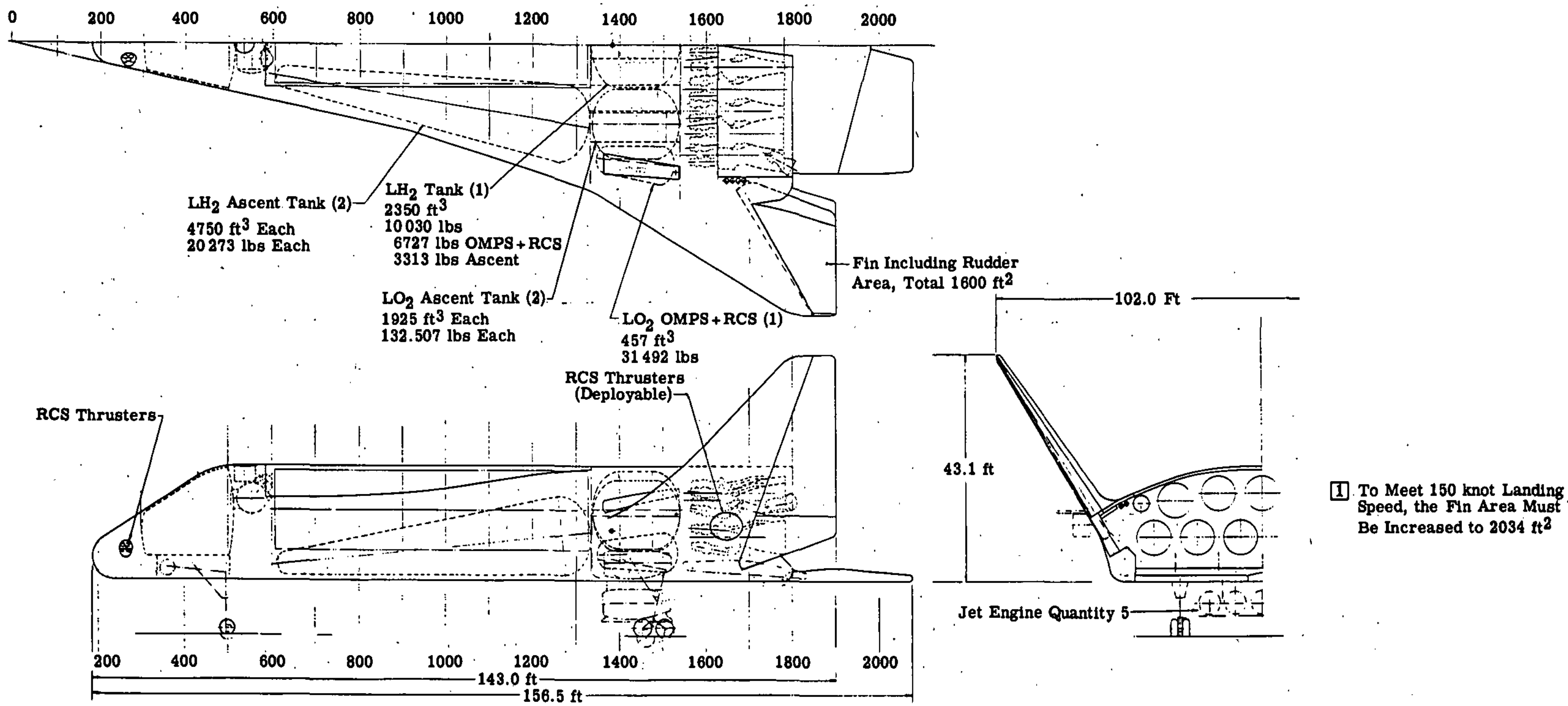

Fig. 3.1-1 Configuration A

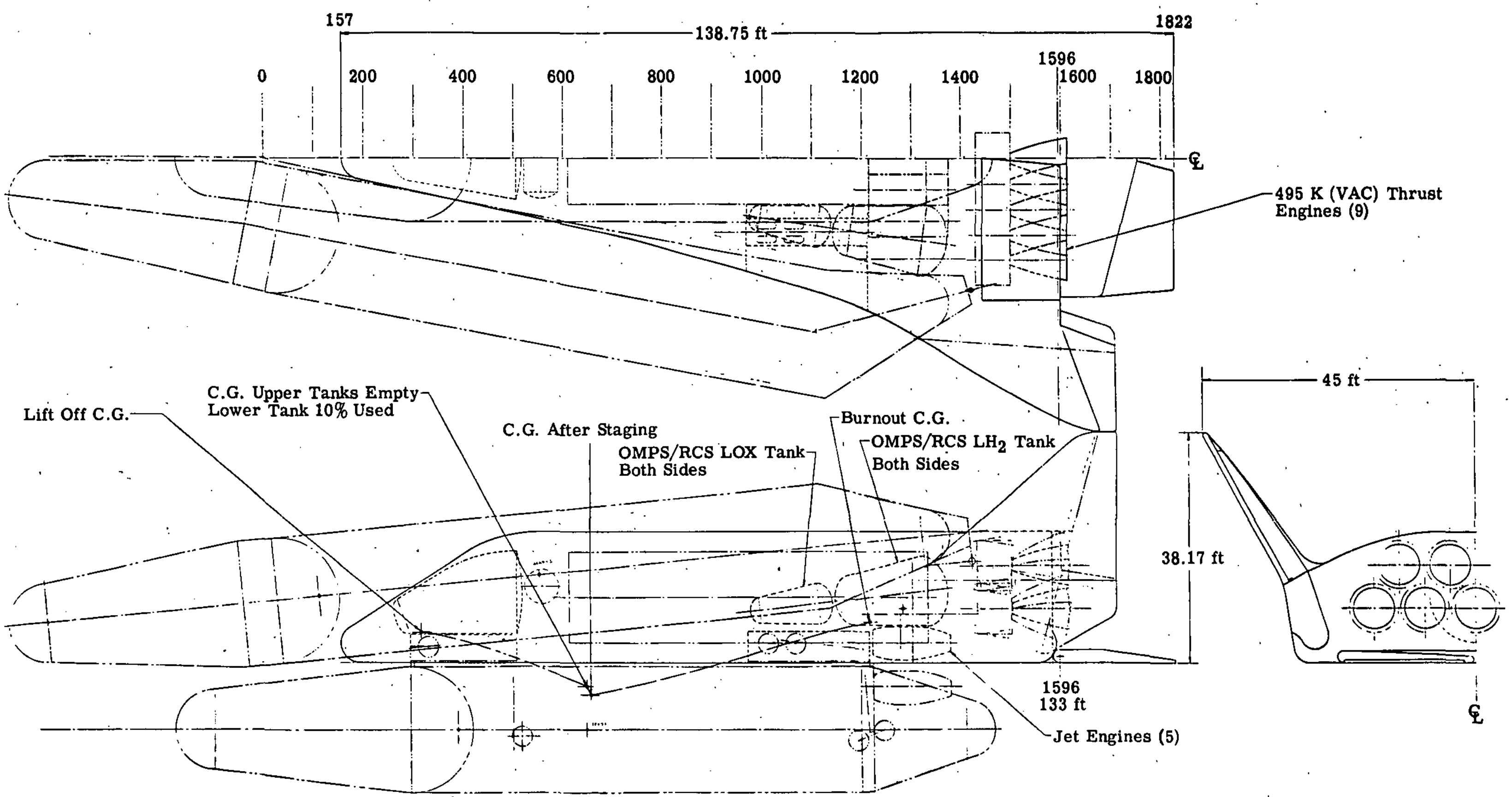

Fig. 3.2-1 Configuration B

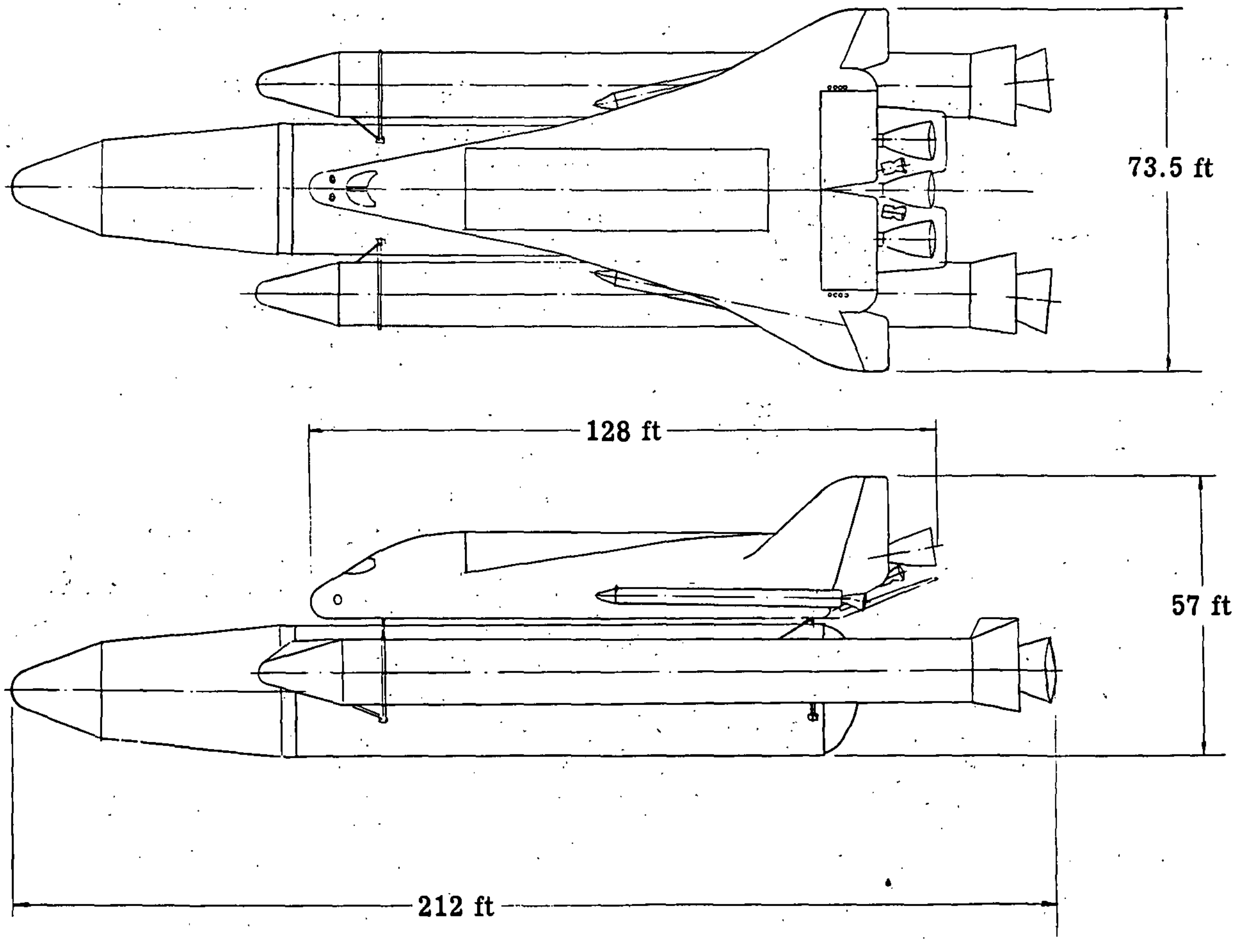

Fig. 3.3-1 Configuration C

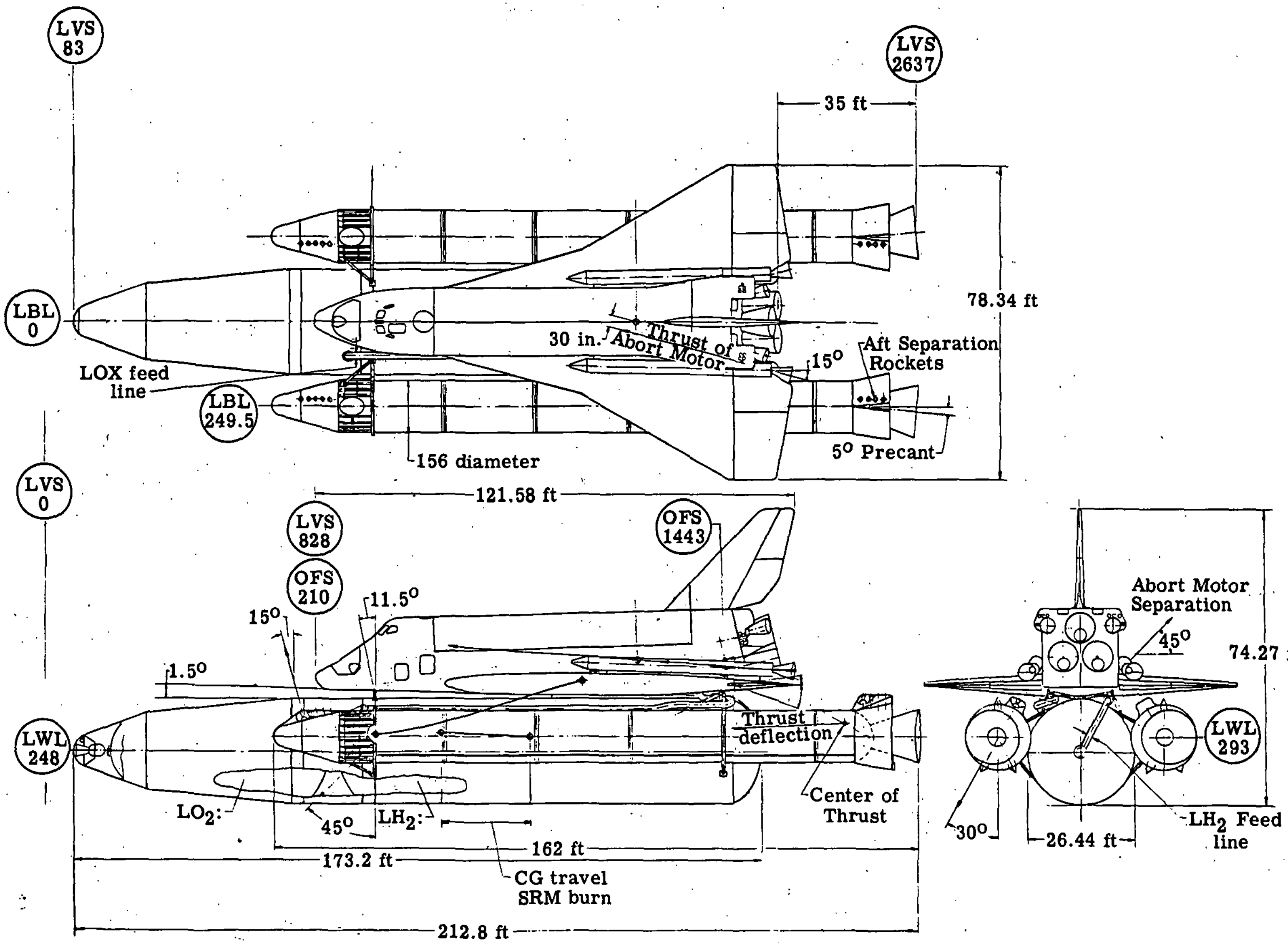

Fig. 3.4-1 Configuration D

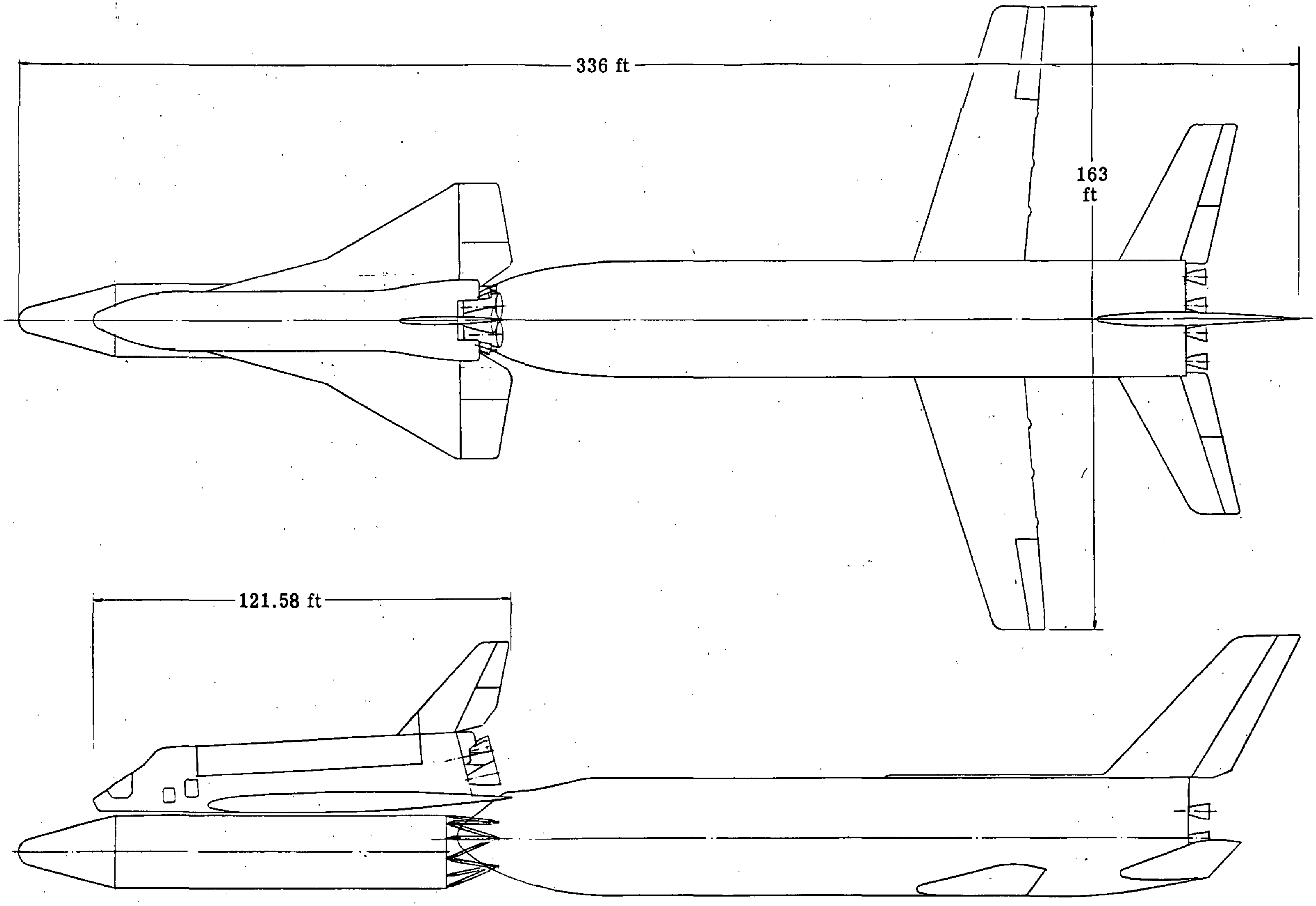

Fig. 3.5-1 Configuration E

Section 4

ANALYTICAL METHODS AND EXAMPLES

4.1 SENSITIVITY ANALYSIS OVERVIEW

Weight and cost sensitivities measure the change in these items when an independent parameter (such as weight of an element of the system) is varied, as certain design and performance re uirements are imposed. Mathematically, they may be either open- or closed-loop partial derivatives subject to a given set of constraints. An open-loop derivative is the standard partial derivative; a closed-loop derivative implies a reoptimization of the perturbed system in some sense (minimum launch weight or minimum cost).

4 1.1 Design Factors

One of the most important factors in understanding sensitivities is the effect of the design approach used in deriving them on their values. As an example, consider the situation if a subsystem of the orbiter for Configuration D undergoes a weight growth of 1000 lb relatively early in the design process, resulting in a payload loss unless the system is increased in size. Several design alternatives exist to recover the payload and other performance requirements. How may this best be done? The external tank, or solid-rocket motors, or both, may be increased in size. If both are increased, in what proportion should they be increased? Should the thrust level of the liquid engines or solid-rocket motors be changed. Which of the above options result in the minimum gross weight increase? Which options result in the minimum DDT&E, recurring, or total cost increases? Also, what is the effect if the orbiter wing size is increased to maintain crossrange? More important, what is the weight and cost increase in each element of the system? These increases in weight or cost, divided by the value of the input weight (1000 lb), determine the sensitivity of the system to weight growth. The values of the sensitivities are highly dependent on the design approach selected to maintain system performance and design requirements. These subtleties are discussed in more detail for Configuration D in Section 4.6.2.

4.1.2 Time of Weight Growth

At different times in the development program, different corrective actions will be taken to maintain system requirements. Early in the program, characteristics of all elements of the system may be changed; i.e., the system is completely rubberized. Later on, certain elements of the system are progressively frozen as the design becomes better defined. Finally, the vehicle's design is completely fixed and any further weight growth causes performance degradation. At each time point in the development program, different weight and cost sensitivities result. These differences are discussed in more detail in Section 4.4 and 4.5.

4.1.3 Derivation of Sensitivities

Weight or cost sensitivities may be classified as either fixed-capability or fixed-vehicle sensitivities. Fixed-capability sensitivities are derived by changing one or more elements of the vehicle's design to accommodate input weight changes, maintaining certain vehicle performance characteristics such as payload, on-orbit velocity, crossrange, and orbiter (or booster) landing speed. Fixed-vehicle sensitivities are derived by determining the change in payload of a vehicle when an input weight is added to an element of the system (orbiter, tank, or booster) and the vehicle's design is not changed. Fixed-capability sensitivities are used during the vehicle preliminary design and detailed design phases; fixed-vehicle sensitivities are used after the design has been frozen, such as during the manufacturing, flight test, and operational phases.

The derivation of fixed-capability weight sensitivities is considerably more complex than that of fixed-vehicle weight sensitivities. Fixed-capability sensitivities require parametric weight scaling and performance relationships which define the system and its capability over a range of different propellant loadings and thrust levels for each element in the system. These relationships may be combined in a vehicle-sizing/synthesis computer program and a solution solved for iteratively, subject to the various design constraints. A baseline vehicle is first defined and a solution obtained. The vehicle is then perturbed by adding an input weight to an element of the system (orbiter, tank, or booster) and a new solution is obtained, subject to design constraints which hold defined characteristics of the baseline unchanged (e.g., same booster or tank propellant as the baseline, minimum gross weight, same thrust or thrust-to-weight ratio, etc.). The change in weight of each element of the system from the baseline to the perturbed vehicle is then divided by the input weight to calculate the weight sensitivities. The sensitivities derived in this manner are numerical derivatives. Alternately, analytical derivatives may be determined using these parametric relationships, but the procedure is quite complex because of the number of terms that vary as a parameter is changed, especially for certain design constraints such as minimum gross weight. In this report, all weight sensitivity values were derived by using the numerical method with a vehicle-sizing/synthesis computer program. However, a further discussion of analytical sensitivities is given in Section 4.7.

After the fixed-capability weight sensitivities are determined, the weight sensitivity of each item in the system is multiplied by its direct cost sensitivity (see Section 4.3) and the terms summed to derive the cost sensitivity of the total system. The cost of the system increases in size and weight. This process is described in more detail in Section 4.4.

Fixed-vehicle weight sensitivities do not involve weight-scaling relationships, since the dry weight of all elements of the vehicle is fixed. These sensitivities can be determined directly from a computer program which calculates the vehicle's ascent trajectory (PRESTO). An optimum (maximum payload) trajectory is first run for the baseline vehicle, and then a second optimum trajectory is run for the weight-perturbed vehicle. The difference in payload is divided by the input perturbed weight to find the sensitivity.

Fixed-vehicle cost sensitivities are fundamentally different from fixed-capability cost sensitivities, in that the cost of loss of payload is considered, rather than the additional cost for a larger vehicle. Fixed-vehicle sensitivities are discussed in Section 4.5.

4.1.1 Use of Sensitivities

Weight and cost sensitivities may be used for vehicle design tradeoff studies, growth allowance analyses, and program risk studies. The role of sensitivities for these uses is shown in Figure 4.1-1. Fixed-capability sensitivities are most useful in conducting weight/cost tradeoff studies. In comparing two alternative subsystems, the first of which is higher in weight but less costly, the effect of the additional weight on the total system cost must be known (via sensitivities) to evaluate the total cost differences between the systems.

In using cost sensitivities, it is important to distinguish between "free" and "costed" input weights. Free input weight cost sensitivities do not include the cost of the input weight in the value of the sensitivity. The cost of the input weight must be accounted for separately, as it might be in a weight/cost trade study, where the input weight might be a difference in weight between two subsystems, the costs of both of which would be known. Sensitivities utilizing costed inputs assume that the input weight is additional vehicle structure at a certain cost per pound (DDT&E, production and operations). This direct cost of the input weight is added to the indirect cost using the free input weight cost sensitivity. This type of cost sensitivity is used for growth allowance analyses and program risk studies, where uncertainties in weights must be translated to uncertainties in costs. Since the free input weight cost sensitivity (which includes cost effects for indirect growth but not for the input weight itself), is more useful, and since the costed input weight sensitivity (which assumes the input weight

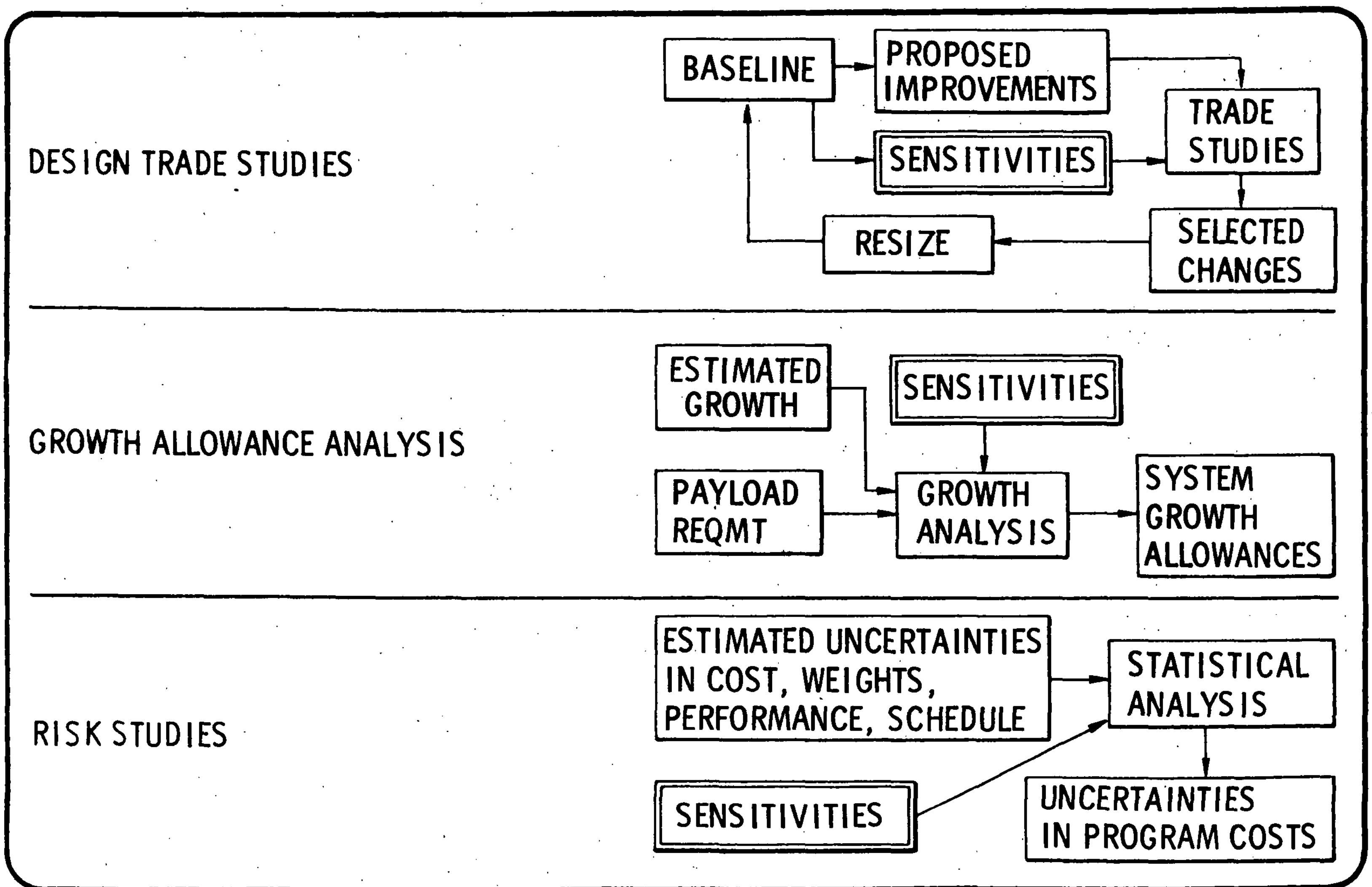

Figure 4.1-1 Applications of Sensitivities

to be costed as structural growth of the vehicle element involved) may be easily determined from the free input weight sensitivity, major emphasis in this report is placed on the free input weight cost sensitivities.

4.2 PARAMETRIC DESIGN RELATIONSHIPS

For the fixed-capability sensitivities, weight-scaling and performance relationships are required to predict the inert weight, specific impulse, and velocity requirements of the system as propellant loading, engine thrust, and input weights vary for the various elements of each system. The specific design relationships themselves are not given in this report.

The basic use of these parametric design relationships is to synthesize a vehicle, stage by stage, that performs the specified mission subject to certain design requirements. This is done by utilizing the basic performance equation.

(1) $\Delta V = g\, I_{sp_i},\ \mu_i = W_{IGN_i}/W_{BO_i}$ where ΔV is the required ideal velocity, I_{sp_i} is the vacuum specific impulse of stage i, and W_{IGN_i} and W_{BO_i} are the respective ignition weight and burnout weights of stage i. A stage is defined as a portion of flight between inert weight drops. The relationships between ignition and burnout weights, and propellant and inert (dry plus payload plus nonpropulsive fluids) weights are given as:

(2) $W_{IGN_i} = W_{BO_i} + W_{P_i}$

(3) $WBO_i = W_{IGN_{i-1}} + W_{IN_i}$

where W_{P_i} and W_{IN_i} are the impulse propellant and inert weights of stage i. For a two-stage vehicle (1) becomes:

(4) $$\Delta V = gIsp_1 \frac{W_{IN_1} + W_{P_1} + W_{IN_2} + W_{P_2}}{W_{IN_1} + W_{IN_2} + W_{P_2}} + g\,I_{sp_2} \frac{W_{IN_2} + W_{P_2}}{W_{IN_2}}$$

Notice that for a constant ΔV, if W_{IN_2} increases by adding an input weight to it, then either W_{P_2} or W_{P_1} or both must increase to maintain the system's velocity capability.

The inert weight of stage i is in general an increasing function of its propellant loading, so that as the propellant increases to maintain the system velocity (ΔV), the inert weight of the stage increases, causing an additional increase in propellant loading (over and above the initial input increase in inert weight). This additional increment is denoted as an indirect increase in inert weight, as opposed to the direct increase (the input weight).

In general, the relationship of a stage's inert weight to its propellant loading, the input weight, and any other parameters such as thrust level of engines, staging velocity (flyback booster for Configuration E) and other stage gross weights (structural loading) is called a weight-scaling relationship. These have been defined for all elements in the system for each configuration, and are further discussed in Section 4.2.1.

The ideal velocity requirement (ΔV) is in general not a constant, but depends upon the shape of the ascent trajectory. This ideal velocity (sometimes called total velocity) is actually a sum of the actual relative (aerodynamic) velocity at orbit injection, plus various velocity loss terms (gravity, attitude, drag, and engine backpressure). When the acceleration time history of a vehicle changes, its optimum (maximum payload) trajectory shape also changes; this can be reflected in differences in its velocity losses, or equivalently, its ideal velocity requirement. Vehicle characteristics which affect the vehicle's acceleration time history are thrust, weight, and drag characteristics. As an example, for fixed-thrust engines, whenever a weight is increased, the thrust-to-weight ratio decreases at all times in the trajectory (except at the 3g acceleration limit) and additional gravity losses result, increasing the ideal velocity requirement. Thus the ideal velocity requirement for a given injection orbit can be expressed as a function of the vehicle's thrust-to-weight ratio at launch and at staging, its staging velocity, and its drag-to-weight ratio, all of which uniquely determine its acceleration history for given specific impulse of each stage and the maximum acceleration limit (assumed to be 3g). The ideal velocity's dependence upon these parameters is discussed in more detail in Section 4.2.2.1 All of the parametric data used in predicting ideal velocity requirements were generated with the use of an optimized ascent trajectory computer program, PRESTO.

Using the weight-scaling and velocity relationships, the basic performance equation (1) may be solved iteratively. A baseline vehicle is first defined and a solution obtained. For fixed-capability sensitivities, an input weight is added to the inert weight of a stage and a new solution is sought, resulting in new propellant loadings and inert weights for some or all of the stages, depending upon the design constraints imposed. This solution for the perturbed case reflects a new design for some or all of the stages and a new reoptimized ascent trajectory. Using the parametric design relationships for weight and velocity allows a rapid, reasonably accurate solution to the vehicle redesign problem which incorporates many complex interactions and satisfies all performance requirements yet does not require individual ascent trajectories to be generated for each sensitivity. This method of handling weight and performance characteristics of a vehicle in a single vehicle sizing/synthesis computer program has been used by LMSC successfully for the past six years on a wide variety of candidate Space Shuttle configurations.

4.2.1 WEIGHT-SCALING RELATIONSHIPS

Weight-scaling relationships express an element's inert weight (dry weight plus nonpropulsive propellants plus payload, if applicable) as a function of other vehicle design variables, the most important of which are propellant loading and engine thrust. These exist for all elements of the vehicle (orbiter, external tank, solid-rocket motor, flyback booster) for each configuration. They have been derived either from analytical considerations, empirical data (including parametric point-design data), or a combination of the two. The most common procedure is to determine a functional relationship by analytical considerations, and then to calibrate coefficients in the functional relationship by comparison with one or more design points. For example, an internal tank may be designed by pressure so that its weight is proportional to its volume, or equivalently, its propellant load, i.e.,

$$\text{Tank Wt} = k \times \text{Propellant Load}$$

Then the coefficient **k** could be derived from a single point-design tank of this type (from its weight and propellant load). Often, examination of a detailed weight statement of a single point-design will allow weights which vary with a parameter to be separated from weights which are independent of that parameter, thus allowing two coefficients to be calibrated from a single point-design.

In the following sections, a discussion of the functional form of the weight-scaling relationships of each major element found in any of the configurations under study is given. In general the values of the coefficients for any given weight-scaling relationship will be different for each configuration and may not be applicable to some configurations.

4.2.1.1 ORBITER. The inert weight of the orbiter for all configurations must be given as a function of input weight (carried throughout the flight, including landing), and main engine vacuum thrust (Configurations A, B, C, and E). To determine these effects accurately, certain subsystem weights must be expressed as a function of orbit maneuvering and reaction control propellant, fin area (Configurations A, B, and C) or wing area (Configurations D and E), and landing weight. The fin or wing area is itself a function of landing weight (so that landing speed is maintained) and landing center-of-gravity (so that stability margin is maintained). The functional form of the weight-scaling relationships for each of the subsystems in the orbiter is given in the following paragraphs, along with a short discussion of how the relationwhip was obtained. Only those subsystems whose weight changes during the generation of a sensitivity are discussed; the constant weight terms have no effect on the value of the sensitivity.

WING GROUP. Configurations D and E (delta-wing orbiters) have a wing designed by the landing speed requirement. Configurations A, B, and C (delta-body orbiters) do not have a wing as such; the fins on these configurations are used to control landing speed. In both cases, the freestream landing speed (V_L) can be expressed as:

$$(1) \quad V_L = \sqrt{\frac{2W_L}{\rho C_L S}}$$

where W_L is the landing weight of the orbiter, ρ the sea-level atmospheric density, C_L the orbiter's lift coefficient at tailscrape attitude, and S the orbiter's aerodynamic reference area, which is related to the exposed wing area. For Configurations D and E, the relation between $C_L S$ and exposed wing area A_W is given as:

$$(2) \quad C_L S = 287 + 0.99 A_W.$$

From equations (1) and (2), the exposed wing area can be expressed in terms of the landing weight, where the requirement that the minimum freestream landing speed be 150 knots is met. Thus

$$(3) \quad A_W = 0.0133W_L - 290$$

The unit weight/area of the wing increases with landing weight nearly linearly so that the wing weight plus control surfaces can be expressed as:

$$(4) \quad \text{Wing Weight} = 6704 + 430 \times 10^{-5} A_W W_L^{.93}$$

The coefficients and the exponent on landing weight were determined from detailed structural analysis and was based on the assumption that if the wing area were to remain constant while the landing weight increased, then the increased loads on the wing would be reflected in heavier spars and webs, but the depth of the wing would be unchanged. Note that if the landing weight of the orbiter increases, as when an input weight is added, then the wing weight increases due to two different effects: the increase in wing area to maintain landing speed, and the increase in wing weight even if wing area is held constant to maintain the structural integrity through heavier loadings.

TAIL GROUP. For Configurations D and E, the tail weight is independent of the parameters under consideration (input weight, propellant, thrust, etc.). For Configurations A, B, and C, the delta-body configurations, the side fins are utilized to control the landing speed as well as balance the vehicle for pitch stability. This additional requirement, not explicitly considered for the delta-wing orbiter, is important especially for Configurations A and B because in these configurations orbiter engine thrust is increased when weight growth occurs, causing an aft shift in the center-of-gravity which must be compensated for by a change in side fin area and toe-in angle.

For Configuration C, the fin area is determined by the relation:

(5) $A_F = 0.0074\ W_L + 0.009\ X\ \ \Delta CG - 529$

where A_F is the total fin area (both sides), W_L is the orbiter landing weight and ΔCG is the shift in center-of-gravity from the baseline system. The fin area for Configurations A and B is predicted by similar relations, with identical coefficients for W_L and ΔCG but a different constant term.

When Equation (5) for the delta-body configurations is compared with Equation (3) for the delta-wing configurations, it can be seen that the delta-body's fin area is less sensitive to landing weight than the delta-wing's wing area. The principal reason for this is that much of the increased lift required to maintain landing speed is provided in the delta-body by altering the fin's toe-in angle rather than increasing its weight, which results in negligible weight gain.

For the delta-body orbiters (Configurations A, B, and C), the total aerodynamic surface group weight is given by Equation (6):

(6) Aerosurface weight $= k_1 + k_2\ A_F$

where k_1 and k_2 are different constants for each configuration. Note that changes in landing weight do not affect the weight/unit of the fin or flap on the delta-body orbiters, whereas the wing weight/unit area was strongly affected by landing weight for the delta-wing orbiter. The fins on the delta-body are designed by maximum αq and βq loads rather than landing loads.

BODY GROUP. The body group structure weight for the orbiters can be conveniently separated into four categories for the purpose of this analysis: (1) cabin, skin, and minor frames, (2) major landing frames, (3) thrust structure, and (4) fin support frame (delta-body orbiters only).

For a given configuration, the cabin, skin, and minor frames are not a function of the parameters under consideration. The major landing frames are scaled with orbiter landing weight to the 0.5 power; thrust structure weight is proportional to the sea-level thrust level of the main engines. For the delta-body orbiters, the fin support frame weight is proportional to the fin weight to the 0.5 power.

THERMAL PROTECTION SYSTEM. The thermal protection system weight of the orbiter increases with the orbiter's wetted surface area and increased flight time during reentry. Increased flight time, and thus total heat load, may be caused by an increase in the parameter $W_E/C_L S$ (reentry weight divided by hypersonic lift coefficient and aerodynamic reference area). For a fixed aerodynamic shape, C_L and S are constant so that the parameter $W_E/C_L S$ increases as reentry weight (W_E) increases.

For the delta-body orbiter (Configurations A, B, and C), as reentry weight increases, fin area also increases (see the section "Tail Group"), thus changing the value of the subsonic lift coefficient to maintain landing speed. This alteration has very little effect on the hypersonic lift coefficient, however, since the body of the delta-body orbiter provides nearly all of the lift at hypersonic speeds. Thus, when an input weight is added to the orbiter, the parameter $W_E/C_L S$ increases linearly with reentry weight so that the reentry time increases to maintain the crossrange requirement (for the same temperature constraints), causing an increase in the thickness of the thermal protection system. This effect can be approximated by scaling the thermal protection system weight to the 0.5 power of reentry wing-loading (W_E/S), based on the results of previous reentry trajectories.

For the delta-wing orbiter (Configurations D and E), the wing area increases with landing weight (or equivalently, reentry weight) to maintain landing speed. Thus the parameter $W_L/C_L S$ (landing weight divided by subsonic lift coefficient and aerodynamic reference area) remains constant as landing weight increases. For the delta-wing orbiter, the hypersonic lift coefficient increases as wing area increases at nearly the same ratio as the subsonic lift coefficient increases, since most of the lift is provided by the wing, rather than the body, at hypersonic speeds. Thus the parameter $W_E/C_L S$ remains nearly constant as weight is added to the orbiter because of wing growth, and the total heat load and thus the thermal protection system thickness does not change. The area covered by the thermal protection system does increase, however, since the area of the wing increases.

PROPULSION. The ascent propulsion system for the orbiter is a function of the main engine thrust level for a given configuration and a fixed number of engines. The main engine weight is a linear function of sea-level thrust (obtained from parametric Rocketdyne engine data). Plumbing is scaled by thrust to the 0.8 power, and various other subsystems are held constant.

The cruise propulsion system is not onboard for the critical sizing mission (polar), so no scaling laws were developed. The orbit maneuvering system engines and the reaction control system thrusters were held constant as the orbiter weight changed, but the propellant tankage was assumed proportional to the propellant used, which is a function of the orbiter's weight in orbit. The delta-body orbiters have an integrated H_2/O_2 OMS/RCS system, whereas the delta-wing orbiters have an N_2O_4/MMH OMS system and a hydrazine RCS system. The greater volume available in the delta-body orbiters allows the H_2/O_2 system to be effectively utilized. The integrated system used in the delta-body has a much lower sensitivity to weight growth in the orbiter than the separate system used in the delta-wing, as well as a lower baseline dry weight.

OTHER SUBSYSTEMS. Other subsystems in the orbiter are either constant or a function of landing weight. The landing gear is proportional to landing weight (for constant landing speed). The hydraulic and surface control system weights are linear with reentry weight and fin area (delta-body orbiter) or wing area (delta-wing orbiter). Prime power, electrical conversion and distribution, avionics, environmental control, and personnel provisions are constant. Contingency, 10 percent of the orbiter dry weight less main engines, increases when any other subsystem increases.

FLUIDS. Orbiter nonimpulsive fluids such as residuals, reserves, and inflight losses are scaled linearly with tank volumes and main engine thrust level, wherever appropriate. Engine startup and shutdown losses and manifold losses are proportional to engine thrust level. Tank residuals are proportional to tank volumes.

Orbiter impulse propellants are computed using the basic performance relation (Equation (1) in Section 4.2) for each burn, with a weight sequence defined from launch to landing with approximately 15 weight drops between engine burns. OMS and RCS propellants increase with the orbiter weight at the time of the burn. The only configuration with internal ascent propellants, Configuration A, has a constant ascent propellant load, consistent with the internal volume available. Determination of the ascent propellant for the external tank is discussed in the Section 4.2.1.2.

ORBITER CONFIGURATION COMPARISON. A summary of the sensitivities of the various orbiter subsystems to an input weight carried in the orbiter from launch to landing is given in Tables 4.2-1 and 4.2-2. The orbiters for Configuration C and D are used as examples. The main engine thrust level is held constant in both cases.

The sensitivity of orbiter dry weight to input weight is considerably higher for the delta-wing orbiter than the delta-body orbiter (0.452 versus 0.287). This results primarily from the higher sensitivity of the wing of the

Table 4.2-1
SENSITIVITY OF DELTA-BODY ORBITER WEIGHT
TO INPUT WEIGHT (MAIN ENGINE THRUST CONSTANT)

CONFIGURATION C

ITEM		BASELINE WEIGHT LB.	MODIFIED WEIGHT INPUT WEIGHT = 1000 LB.	Δ ITEM / Δ INPUT WT.
1	Wing Group	0	0	0
2	Tail Group	12,353	12,423	0.070
3	Body Group	46,091	46,118	0.027
4	Thermal Protection	23,200	23,301	0.101
5	Landing/Docking	10,746	10,796	0.050
6	Propulsion - Ascent	22,881	22,881	0
7	Propulsion - Cruise	0	0	0
8	Propulsion - OMS/RCS	6,924	6,936	0.012
9	Prime Power	4,123	4,123	0
10	Elect. Conv/Dist.	2,810	2,810	0
11	Hyd. Conv/Dist	1,773	1,775	0.002
12	Surface Controls	2,620	2,628	0.008
13	Avionics	7,344	7,344	0
14	Environment Control	4,456	4,456	0
15	Personnel Provisions	1,269	1,269	0
18	Contingency	12,946	12,973	0.027
	Dry Weight	159,536	159,833	0.297
20	Personnel	1,621	1,621	0
21	Cargo	40,000	40,000	0
23	Residuals	1,962	1,963	0.001
25	Reserves	1,624	1,630	0.005
	Input Wt	0	1,000	1.000
	Landing Weight	204,743	206,046	1.303
26	Inflight Losses	4,705	4,705	0
29	Propellant - OMS	7,475	7,521	0.046
30	Propellant - ACS	3,889	3,912	0.023
	Gross Weight	220,812	222,184	1.372

Table 4.2-2
SENSITIVITY OF DELTA-WING ORBITER WEIGHT TO INPUT WEIGHT
(MAIN ENGINE THRUST CONSTANT)

CONFIGURATION D

ITEM		BASELINE WEIGHT	MODIFIED WEIGHT INPUT WEIGHT = 1000 LB	$\frac{\Delta \text{ ITEM}}{\Delta \text{ INPUT WT.}}$
1	Wing Group	16,974	17,144	0.170
2	Tail Group	4,345	4,345	0
3	Body Group	45,005	45,067	0.062
4	Thermal Protection	20,946	21,010	0.064
5	Landing/Docking	11,733	11,787	0.054
6	Propulsion - Ascent	22,880	22,880	0
7	Propulsion - Cruisd	0	0	0
8	Propulsion - OMS	3,892	3,907	0.015
8a	Propulsion - RCS	5,891	5,926	0.035
9	Prime Power	4,123	4,123	0
10	Elect. Conv/Dist	2,914	2,914	0
11	Hyd. Conv/Dist	1,417	1,420	0.003
12	Surface Controls	3,995	4,003	0.008
13	Avionics	7,344	7,344	0
14	Environment Control	4,456	4,456	0
15	Personnel Provisions	1,269	1,269	0
18	Contingency	14,005	14,046	0.041
	Dry Weight	171,189	171,641	0.452
20	Personnel	1,621	1,621	0
21	Cargo	40,000	40,000	0
23	Residuals	2,051	2,059	0.008
25	Reserves	2,445	2,481	0.036
	Input Weight	0	1,000	1.000
	Landing Weight	217,306	218,802	1.496
26	Inflight Losses	4,705	4,705	0
29	Propellant - OMS	11,682	11,760	0.078
32	Propellant - ACS	7,747	7,801	0.054
	Gross Weight	241,410	243,068	1.628

delta-wing compared to the fin of the delta-body (see "Wing Group" and "Tail Group" above), and the higher sensitivity of the separate N_2O_4/MMH OMS and hydrazine RCS of the delta-wing orbiter compared to the integrated H_2/O_2 OMS/RCS system of the delta-body orbiter (see "Propulsion" above). The body group structure weight of the delta-wing orbiter is also more sensitive to landing weight than that of the delta-body, primarily because the large cross-section of the delta-body orbiter provides a larger bending moment-of inertia which is less sensitive to landing loads. Also much of the body structure for the delta-body orbiter is minimum gage thickness because of low loads, and is not a function of the landing and reentry loads.

The OMS and RCS propellant weight sensitivity of the delta-body is lower than that of the delta-wing because of the higher specific impulse from the H_2/O_2 propellants. The lower sensitivity in propellants is also reflected in the lower sensitivity in propellant tankage weight.

4.2.1.2 EXTERNAL TANKS

DRY WEIGHT. The H_2/O_2 external tank dry weight is in general a function of propellant load and to a lesser extent engine vacuum thrust. An increased propellant load requires a tank of larger volume, with higher structural and insulation weights. As main engine thrust is increased on the orbiter, propellant flowrate must also increase so that larger diameter feedlines are required. For all configurations, the external tank dry weight is computed by the expression:

$$\text{Tank Dry Weight} = k_1 + k_2 W_p + k_3 T$$

where W_p is the tank propellant load, T is the main engine sea-level thrust and k_1 k_2, and k_3 are constant coefficients established for each of the five configurations by comparison of parametric point designs. For Configurations A and B, the droptanks (set of tanks staged first) are also regarded as external tanks.

FLUIDS. The nonimpulsive propellants in the external tank are a function of tank volume and engine thrust level. Residuals are proportional to tank volume and main engine startup and shutdown transients. Line losses and engine vents are proportional to engine sea-level thrust (or propellant flowrate).

The impulsive H_2/O_2 propellant may be either input or calculated, depending on the design constraints. For some sensitivities, the tank size is fixed, so the propellant load and dry weight do not change. For other sensitivities, the tank size varies, and the propellant load is selected to meet the orbit injection requirements determined by the basic performance relationship (Equation (1) of Section 4.2).

4.2.1.3 SOLID ROCKET MOTORS. The 156 inch diameter solid-rocket motor (SRM) dry weight is computed on the basis of propellant load and burn time. The burn time of the SRM is determined from the propellant load, specific impulse, and thrust level of the motor. The relationship of these parameters that is actually used is complicated because the thrust-time history utilized is rather complex (to reduce peak dynamic pressure and maintain a 3g maximum acceleration). However, it may be approximated by the equation below:

$$t_B = \frac{W_p I_{sp}}{T_{av}}$$

where W_p is the propellant load, I_{sp} is the vacuum specific impulse, and T_{av} is the average vacuum thrust of the SRM. Note that the burn time increases with propellant load but decreases with thrust. In general, during the generation of weight sensitivities, both SRM propellant load and thrust level will increase (for a positive input weight), so that the burn time may either increase or decrease. For Configurations C and D, the system launch thrust-to-weight ratio is maintained for maximum performance, causing a change in SRM thrust (main engine thrust is constant).

The relationship between SRM dry weight, propellant loading, and burn time is of the form:

$$\text{SRM dry weight} \quad k_1 + k_2 W_p + k_3 t_B$$

where k_1 and k_2 are positive constants and k_3 is a negative constant. An increase in propellant loading causes an increase in SRM length (diameter is constant at 156.in.) and an increase in case weight. An increase in burn time implies a decrease in throat diameter so that the volumetric efficiency inside the case increases. This allows a smaller case length to be used for the same propellant loading, causing a decrease in case dry weight. Changes in either propellant loading or burn time thus imply a complete motor re-design. It is assumed that the maximum expected operating pressure (MEOP) and the nozzle exit diameter are not changed when either propellant loading or burn time are changed.

4.2.1.4 HEAT SINK BOOSTER. The heat sink booster, used in Configuration E, is scaled from data generated during the Alternate Space Shuttle Concepts (ASSC) study performed by Grumman/Boeing during 1970 and 1971. This booster design was valid for staging velocities of less than 7000 ft/sec. A linearized equation was derived from these data of the form:

$$\text{Dry Weight} - k_1 + k_2 W_P + k_3 \, (T/W)_0 + k_4 V_{stage} + k_5 \cdot W_{IGN_2}$$

where W_P is the booster's propellant load, $(T/W)_0$ is the system's thrust-to-weight ratio at launch, V_{stage} is the relative (aerodynamic) staging velocity of the booster, and W_{IGN_2} is the ignition weight of the orbiter plus external tank. The coefficients k_1, k_2, k_3, k_4, and k_5 are all positive constants. Dry weight increases (1) with propellant load, because of the larger surface area and loads on the booster; (2) with thrust-to-weight ratio, because of the greater engine thrust and higher peak dynamic pressures encountered; (3) with staging velocity, because of the higher reentry temperatures and longer cruiseback range; and (4) with orbiter/tank ignition weight because of the higher structural loads (and indirectly, because of the greater engine thrust required to maintain the same thrust-to-weight ratio). This equation for booster dry weight does not give information about the subsystem weight breakdown. For costing purposes, the major weight elements contributing to cost sensitivity (structure, thermal protection, plumbing, and main engine weight) were also scaled with propellant loading, thrust-to-weight ratio, staging velocity, and orbiter/tank ignition weight.

The cruiseback fuel (JP-4) increases linearly with booster cruiseback weight and range. The cruiseback range is a nonlinear function of staging velocity. Residuals are proportional to impulse propellant volume (and thus weight), and other inflight losses are proportional to main engine thrust level. The main impulse propellant is determined from the basic performance equation.

4.2.2 PERFORMANCE RELATIONSHIPS

In the beginning of Section 4.2, it was stated that using the basic performance equation (1) any launch system can be sized and sensitivities derived if necessary, if certain other relations are established. The key items are the inert-weight scaling laws, the dependence of the ideal velocity requirement on vehicle characteristics, and the specific impulse variation (if any) with certain design variables. In this section, these latter two items will be discussed.

4.2.2.1 VELOCITY REQUIREMENTS. The energy required to achieve orbit injection can be conveniently expressed in terms of the ideal (total) velocity required. To get an accurate value for this parameter, it is necessary to run a computer simulation of the optimized (maximum payload) ascent trajectory with all of the pertinent weight and performance characteristics of the vehicle simulated. Whenever the weight, thrust, or aerodynamic drag of the vehicle is changed, the ideal velocity required for orbit injection changes, because the vehicle's acceleration history changes. It is the applied acceleration history which determines the ideal velocity requirements. The applied acceleration is defined as

$$\text{Applied Acceleration} = \frac{T(t) - D(t)}{W(t)}$$

where the vehicle's thrust is T, its drag is D, and its weight is W at time t.

For Space Shuttle-type vehicles, the parameter T/W (thrust-to-weight ratio) is the dominant factor in the applied acceleration term. All of the configurations in this study have two ascent stages, both of which are limited to a 3g maximum axial acceleration; i.e.,

$$T/W \leq 3$$

The value of T/W is $(T/W)_o$ at launch $(t = o)$ and increases as the vehicle's weight decreases, the thrust increasing slightly with altitude (except for Configurations C and D in which the solid-rocket motor thrust varies to decrease peak dynamic pressure). When the 3g acceleration limit is reached, the thrust is reduced to maintain the acceleration limit. At staging, the thrust level is usually reduced to below the maximum acceleration and the process is repeated. A representative thrust-to-weight history is shown in Figure 4.2-1 below:

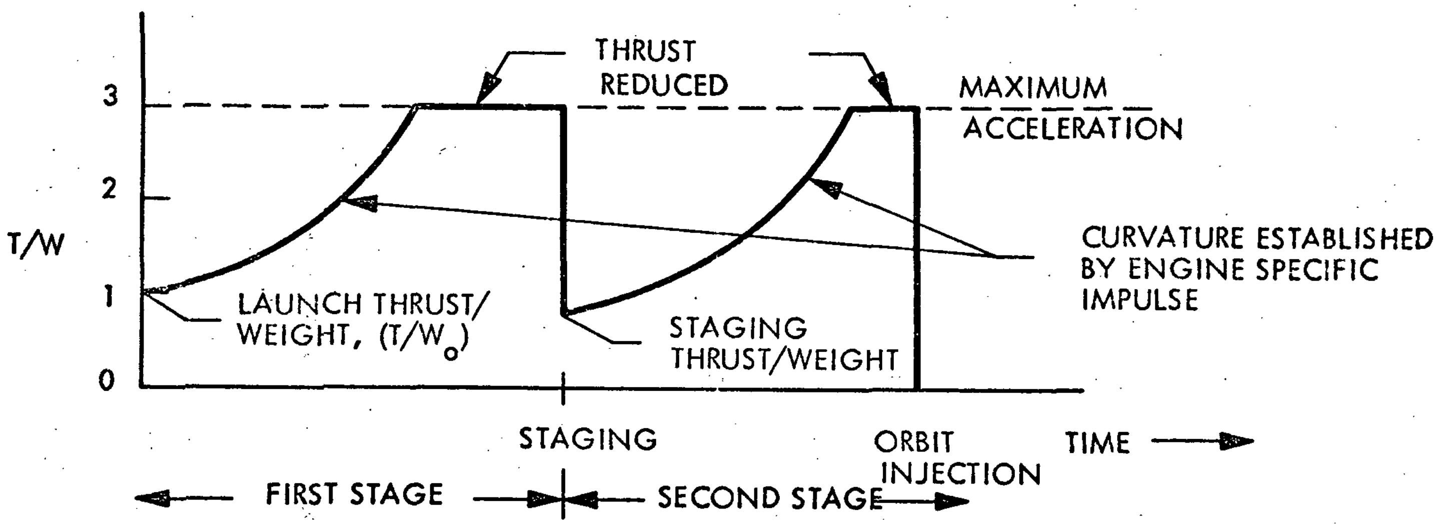

Figure 4.2-1 Typical Thrust/Weight History

For this type of history, the profile can be established using only five parameters: launch thrust/weight, staging thrust/weight, staging time, and the specific impulse of both stages. The time of orbit injection can be determined from the ideal velocity requirement. Similarly, the time of staging can be calculated from the ideal velocity obtained from the first stage, using the equation below:

$$\Delta V_{Staging} = \int_{o}^{t_{staging}} T(t)/W(t) \; dt$$

For a given configuration, the specific impulse does not vary at all or only slightly during generation of a sensitivity so that the variation of ideal velocity with specific impulse need not be determined. Note that specific impulse determines the propellant flowrate, and thus the curvature of the increase in T/W with time. T/W for a given configuration as a function of time is completely determined by the launch and staging thrust/weight, and staging ideal velocity.

The other part of the applied acceleration term, drag/weight (D/W), is determined approximately if the drag coefficient at peak dynamic pressure ($C_{D_{max}}$), the aerodynamic reference area , the peak dynamic pressure (q_{max}), and the weight of the vehicle at time of peak dynamic pressure are all known. For a given configuration q_{max} is primarily a function of $(T/W)_0$.

The ideal velocity required for orbit injection for a given mission and a given configuration can be computed for a wide variation in vehicle weights and thrust if the following four parameters are known:

(1) launch thrust/weight, (2) staging thrust/weight, (3) staging ideal velocity, and (4) $C_{D_{max}}S/W$.

Fortunately, these parameters are easily computed from known vehicle characteristics. The procedure then is to determine these ideal velocity requirements for the polar mission from optimized ascent trajectory simulations using the PRESTO (Preliminary Rapid Earth-to-Space Trajectory Optimization) computer program over a range of these four parameters for each configuration.

The variation of ideal velocity required with these parameters can be easily explained in terms of the shape of the optimum ascent trajectory. As either the launch or staging thrust/weight decreases, or staging velocity decreases, the vehicle flies more vertically early in the trajectory causing a "roller-coaster" effect in the latter part of the trajectory (see Figure 4.2-2).

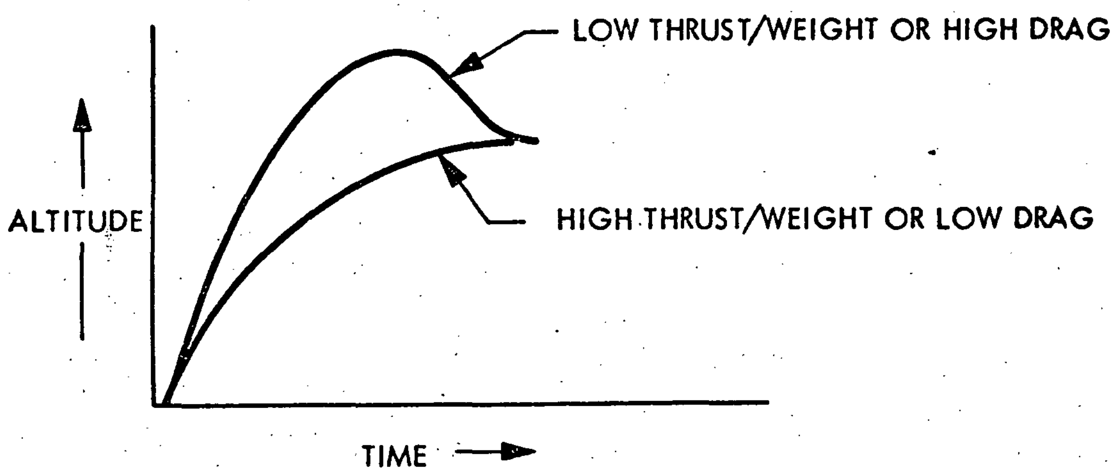

Figure 4.2-2 Ascent Trajectory Shapes

This maneuver is necessary to keep the vehicle aloft with a lower spplied acceleration, but increases the gravity velocity losses and thus the ideal velocity required. Higher thrust/weight ratios cause flatter trajectories. Increasing the drag causes a more vertical trajectory also, causing greater gravity as well as drag velocity losses.

On-orbit velocities provided by the OMS and RCS system are independent of vehicle characteristics, and so are known from mission requirements.

4.2.2.2 SPECIFIC IMPULSE. The specific impulse (I_{sp}) of the engines of the various elements for each of the configurations show little variation with vehicle parameters. The vacuum specific impulse of the main engines varies slightly with sea-level thrust level, with an increase of 0.1 sec in I_{sp} per 100,000 lbf change in engine thrust level. The specific impulses of all OMS and RCS systems are constant since the engine thrust levels are fixed.

The solid-rocket motors show the greatest variation of specific impulse, since it is a function of both propellant load and burn time, given by the relation below:

$$I_{sp_V} = 262.67 - 1.05 \times 10^{-5} (W_P - 2869000) + .0943 (t_B - 131.2)$$

where I_{sp_V} is the vacuum specific impulse, W_P is the propellant load and t_B is the burn time. As propellant load increases, the length of the SRM decreases (diameter held constant at 156 in.) causing a decrease in nozzle expansion ratio (nozzle exit diameter limited to 156 in.) and a decrease in vacuum specific impulse. Increasing the burn time decreases the throat diameter, increases the nozzle expansion ratio and increases the vacuum specific impulse. It is assumed that the SRM maximum expected operating pressure (MEOP) does not change with propellant load or burn time.

4.3 COSTING METHODS

There are basically two ways to estimate Space Shuttle costs. Parametric costing based on Cost Estimating Relationships (CERs), and bottom-up costing based on detailed projected manhour, materials, and facility requirements. Bottom-up costing is more accurate but requires a degree of design and program definition that is impractical to achieve in preliminary design. Also, it is impractical to derive cost sensitivities to design changes from bottom-up costing. For trade and optimization studies, parametric cost estimation is necessary when cost elements are linked by simple relationships to appropriate system design parameters called "cost drivers". These CERs are usually simple, single-parameter expressions which can be easily evaluated. Also, for purposes of sensitivity studies, they can be easily differentiated.

4.3.1 Assessment and Updating of Cost Estimating Relationships

Cost Estimating Relationships (CERs) are derived from historical data or specific bottom-up cost estimates by making simplifying assumptions that permit generation of the data. Cost estimates made on the basis of CERs therefore have fairly large uncertainty. Much of the uncertainty, however, is of a systematic nature over the range of costs for which the CER is valid. Therefore, the cost derivatives can be expected to reflect cost sensitivities to design changes with somewhat greater accuracy.

As this study pivots around the generation of cost/design sensitivities, primary emphasis in updating existing CERs has been placed on cost slopes rather than magnitude.

For a typical parallel staged solid-boosted orbiter with external tank (similar to Configuration D), the significance of the major cost contributors was assessed by calculating their contribution to cost increments due to adding 1 lb of inert weight to the orbiter (fixed performance sensitivities) as shown in Table 4.3-1.

Table 4.3-1

TYPICAL COST SENSITIVITIES DUE TO WEIGHT ADDITION TO ORBITER
(ROCKET-ASSISTED ORBITER, 500 FLT. PROGRAM)

	d RDT&E / d WORB	% OF TOTAL	d RECURRING / d WORB	% OF TOTAL	d PROG COST / d WORB	% OF TOTAL
	$/LB	%	$/LB	%	$/LB	%
Orbiter	$\boxed{592}$	2.2 (20.6)	441	1.7 (15.6)	1033	3.9 (36.2)
External Tank	379	1.4 (2.5)	$\boxed{4653}$	17.5 (9.4)	5032	18.9 (11.9)
Solid Rocket	816	3.0 (2.2)	$\boxed{17782}$	68.0 (30.1)	18598	70.1 (32.3)
Subtotal	1787	6.7 (25.3)	22876	86.2 (55.1)	24663	92.9 (80.4)
Other		(11.5)		(8.3)	1881	7.1 (19.6)
Total		(36.6)		(63.4)	26544	100

Note: (x) = % of program cost represented by this element.

In addition to giving the percentage distribution of the cost sensitivities, the table gives, in parentheses, the percentage distribution of program cost that is accounted for by the same CERs. It is interesting to note that while the major vehicle elements account only for some 80 percent of baseline cost, they do account for 93 percent of the cost increment due to resizing. Also, while the orbiter accounts for 36 percent of cost, it contributes only 4 percent of the cost increment. This drastic switch is, of course, due to some extent to the selected concept with its high degree of hardware attrition in both solid rockets and external tanks. A fully reusable booster will give more weight to the RDT&E CERs because the booster CERs are largely identical with the orbiter CERs.

The fact remains that there are many cost elements in the cost program that will not change as a result of vehicle resizing, and that a small number of CERs will reflect almost the entire cost increment.

Rather than trying to update CERs across the board, which would have been beyond the scope and means of this study, the three most significant CERs were picked for critical review and updating. These are:

- Orbiter Structure Design and Engineering ($519 of the $592 shown in Table 4.3-1)
- External Tank Recurring Production ($4286 of the $4653)
- Solid-Rocket Recurring Production and Refurbishment (the entire $17,782)

4.3.1.1 <u>Orbiter Structure Design and Engineering Costs.</u> The cost of orbiter airframe development is reflected by two CERs in the LMSC cost program:

- Orbiter Structure Design and Engineering
- Main Propulsion Design and Engineering

Of these, the second item accounts for only about 17 percent of the first in terms of cost and about 14 percent in terms of cost increment due to orbiter weight change. We will therefore concentrate on discussing the Orbiter Structure Design and Engineering CER, which was previously listed as:

$$OSDC = 4.35\ (10^{-2})\ \ (OWS)^{\ 0.762}$$

with costs in $\$10^6$ and orbiter dry structure weight, OWS, in pounds.

This CER was originally extrapolated from a subsonic airframe CER by estimating the additional requirements primarily for aerodynamic configuration development. The high-speed thermal environment does not greatly affect this CER, since the airplane structure, exclusive of its heat shield, can be considered similar to that of a subsonic airplane.

Heat shield development is accounted for under a separate CER. At the time this CER was evolved, a 50 percent increase was allowed for structural complexity and testing and about $90 million for aerodynamic configuration development (mostly wind-tunnel testing). At the design point, this brought the total up to 2.5 times the subsonic airplane CER Value. Since magnitude of cost rather than slope was of primary concern, the coefficient in the CER was multiplied by that amount. This implied, however, that the cost of the aerodynamics program was a strong function of vehicle weight, contributing to a steep cost slope. Further investigation showed, however, that the cost of the aerodynamics program is essentially independent of vehicle weight as long as no switch in the mode of testing is involved. The $90 million for the aerodynamics program has therefore been extracted from the CER and expressed in terms of cost per wind-tunnel hour.

The new Orbiter Structure Design and Engineering CER, brought up to 1971-dollar level is:

$$\mathrm{OSDC} = 2.795 \ (10^{-2}) \ (\mathrm{OWS})^{0.762} + 2 \ (10^{-3}) \ (\mathrm{HR})$$

At an OWS of 72,956 lb and 45,000 hr (HR = hours of wind-tunnel occupancy) the cost slope is reduced from $2307/lb to $1482/lb.

A comparison of the old and the new CERs is shown in Figure 4.3-1.

Prior to developing the above logic for an improved CER for structure development, other sources of data were reviewed which gave indications that the slope of the previous CER was too great. A source of particular interest is an unpublished report entitled "A Model for Estimating Total Program Cost of Aircraft, Spacecraft, and Reusable Launch Vehicles," prepared in 1971 by Darrell E. Wilcox of NASA OART, Advanced Concepts and Missions Division. This paper gives cost estimating relationships based on data from high-speed aircraft and spacecraft programs including X-15, XB-7, XF-104, XF-106, BGRV, Asset, and Gemini. The CER comparable to the LMSC

CER for structure development has airframe weight and aircraft speed as parameters. A Mach number can be computed at which this OART CER will produce results comparable to those of the new LMSC CER. For the Configuration D design point, a Mach number of 11.8 produces comparable cost and a Mach number of 6.9 produces comparable sensitivity of cost to structure weight.

This is a reasonable confirmation of the validity of the new LMSC CER since the structural design problems of a high-speed aircraft for speeds of Mach 7 and above could be expected to be similar to those of the Space Shuttle orbiter. The slope of the OART CER (not a function of Mach number on log log paper) is depicted in Figure 4.3-1 for visual comparison.

4.3.1.2 External Tank Production Costs. Over the years, many companies and government agencies have contributed to the improvement of CERs for external tank production costs. At the time of this CER assessment, nine production bid-type cost estimates were available for tanks with slightly different but well-defined characteristics. Of these, four cost quotes were available for a single tank specification. Of the sample tanks, the stage-and-one half tanks had no deorbit system; all the other tanks had. First, the tank weight and costs were normalized to exclude the deorbit systems where applicable according to:

Cost of Deorbit System = $3 \, (10^{-6}) \;\; W$, ($\$10^6$)
Weights of Deorbit System = $0.05 \, W$, (lb)
with W being the dry weight of an individual tank in pounds.

The new CER was generated by log log least-squares curve fitting. One question arose as to what weight to attach to the group of four "identical spec" bids. A variation of this weighting factor was performed and resulted in the following CERs for droptank first unit cost (DTTFU):

$$DTTFU = A \, (W)^{B}, \; (\$10^6)$$

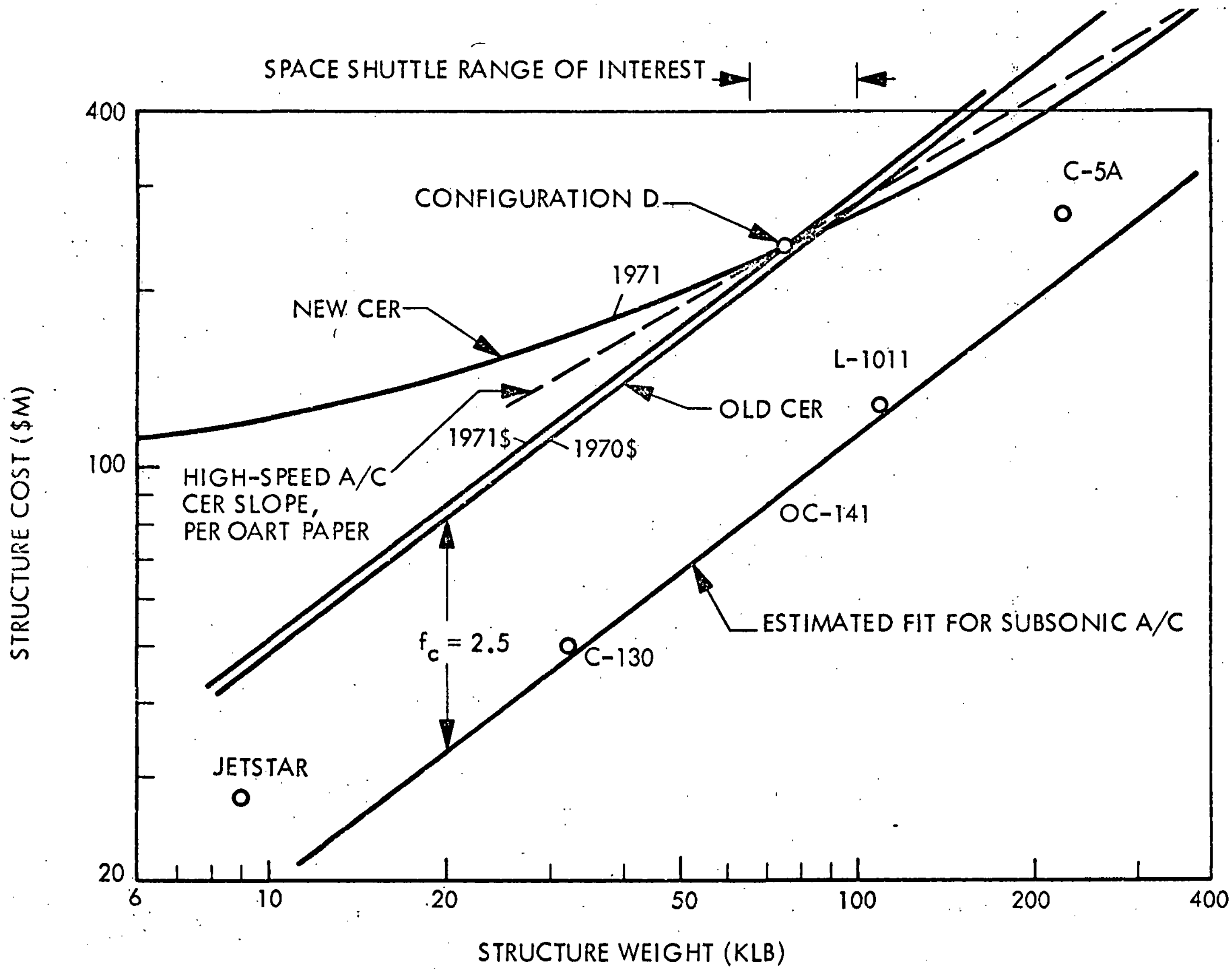

Figure 4.3-1 Comparison of Orbiter Structure Design and Engineering CERs

Selective Weighting Factor		0	1.0	1.5	2.0	
A	-	16.794 (10^{-3})	8.725 (10^{-3})	7.895 (10^{-3})	7.417 (10^{-3})	
B	-	0.501	0.572	0.583	0.590	
DTTFU	M$	4.405	5.083	5.196	5.267	@67,740 lb 1970 dollars
$\frac{dc}{dw}$	$/lb	32.6	43.0	44.7	45.9	

It was concluded that the same level of detail was included in all cost estimates and that a weighting factor of one should be applied. This decision was simplified by the fact that the cost slope does not change much if the weighting factor is changed from 1.0 to 1.5.

After adjusting to 1971 dollars, the new CER for External Tank First Unit Cost becomes:

$$DTTFU = 9.161\ (10^{-3}) \left(\frac{DTDRYW - DODTW}{TPVN}\right)^{0.572} (DTK)$$

Where DTFRYW = Drop Tank Dry Structure Weight

DODTW = Deorbit System Dry Weight

TPVN = No. of Tanks per Flight Set

DTK = Complexity Factor

This CER is plotted in Figure 4.3-2 for several complexity factors along with the nine design points used as a basis for least-squares fitting.

4.3.1.3 Solid Rocket Recurring Production and Refurbishment Costs. A booster system for the Space Shuttle is considered here to consist of a number of solid rocket motors (2 in Configurations C and D).

Each solid-rocket motor in turn includes a solid motor and the subsystems. The focal point of solid-rocket production cost estimation is the expendable solid motor. Cost data from four contractors were available * to describe the relationships between solid-motor average annual production costs, solid-motor propellant weight, and total annual propellant batch production

* Report, Alternate Concepts Study Extension, Vol. II, Concept Analysis and Definition, Part 3, SRM Boosters, LMSC-A995931, 15 Nov. 1971, (U).

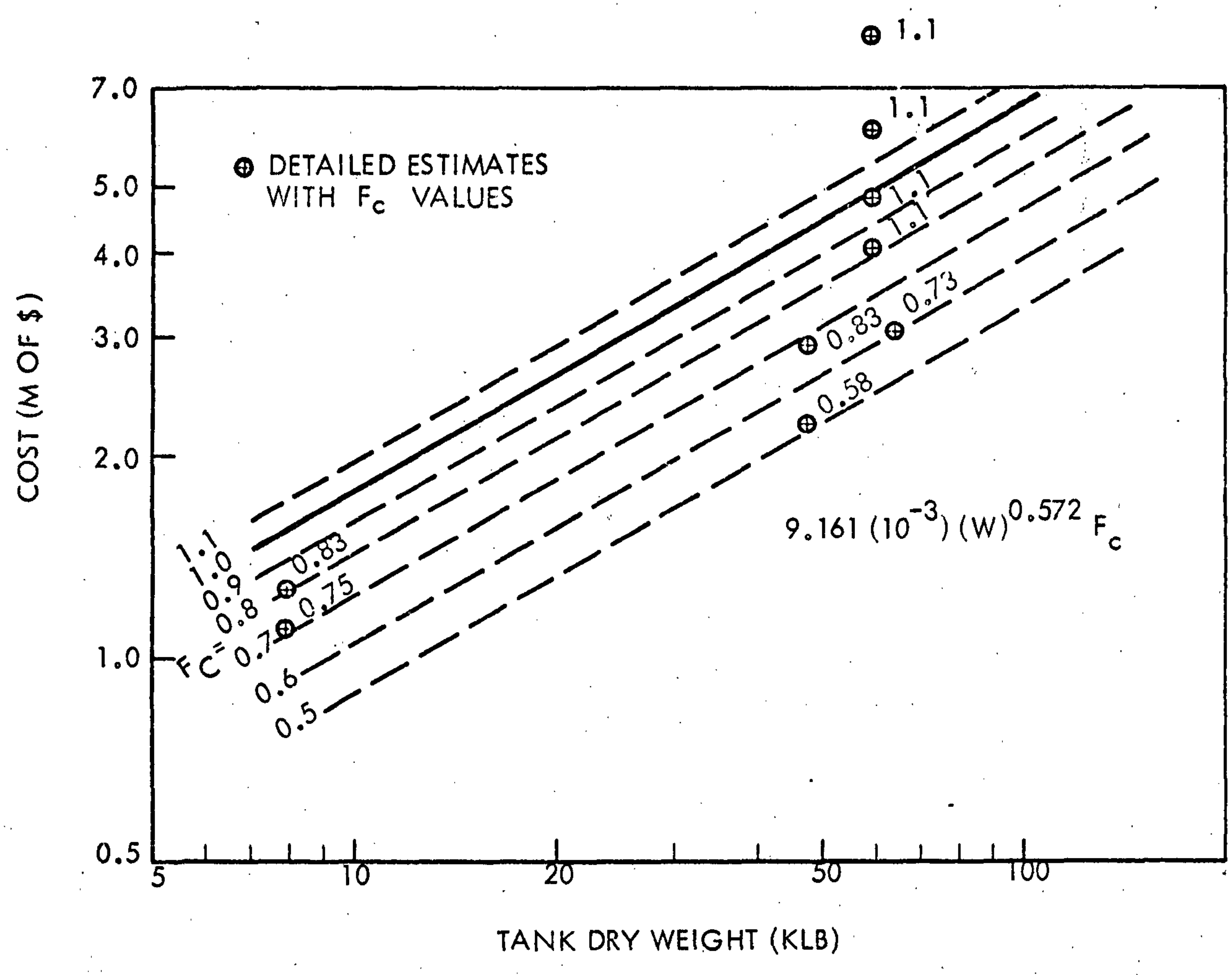

Figure 4.3-2 External Tank First Unit Cost CER

as shown in Figure 4.3-3. This presentation is based on having two solid rockets per flight, as is the case for both Configurations C and D of this study.

Between propellant weights of 1 and 1.6 million pounds, the cost data are approximated by C = A + B Wp.

The scaling factors A and B are given in Figure 4.3-4 as a function of number of flights per year. The buildup schedule for flights per year is patterned after but somewhat simplified from the schedule used by LMSC for the Space Shuttle proposal effort. The three programs of 250, 500, and 750 operational flights each start after an invariant 6 R&D flights, followed by a geometrically similar launch rate buildup as shown in Figure 4.3-5.

Solid-motor subsystem first unit costs are given by:

$$\text{Cost } (\$10^6) = 0.078\ (10^{-3})\ (W_{ss})$$

Where W_{ss} = Subsystem weight in pounds

The subsystem weight is a percentage of the Solid propellant weight (Wp) as follows:

For recoverable solid rockets, the subsystem weight includes:

TVC	0.3% Wp
Parallel Staging	1.1% Wp
Thrust Termination System	0.1% Wp
Recovery S/S Complex	0.5% Wp
Wss =	2.0% Wp

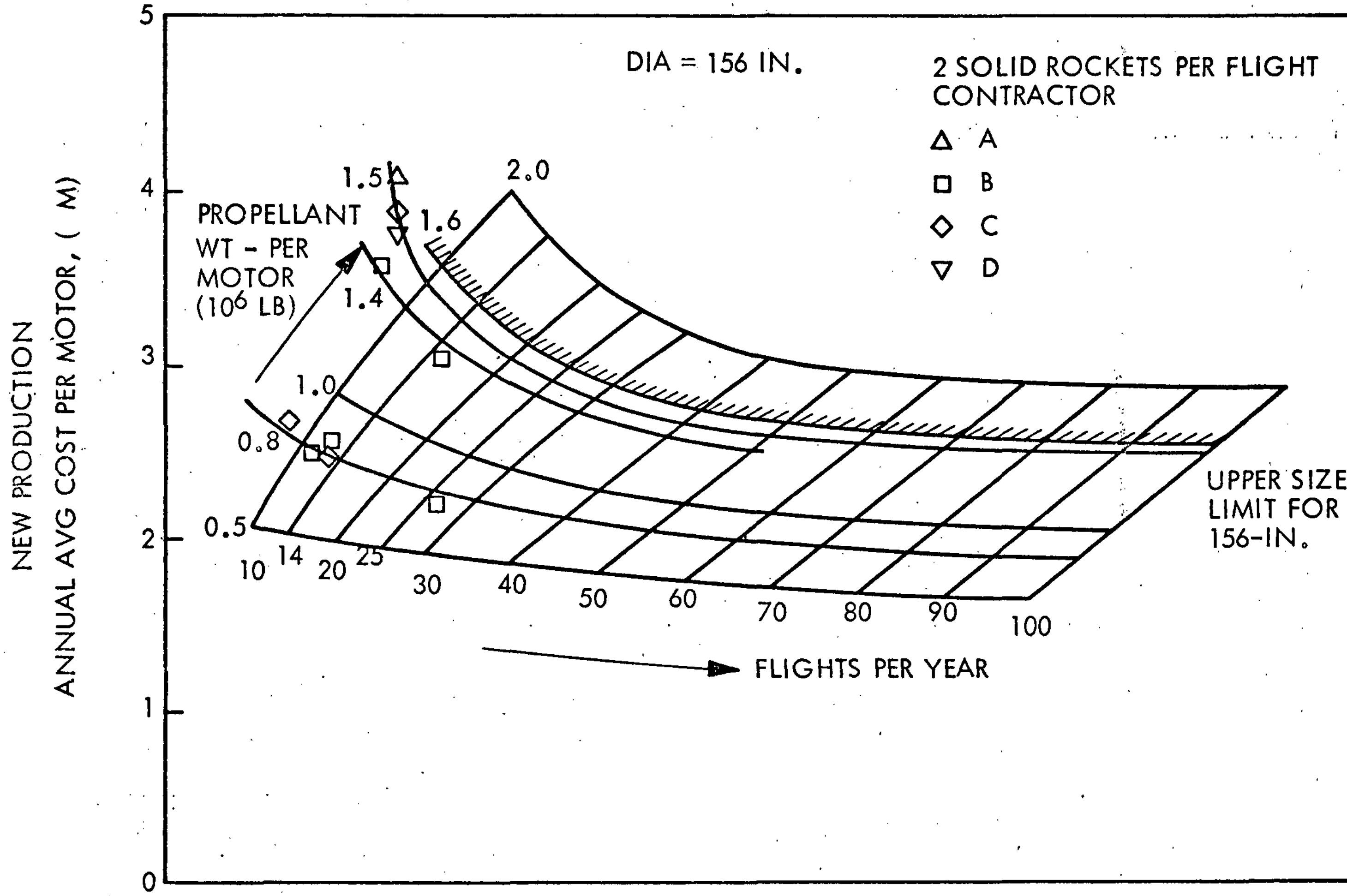

Figure 4.3-3 Production Cost of Expendable Solid Motors

$$COST = A + B\ W_p$$

$$1.0 < W_p < 1.6 \quad , \quad W_p \text{ IN } 10^6 \text{ LB}$$

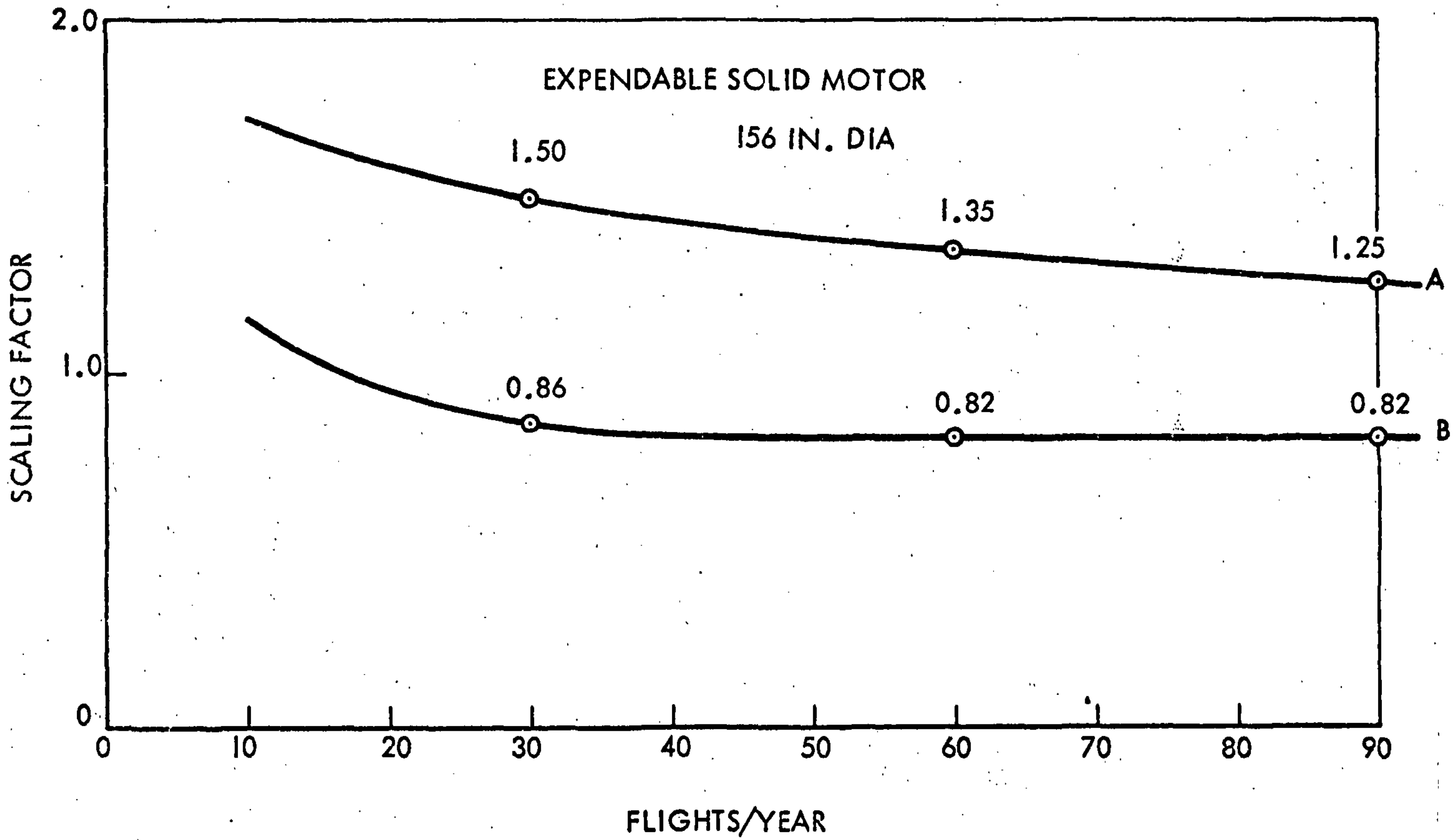

Figure 4.3-4 Scaling Factors for Average Annual Solid-Motor Production Cost

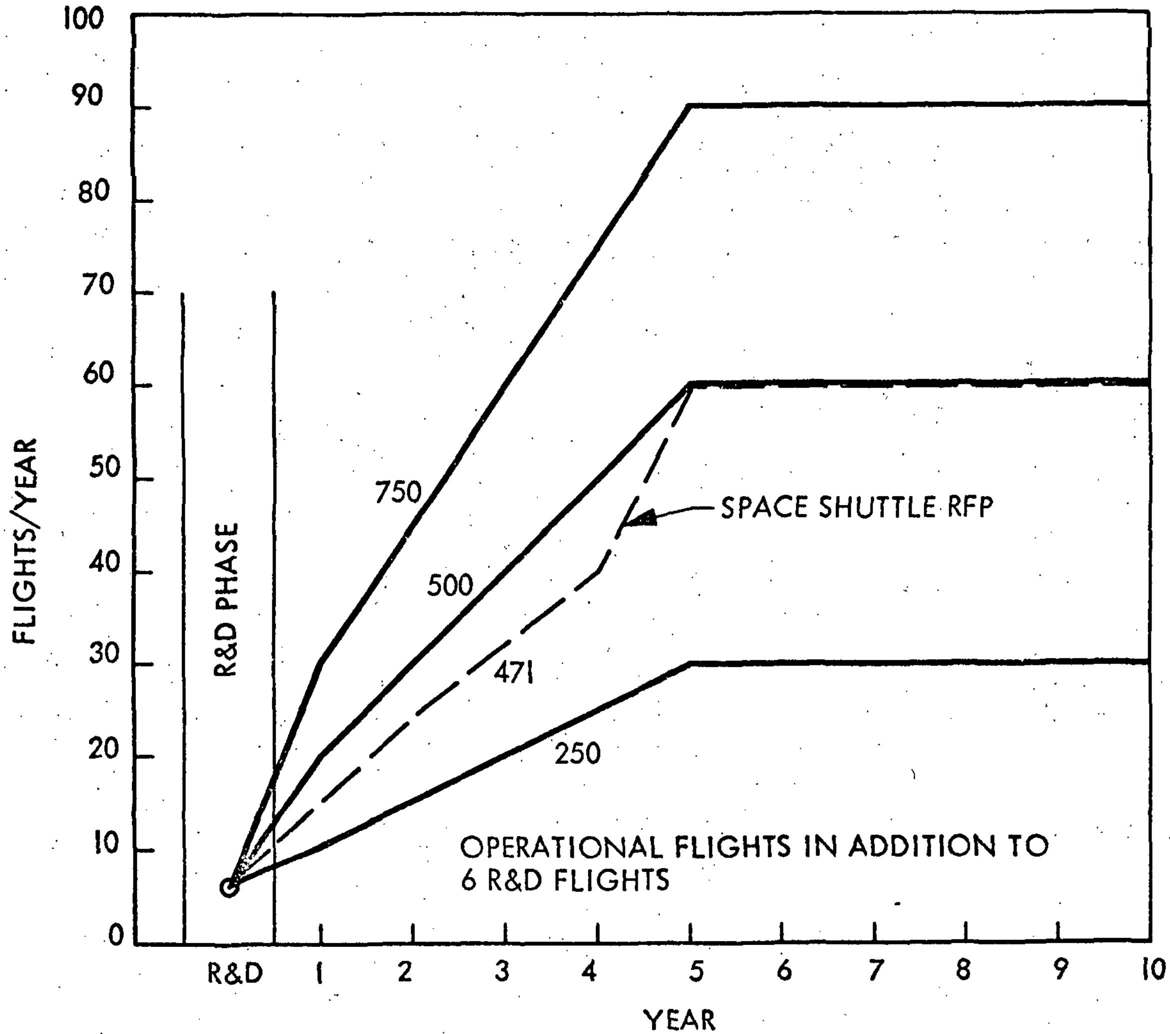

Figure 4.3-5 Mission Models Use in Analyses

Subsystems unit cost then is:

$1.56\ (W_p)\ (CA)$, $\$10^6$

Where (CA) is the cumulative average learning factor at 90 percent learning.

The combined CER for production cost per new reusable solid rocket is given by:

Cost = 1.11 (cost of expendable solid motor + cost of subsystems)

The factor 1.11 accounts for an overall beefup of subsystems and motor to provide reusability.

The production cost per new reusable solid rocket is:

$$1.11\ [A+B\ W_p + 1.56\ (W_p)\ (CA)]$$

The cost of recurring solid-rocket refurbishment has been estimated to be 55 percent of the new production cost. Since production costs have been found to be a function of total annual propellant production (both new production plus refurbishment), the factors A and B are functions of the total number of flights per year. The three flight schedules are satisfied with reusable solid rockets that are assumed to be good for $n = 10$ design uses. If the probability of recovery failure is $p=0.1$, then the effective number of uses is $n_{eff} = \frac{1}{p}\ (1 - (1-p)^n) = 6.51$ which is rounded off to 6.0. It is assumed that solid-rocket refurbishment requires a 6-month period. Previous studies showed that this was the pacing time on laying out solid-rocket production schedules.

For 500 total flights in 10 operational years, for example, the production schedule is as shown in Table 4.3-2. Varying the total number of flights by some factor can be shown not to affect the scheduling. Rather, a lateral expansion on production volume occurs. Therefore, for our three geometrically similar flight schedules, the ratio of new to total production is only a function of the calendar year rather than total production level.

TABLE 4.3-2

SOLID ROCKET PRODUCTION SCHEDULE, 500-FLIGHT PROGRAM

OPERATIONAL YEAR	1	2	3	4	5	6	7	8	9	10
SOLID ROCKETS REQU. PER YR.	40	60	80	100	120	120	120	120	120	120
CUM SOLID ROCKET REQU. PER YR.	40	100	180	280	400	520	640	760	880	1000
NEW SOLID ROCKET PROD. PER YR.	14	20	24	26	24	22	20	18	16	-
CUM NEW PROD. PER YR.	14	34	58	84	108	130	150	168	184	184
NEW/TOTAL PRODUCTION=a	0.35	0.33	0.30	0.26	0.20	0.18	0.17	0.15	0.13	0
% CUM PRODUCTION LEAD @ 6.51 FLTS/NEW	128	121	110	95	76	63	53	46	36	20

Due to scheduling effects, the average number of new production is a=0.184, which is equivalent to 5.43 effective uses per solid rocket, 16.5 percent less than theoretically available. This excess capability is in the program in terms of production lead to reduce program risk and in residual life of solid motors at the end of the program. The resulting solid-rocket recurring cost CER is therefore conservative.

The equation previously given for production cost per new reusable solid rocket can be expressed as

$$A_1 + B_1\,Wp \quad \text{with}$$

$$A_1 = 1.11A \text{ and } B_1 = 1.11\,[B + 1.56\,(C.A.)]$$

with N_i flights per year and $A_i = \dfrac{\text{New Production}}{\text{Total Production}}$ in year i, and assuming that the cost of a refurbished solid rocket is $F_R = 0.55$ (new cost), we obtain the total production cost of new solid motors and the total refurbishment costs as follows:

$$\text{Total New Production Costs} = \sum_i a_i N_i A_{1i} + \left[\sum_i a_i N_i B_{1i}\right] Wp$$

$$\text{Total Refurbishment Costs} = F_R \sum_i (1-a_i)\, N_i A_{1i} + \left[F_R \sum_i (1-a_i) N_i B_{1i}\right] Wp$$

The CERs are referred to solid booster propellant weight WPSB = 2 Wp, (2 solid rockets per booster). For reusable solids, WPSB is 0.88 of the total solid booster weight. The resulting CERs are in $\$10^6$ and based on WPSB in 10^6 lb.

Total Flts.	No. New Sets	Total New Production Costs (1)	No. Refurb Sets	Total Refurbishment Costs (2)
250	46	158.3 + 98.7 WPSB	204	380.5 + 232.4 WPSB
500	92	285.8 + 180.8 WPSB	408	685.0 + 427.9 WPSB
750	138	400.8 + 261.0 WPSB	612	956.6 + 620.3 WPSB

Total recurring solid booster cost = (1) + (2)

4.3.2 Derivation of Direct Cost Sensitivities

As shown in Paragraph 4.3.1, the significant CERs for the vehicle elements take the general form:

$$\text{Cost} = A\ (W)^B$$

Where W is a primary costing parameter or "cost driver" (usually weight) and A and B are constants.

To derive direct cost sensitivity, the CER can be simply differentiated:

$$d\ (\text{cost}) = A\ B\ (W)^{B-1} dW$$

It can be seen that to compute a total change in cost resulting from changes in several cost drivers, the contribution due to each driver with a different value of the exponent (B) must be computed separately before summing. Table 4.3-3 lists the principal vehicle elements and the sensitivity exponents from both the RDT&E and the first unit CERs applicable to Configurations A through D.

The total RDT&E cost for the orbiter includes the sum of the subsystem development costs plus a percentage of these costs for management and integration plus the cost of the development hardware. The total program RDT&E costs include the orbiter, tank, SRM, and support system development costs plus the costs for development flight testing. The production costs include the costs for building the flight hardware for the operational program plus the costs for converting development flight orbiter to operational status. Operational costs include the launch costs, orbiter refurbishment costs, tank production costs, and SRM production/recovery refurbishment costs.

While it is possible to compute direct cost sensitivities by differentiating the CERs and evaluating these expressions, a perturbation method was used in this study. After computing baseline costs, a series of perturbed cases were computed by changing one cost driver at a time and computing a new

Table 4.3-3 VEHICLE ELEMENT CER EXPONENTS

Vehicle Element	Subsystem	Primary Costing Parameter	Exponent RDT+E	Exponent 1st Unit
Orbiter	Total Structure	Structure Weight	0.762	0.716
	Main Engines	Engine Vac. Thrust	0.504	0.536
	Plumbing + Actuation	Plumbing + Actuation Weight	0.312	0.805
	Thermal Protection	Thermal Protection Weight	0.454	0.5
	Airbreathing Propulsion	Airbreather S.L. Thrust/Engine	0	0.824
	Landing Gear	Landing Gear Weight	0.766	0.766
	Orbit Maneuvg. Sys.	Parametric Cost Data	0	0
	RCS Sys.	Parametric Cost Data	0	0
External Tank	Total Structure	Structure Weight	0.312	0.572
	Deorbiting System	Deorbiting System Weight	1.0	1.0
SRMs	Solid Motor (Incl. Case)	Propellant Weight/Motor	1.0	0.717
	Subsystem	Stage Weight	1.0	1.0
	Recovery Sys.	Parametric Cost Data	0	0

set of costs. Direct cost sensitivities were then obtained by taking system cost differences between the perturbed case and the baseline and dividing by the amount of change in the driver. This approach was easily accomplished using the costing computer program; it provides assurance that all the secondary cost effects, such as those for system engineering and management, are included.

4.4 FIXED-CAPABILITY SENSITIVITIES

Fixed-capability sensitivites predict the change in weight or program cost of various elements of the system when the weight of one of the elements of the system is changed and the system is redesigned to meet all mission and design requirements. This would be the case during the preliminary or detailed design phases of the vehicle development program.

4.4.1 Weight Sensitivities

Fixed-capability weight sensitivities are derived in the following manner. First, a baseline design and ascent trajectory for the configuration is established through detailed engineering analysis. The baseline design is then perturbed by adding an input weight to one of the elements of the system, and using the parametric design relationships discussed in Section 4.2, a new design meeting all system requirements is determined. This perturbed design thus reflects the weight differences attributable to the input weight for no change in system capability or performance. The procedure is illustrated in Figure 4.4-1 for the case of Configuration D. The process is iterative in nature and has been mechanized by a computer program for each configuration.

The two most important factors in deriving fixed-capability sensitivities are the following: What performance or capability requirements should be maintained and what redesign approach should be taken to meet those requirements? For example, delivering payload to a given orbit is a given performance requirement. If weight growth occurs in Configuration D should the external tank or SRM or both be resized to meet that requirement?

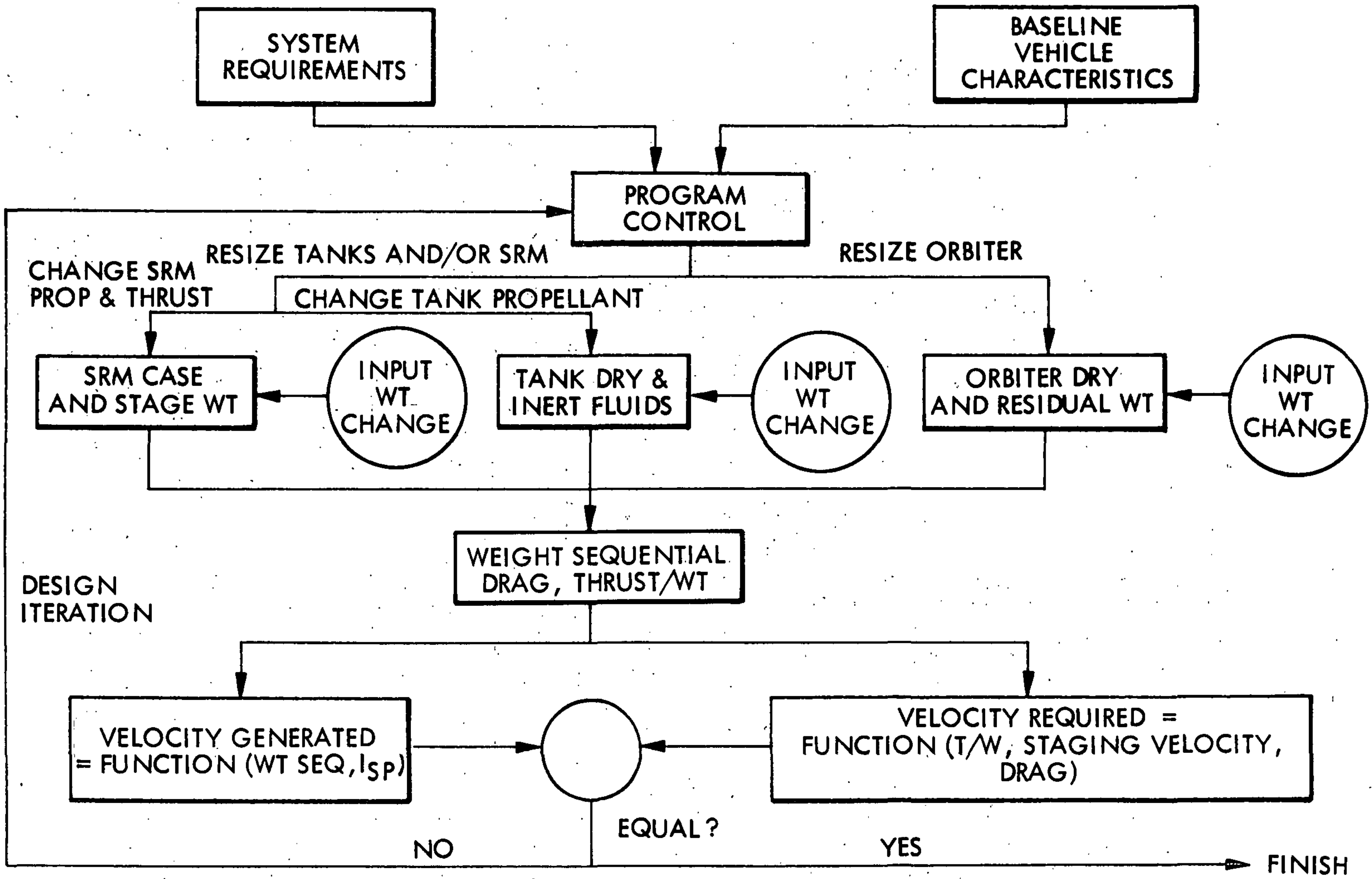

Figure 4.4-1 Vehicle Resizing Process

Different design approaches to accomodate weight changes are taken during the vehicle preliminary design phase than during the detailed design phase. During vehicle preliminary design, more design parameters are available to be changed; less vehicle elements have been frozen. Usually, the propellant loadings for all elements may be varied in any way, and both main engine and solid-rocket motor thrust may be altered (except for Configurations C and D, in which the main engine thrust is fixed at the ICD values to give more realistic values for the current Space Shuttle Configuration). During detailed design, main engine thrust levels are fixed. and only one element of the system, the one with the shortest development time, is allowed to vary. The design approaches for the five configurations are summarized in Figure 4.4-2. Note that there is considerable variation in the approach taken with different configurations. Each configuration must be analyzed with its own characteristcs in mind, making a comparison of weight sensitivities between configurations difficult to interpret.

Weight sensitivities may either increase or decrease as the design freeze progresses (see Table 5.1-1 in Section 5). This depends on the relative values of two conflicting effects: (1) early in the development program, during the preliminary design phase more flexibility exists to minimize weight changes by an optimum choice of design variables (such as propellant loads), causing lower sensitivities, and (2) later in the program, as the design definition progresses, growth margins maintained early in the program may be relaxed, allowing for a lower weight change to maintain requirements as growth occurs, which also causes low sensitivities. Depending on which of these effects dominates weight sensitivities may either increase or decrease in going from vehicle preliminary design to detailed design. Examining Figure 4.4-2, it can be seen that for Configurations C, D, and E, launch weight was minimized as weight growth occurs during preliminary design, but was not minimized during detailed design; therefore, the sensitivities during detailed design would be expected to be higher (also see Table 5.1-1). For Configurations A, and B, the launch

YES: ITEM VARIES NO: ITEM FIXED NO ENTRY: NOT APPLICABLE

		ORBITER WEIGHT	EXTERNAL TANK SIZE	DROPTANK SIZE	MAIN THRUST ENGINE	SRM SIZE THRUST	FLYBACK BOOSTER SIZE
PRELIMINARY DESIGN							
CONFIGURATION	A	YES		YES	YES[3]		
	B	YES	YES[1]	YES[1]	YES[3]		
	C	YES	YES[2]		NO	YES[2,3]	
	D	YES	YES[2]		NO	YES[2,3]	
	E	YES	YES[2]		YES[2,3]		YES
DETAILED DESIGN							
CONFIGURATION	A	YES		YES	NO		
	B	YES	YES	NO	NO		
	C	YES	NO		NO	YES[3]	
	D	YES	NO		NO	YES[3]	
	E	YES	YES		NO		NO

(1): External tank and droptank varied with their lengths fixed and only the diameter (common to external tank and droptank) varied

(2): Launch weight (Glow) minimized by choosing optimum propellant loadings in tank and booster

(3): Launch thrust/weight ratio maintained at baseline value

Figure 4.4-2 Vehicle Design Approach to Accommodate Weight Growth thrust-to-weight ratio was maintained at its baseline value for weight growth during preliminary design so that further weight growth might be accomodated during detailed design. During detailed design, the thrust was fixed and the thrust/weight ratio was reduced as weight growth occurs. Since both these sensitivities have minimum launch weights at thrust/weight ratios lower than their baseline values, this effect reduced the sensitivities considerably during the detailed design phase. In effect, a weight growth margin was utilized to reduce a weight sensitivity. This is true in general: a system which is initially overdesigned for its requirements will have a lower sensitivity to weight growth later. Another example of this is the case in which Configuration C or D is initially designed at a higher than optimum staging velocity (larger SRM and smaller tank than for the minimum launch weight system). If this is done, and only the tank size is changed when weight growth occurs, the system will show a lower sensitivity than if it were initially designed at the optimum staging velocity, because as the tank size is increased the staging velocity is reduced, moving toward the optimum value. This effect is discussed further in Section 4.6.3.

4.4.2 Cost Sensitivities

Fixed capability cost sensitivites are determined by calculating the cost of the weight increases of the elements and subsystems occurring during the generation of fixed-capability weight sensitivities. These weight increases are costed by utilizing the direct cost sensitivities discussed in Section 4.3, which are based on cost estimating relationships. The incremental cost of each weight (or thrust) increase is calculated, and the total summed to give the system cost. The input weight may be considered "free", with no cost associated with it (its cost would be determined by separate analysis), or it may be considered as structure weight, with its associated cost. The former type of cost sensitivity is used for design weight/cost tradeoff studies. The procedure described above is illustrated in Figure 4.4-3, where an example is given.

$$\underbrace{\frac{\Delta(\text{COST})}{\Delta(\text{INPUT WT})}^{\wedge}}_{\substack{\text{FIXED PERFORMANCE}\\ \text{COST SENSITIVITY}}} = \underbrace{\sum_{\text{ELEMENTS}}}_{\substack{\text{(ORBITER, TANK}\\ \text{AND SRMs)}}} \cdot \underbrace{\frac{\partial(\text{COST})}{\partial(\text{ELEMENT WT})}}_{\substack{\text{DIRECT COST}\\ \text{SENSITIVITY}}} \cdot \underbrace{\frac{\partial(\text{ELEMENT WT})}{\partial(\text{INPUT WT})}}_{\substack{\text{FIXED CAPABILITY}\\ \text{WEIGHT SENSITIVITY}}} + \underbrace{\frac{\partial(\text{COST})}{\partial(\text{INPUT WT})}}_{\substack{\text{ADDITIONAL DIRECT}\\ \text{COST IF INPUT WEIGHT}\\ \text{NOT "FREE"}}}$$

EXAMPLE: CONFIGURATION D, DETAILED DESIGN RECURRING COST PER FLIGHT

$$\left.\frac{\Delta(\text{RECURRING COST})}{\Delta(\text{ORBITER INERT})}\right|_{\substack{\text{TANK}\\ \text{FIXED}}} = \frac{\partial(\text{REC COST})}{\partial(\text{SRM PROP})} \cdot \frac{\partial(\text{SRM PROP})}{\partial(\text{ORB INERT})} + \frac{\partial(\text{REC COST})}{\partial(\text{SRM DRY})} = \frac{\partial(\text{SRM DRY})}{\partial(\text{ORB INERT})} + \frac{\partial(\text{REC COST})}{\partial(\text{ORB DRY})} \cdot \frac{\partial(\text{ORB DRY})}{\partial(\text{ORB INERT})}$$

$$= 0.38\ (48.1) + 8.60\ (5.47) + 0.85(0.45)$$

$$= \$65/\text{lb PER FLIGHT}$$

Figure 4.4-3 Derivation of Fixed Capability Cost Sensitivities

4.5 FIXED VEHICLE SENSITIVITIES

During the early phases of the Space Shuttle development program, the sizing of the various vehicle elements must be frozen. The last element size freeze may be less than 18 months into a 6-year development program. As design changes occur during the remainder of development, weight growth and reduction of contingencies can be expected. If remaining contingencies approach zero, either weight reduction design changes are introduced, usually at considerable cost to the program, or planned payload capability is reduced.

Estimating the cost of weight reduction program is best done on the basis of specific design change possibilities. The value of such programs can be estimated, however, by considering the alternative of accepting reduced payload and evaluating fixed vehicle cost sensitivities. These are measures of program cost increases which would result from reduction of payload capability. A fixed vehicle cost sensitivity can be expressed as follows, PCL being payload capability loss.

$$\frac{\Delta\ \text{Cost}}{\Delta\ \text{Input Weight}} = \frac{\Delta\ \text{Cost}}{\text{PCL}} \times \frac{\text{PCL}}{\Delta\ \text{Input Weight}}$$

The former factor is a direct cost sensitivity and the latter is a fixed vehicle performance sensitivity.

4.5.1 Fixed Vehicle Performance Sensitivities

The ascent payload loss resulting from changes in the weight of a system element is computed by reference to ascent and reentry simulations which assume the same vehicle, except for the addition of the input dry weight. The changes made in the operation of the vehicle allow for addition of propellant up to tank capacities (OMPS and RCS tanks were offloaded in the baseline design on those missions which established vehicle sizing).

Weight increases without redesign also often imply relaxation of secondary requirements such as landing speed and structural design margins. In any case, no redesign penalties are assumed in this analysis to maintain these design values constant.

The primary driver of fixed vehicle performance sensitivities is the point in the mission (ideal velocity) at which the input weight is staged relative to that at which payload is delivered. If orbiter dry weight is increased, and the return payload requirement is maintained, propellants must be added to maneuver and retro fire after payload delivery and to control the greater inertia during reentry. The ratio of ascent payload capability loss to input weight is greater than unity in this case. On the other hand, it can be argued that a decrease in ascent payload could be accompanied by an equivalent decrease in the return payload requirement. This is likely since most missions which are return-payload critical return the complete ascent payload. In this case (to be assumed here), PCL/Input Weight equals unity. If the input weight is in the external tank or booster, PCL/Input Weight is always less than unity. More refined assumptions, using different performance sensitivities for different missions, will be appropriate when realistic mission models become available.

4.5.2 Cost Sensitivity to Payload Capability Loss

If payload capability of the Shuttle is reduced below the planned level, the program cost increase to accomplish a given set of missions is greater than if the same potential loss had occured earlier, when it was still possible to resize the vehicle to retain the planned payload capability. An analysis summarized in this section provides an estimate (on a conservative basis) of $85,000 of total program costs per lb of payload loss (per flight) on a 500-flight program. The largest corresponding value for resizing is that for Configuration D and is $36,900/lb.

An extensive economic study of missions which require a tug (or kick stage) has been done by LMSC under contract to NASA (Marshall Space Flight Center contract NAS 8-27709, "Space Tug Economic Analysis", completed in June 1972). Since tug flights constitute about 70 percent of all Shuttle flights in a typical mission model, it was appropriate to use the capabilities developed in the tug study as the principal basis for estimating the cost of payload capability loss.

A computer program for the tug study sought the minimum cost approach to meeting a given set of mission requirements. It employs a reusable tug and either reusable or expendable payloads in various modes (including expending the tug when appropriate to get sufficient benefit from larger payloads) and chooses the least costly mode for each mission. The particular choice for each mission depends on the Shuttle payload capability and tug size. The program was designed primarily to seek the optimum tug design but was found quite adaptable to the current problem.

One reservation in this application concerns an assumption used in the tug study that only one mission type is considered in each Shuttle flight. The introduction of multiple-mission flights would increase the load factor from the 70 to 80 percent typically found in tug study results. The cost effects of payload capability loss would be greater at greater load factors and sensitivity results from the tug study program can be considered conservatively low.

Figure 4.5-1 summarizes the results of 5 runs of the tug economic study computer program. All cases meet the same mission requirements of 422 payload placements for 57 space programs. The reference case assumes 65 Klb Shuttle capability on an east mission and an H_2/O_2 tug with 50 Klb propellant capacity which has been found optimum for the 65 Klb Shuttle case. When the Shuttle payload is decreased, the mission modes shift toward less reusable and more expendable payloads and more cases of expending the tug. Also, the tug is flown offloaded in more cases and additional Shuttle flights are sometimes used for completion of

57 Missions, 422 Placements, $10M Shuttle Users Cost

Case		Ref	Same Tug	Resized Tug		
Shuttle Payload Capacity (East)		65K	45K	45K	50K	55K
Tug, Payload Cap Loss		0	20K	20K	15K	10K
Tug, Propellant Capacity		50K	50K	36K	36K	36K
No. of Shuttle Flights		498	597	523	449	447
Shuttle Load Factor (Overall)		0.698	0.785	0.847	0.776	0.698
Costs ($M)	Shuttle	4,980	5,970	5,230	4,490	4,470
	Tug	1,265	2,251	1,762	1,550	1,549
	Payloads	14,177	15,290	15,295	15,332	15,190
	Total	20,422	23,511	22,287	21,372	21,209
ΔTotal Cost ($M)		0	3,089	1,865	950	787
ΔCost/Payload Loss ($/Lb)		0	154.5K	93.2K	63.3K	78.7K
Average ($/Lb/Flight) (Per Ref Case Shuttle Flight)		0	$310/Lb		$127/Lb	

Fig. 4.5-1 Cost Effects of Payload Capability Loss – Tug Missions

fueling. In some cases, payloads are assumed to be redesigned to lighter weight configurations using more costly technology.

If a significant payload capability loss occurs and the tug is not resized, the penalty could be $154,500 per lb for this mission model (with 498 Shuttle flights in the reference case) or $310 per lb per flight. If, on the other hand, the amount of payload loss is accurately predicted and an optimum-sized tug developed for the new Shuttle capability (36 Klb tug is near-optimum for Shuttle with 50 Klb east capability), the program cost penalty can be reduced to $63,300 per lb, or $127 per lb per flight.

To estimate the cost effects on a mixed mission model, reference was made to a payload listing dated August 1971 and supplied to Phase B extension contractors in December 1971 (as attachment to Technical Directive L-2, "Payload Impact Analysis on Orbiter Subsystems"). This list totaled 695 missions (placements and revisits) and can be broken down as shown in Fig. 4.5-2. An estimated 700 Shuttle flights may be required for the 695 missions distributed as shown. This might be 488 tug flights, greater than the 413 placements by the same ratio as in the reference case in Figure 4.5-1. On the direct placement missions, those with a ratio of payload to capability near unity would of course require one flight per placement. Those with payloads less than 1/5 capability (many are revisits with only a few hundred lb) can either be flown on multiple missions or carried piggy-back on other flights (e.g., on tug flights, the load factor is only about 70 per cent). Perhaps 24 new multiple flights would be needed for the 94 light payloads; at an average of 3 missions per flight, 72 would be handled, leaving 22 to be flown piggy back, about 2 per year.

A rationale for estimating Δ Cost/PCL for all mission types except the last column of Figure 4.5-2 has been developed. As shown, proportioning the previous 700 flights down to a 500-flight program leaves only 5 flights in this last column. Tug mission cost sensitivity (the vast majority of the

	Tug Missions	Direct Placement Missions				Total
		$W_{PL} \div$ P/L Cap $\rightarrow 1$		< 0.2	Other	
		Sta. Supply	Specific			
No. of Missions in Reference Model	413	65	116	94	7	695
Estimated Shuttle Flights	488	65	116	24	7	700
Shuttle Flts – 500 Flt Prog.	349	46	83	17	5	500
Sensitivities to P/L Cap*						
o Flt Cost/PCL ($/lb/flt)	125	250	350	0	200	
o Prog Cost/PCL ($/lb)	43,600	11,500	29,000	0	1,000	$\boxed{\$85,100/Lb}$

*PCL = Payload Capability Loss

Figure 4.5-2 Cost Effects of Payload Capability Loss
500-Flt Program

500) has already been discussed. The \$250/lb for station supply assumes: (1) the airbreathing engine system is not used on the majority of these flights so that the reference payload is about 40,000 lb, (2) the same total payload weight must be delivered after loss in capability, and (3) the Shuttle cost is \$10 million per flight. Thus, the cost per lb per flight is \$10M divided by 40,000 lb or \$250/lb. If a small PCL occurs, additional flights would be added at this penalty. A much larger penalty (about \$500/lb) occurs if airbreathing engines are assumed.

Many of the specific-purpose missions which use most of the shuttle capability are flown at 200 to 300 mile altitude at 55 deg to 75 deg inclination and weigh 25,000 to 30,000 lb.

A loss in payload capability would imply using more costly technology to reduce weight in these payloads. Such cost effects vary widely and depend on the character of the baseline design case. If the payload baseline design is driven by cost so as to minimize the sum of transportation and payload costs, an estimate can be made. The users fee schedule might be expected to have a derivative with respect to payload weight which is approximately the ratio of Shuttle flight cost to payload capability on the mission in question; \$10M divided by 30,000 is about \$350/lb. The user would reduce his total cost to a minimum if his baseline design point were chosen so that the slope of the payload total cost is the negative of the slope of his transportation fee or $\Delta \$PL / \Delta WPL = - \$350/lb$. If a weight-saving redesign is needed as a change from this baseline, the user would need to make the most cost-effective changes at a rate slightly greater than this value. No penalty is assumed for the 17 multiple-mission flights with small payload, since they are likely to be volume limited or opportunity limited and do not use full capability. The 5 others are assigned an average value of \$200/lb per flight.

Multiplying the number of flights of each mission type by the cost per lb per flight and summing gives $85,000/lb for the total program cost effect of payload capability loss on a 500-flight program. Because of the conservative assumptions throughout this analysis, this value may be considered a lower limit. An upper limit may be estimated by assuming that the same total payload weight is to be delivered by additional flights. If the average payload is 40,000 lb the cost is about $250/lb/flt or $125,000/lb for 500 flights.

4.6 FACTORS AFFECTING WEIGHT SENSITIVITY VALUES

The value of fixed-capability weight sensitivities is affect d by various factors. These factors can make major differences in the values of sensitivites (a factor of 2 or more). To understand sensitivities it is necessary to understand how these factors can affect sensitivity values.

In general, the more stages a launch vehicle has, the less sensitive it is to weight growth. A single-stage system is much more sensitive to weight growth of its single stage than a three-stage system is to its final stage. Systems with high specific impulses have lower sensitivity to weight growth than systems with low specific impulse. Vehicles with stages which have a high structural efficiency (low structure weight/propellant ratio) have a lower sensitivity than those with low structural efficiency. Space Shuttle vehicles which take all of their main thrust engines to orbit (such as Configurations A and B) show a higher sensitivity than if some of the engines were dropped earlier in the trajectory. The ratio of engine weight to thrust has a very strong effect on such sensitivities. The configurations in this study have many of these characteristics, some tending to increase their sensitivities and some tending to decrease them. Configuration A has a high specific impulse in both stages and a high structural efficiency in the first stage (droptanks) but a low structural efficiency in the second stage and carries its engines to orbit. The former characteristics giving low sensitivities are offset by the latter characteristics. Configurations C and D have a low specific impulse and a relatively low structural efficiency in the first stage (solid-rocket motors) but a high specific impulse and high structural efficiency in the second stage (external tank).

Other factors which affect the sensitivity values for a given configuration are discussed in the following sections. These are categorized as follows: requirements, design approach, weight-scaling relationships, and baseline design selection.

4.6.1 Requirements

Performance and design requirements are major drivers for fixed-capability sensitivities. If there are no requirements, all sensitivities are zero as weight growth occurs. The major performance and design requirements, and their effect on the vehicle elements and subsystems, is shown in Figure 4.6-1 for Configuration D. A breakdown of orbiter indirect weight changes by requirement for Configuration D is shown in Figure 4.6-2. If any of these requirements are relaxed, the orbiter weight sensitivity (and hence the system gross weight sensitivity) will be reduced.

4.6.2 Design Approach

The design approach used in generating sensitivities is essentially the process of deciding which vehicle parameters will be varied and which will be held constant. These decisions will affect the value of the sensitivities somewhat. The only difference between preliminary design weight sensitivities and detailed design weight sensitivities is in the design approach. The requirements in both cases are identical. The design approach is dictated by which items of the vehicle are available to be changed (no design freeze) and what growth margins are desirable to maintain.

The major items to be considered in the design approach are propellant load and thrust of the major elements. Configuration D may be used as an example. Figure 4.6-3 demonstrates various options of resizing the system to maintain payload. Either the external tank or the solid-rocket motor or both may be increased as weight growth occurs. Also, either the main engine or solid-rocket motor thrust may be changed (or both or neither). These choices are shown schematically in Figure 4.6-3.

During preliminary design, both the tank and SRM sizes are varied to minimize the launch weight increase when weight growth occurs. The main engine thrust is fixed, but the SRM thrust varies to maintain the launch thrust/weight ratio.

WEIGHT CHANGE OCCURS IN ORBITER

SYSTEM REQUIREMENT	DESIGN CHANGE TO MAINTAIN REQUIREMENT
PAYLOAD (FOR GIVEN ORBIT)	NOT ALL REQUIRED: SRM SIZE; SRM THRUST LEVEL; TANK SIZE; ORBITER MAIN ENGINE THRUST LEVEL
OMS VELOCITY	ORBITER OMS TANKS
CROSSRANGE	THERMAL PROTECTION SYSTEM
ORBITER LANDING SPEED	ORBITER WING SIZE
STRUCTURAL INTEGRITY	ORBITER FUSELAGE, WING, LANDING GEAR, THRUST STRUCTURE
ORBITER ATTITUDE CONTROL RATES	RCS THRUSTERS, TANK SIZE
AERO ATTITUDE CONTROL RATES	CONTROL SURFACE ACTUATORS, APU, HYDRAULIC SYSTEM
AERODYNAMIC STABILITY AND HANDLING QUALITIES	WING AND AEROSURFACE SIZE AND LOCATION
ABORT	ABORT SOLID ROCKET MOTOR SIZE

Figure 4.6-1 Fixed Capability Requirements and Resulting Design Changes
Configuration D - Delta Wing-Orbiter

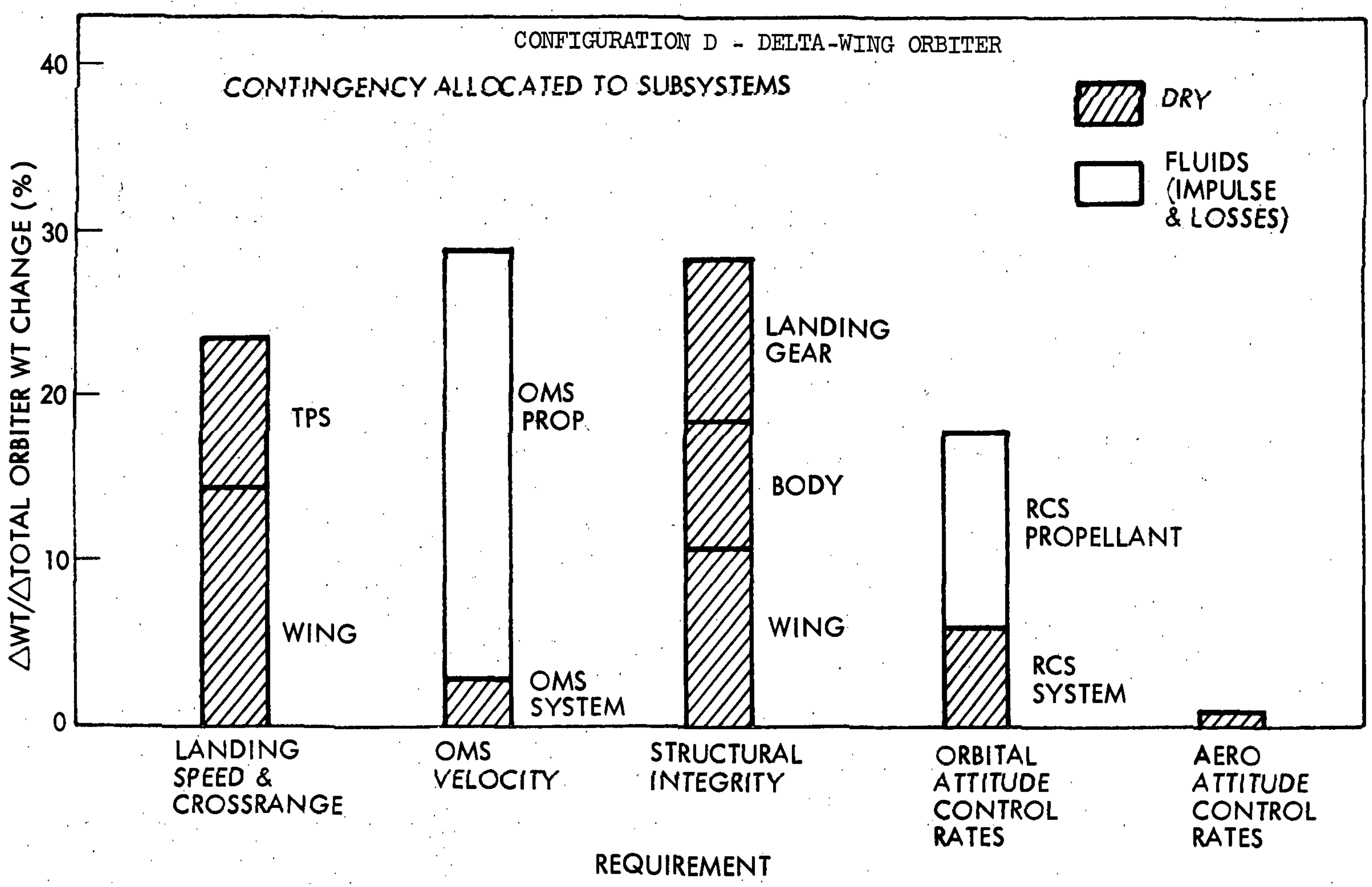

Figure 4.6-2 Orbiter Weight Changes by Requirement (Input Orbiter Weight Change)

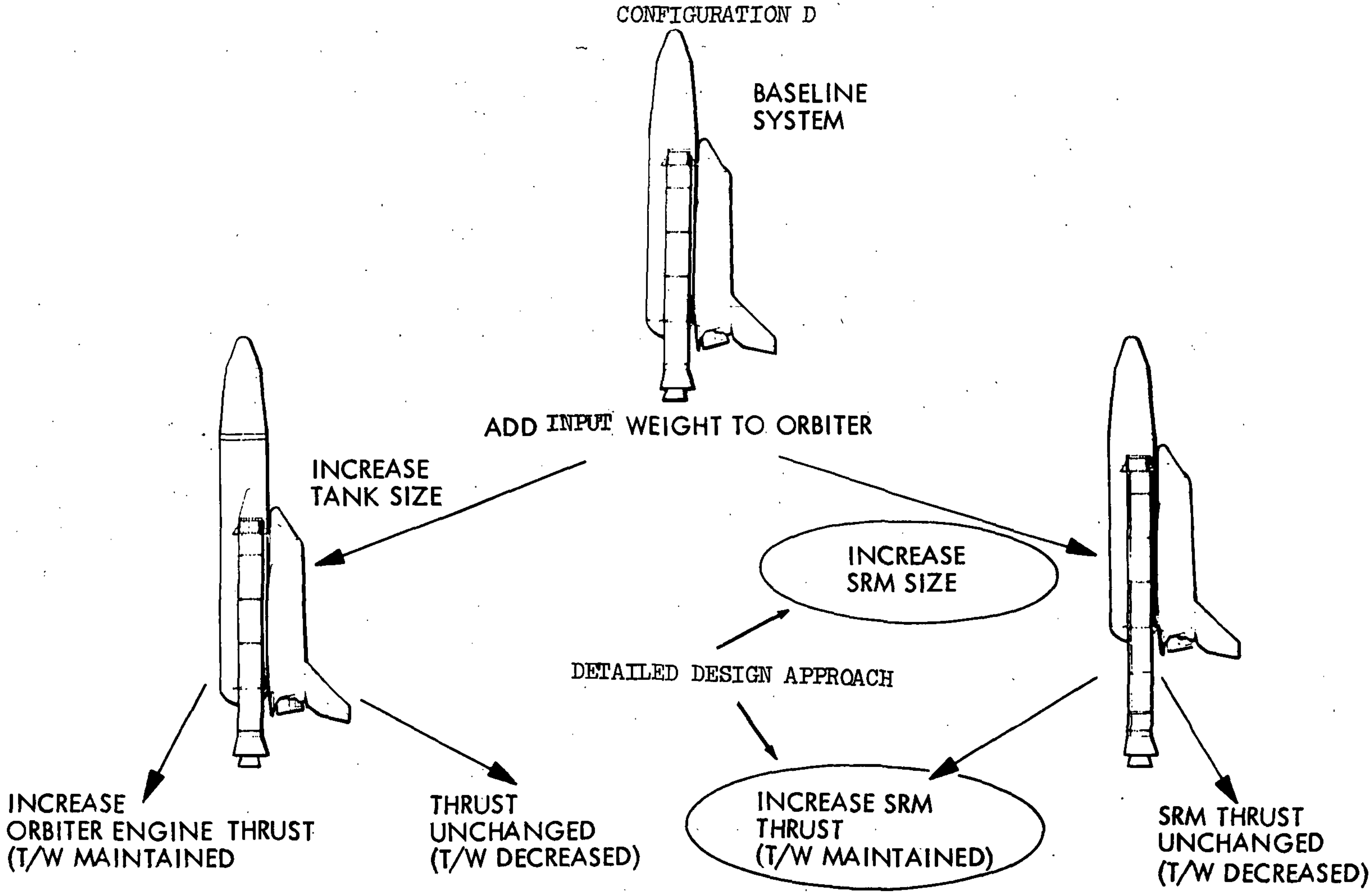

Figure 4.6-3 Optional Modes of Resizing System to Maintain Payload

During detailed design, the external tank size is frozen and only the SRM propellant and thrust is varied. An alternate approach (effects discussed in Section 4.6.3) might be to freeze the SRM propellant and thrust during detailed design and change only the tank size. This latter procedure is not used, since it provides only a small amount of additional weight growth (since the main engine thrust is fixed as tank propellant is added the staging thrust/weight is reduced, causing additional gravity velocity losses which nearly offset the additional ideal velocity gained from the added tank propellant). The three methods are shown in Figure 4.6-4, where gross weight as a function of staging velocity is shown for two cases: the baseline system and a perturbed system, where an input weight of 1800 lb is added to the orbiter. Note that the baseline system is chosen at the minimum launch weight design point. When the SRMs only are changed (fixed tank), the staging velocity increases; when the tank only is changed (fixed SRM), the staging velocity decreases. The minimum launch weight sensitivity is obtained when both the SRMs and external tanks are changed, and staging velocity is approximately unchanged. The fixed capability weight sensitivities for these three cases are shown in Figure 4.6-5.

4.6.3 Baseline Vehicle Selection

Weight sensitivities are also a strong function of the baseline selection. As an example, for Configuration D, if a system with a different staging velocity (different sizes of SRMs and external tank), were chosen as the baseline system, substantially different sensitivity values would be obtained, especially for the case in which the SRM size is fixed. The reason for this trend is illustrated in Figures 4.6-6 and 4.6-7. As the staging velocity of the baseline vehicle is increased, the sensitivity of gross weight to orbiter input weight is reduced. For sufficiently low staging velocities, this sensitivity approaches infinity; as propellant is added to the tank, payload capability is actually reduced because the increased velocity losses resulting from a lower thrust/weight ratio are greater than the increased ideal velocity supplied by the additional propellant. This effect can be shown in a slightly different manner in Figure 4.6-8.

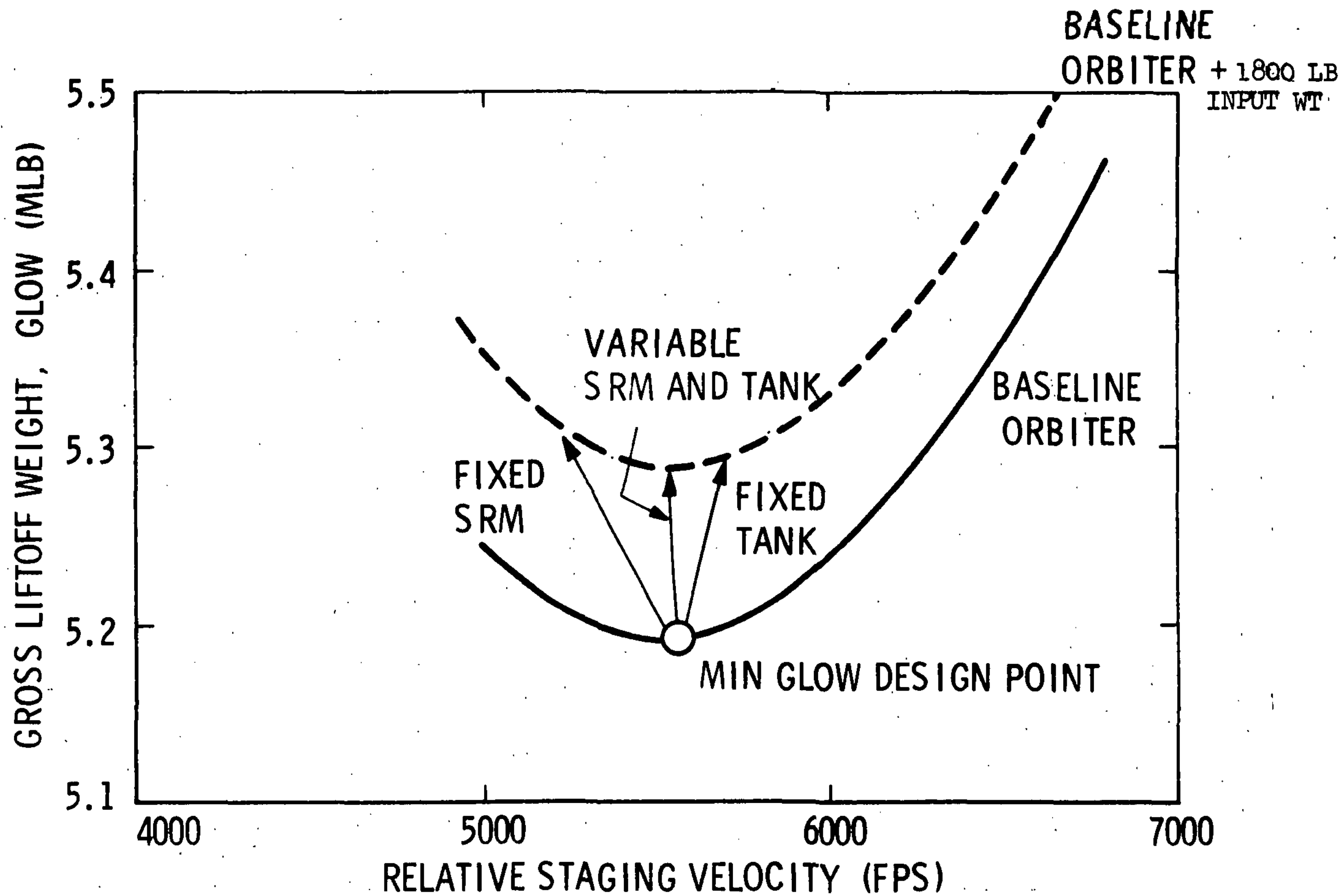

Figure 4.6-4 Fixed Capability Sensitivities Vary With Redesign Approach

INPUT ORBITER WEIGHT

ALL SYSTEM REQUIREMENTS MET

POLAR MISSION

ORBITER ENGINE FIXED

MINIMUM GLOW DESIGN POINT

WEIGHT CHANGES	SRMs VARIABLE, TANK FIXED		TANK VARIABLE, SRMs FIXED		SRMs AND TANK VARIABLE (MINIMUM) GLOW CHANGE	
Δ INPUT (ORBITER) (LB)		1.00		1.00		1.00
Δ ORBITER DRY	0.45		0.45		0.45	
Δ ORBITER OMS, ACS PROP	0.18		0.18		0.18	
Δ ORBITER GROSS (INDIRECT)		0.63		0.63		0.63
Δ TANK DRY	0		2.7		.3	
Δ TANK PROPELLANT	0		63.5		6.4	
Δ TANK GROSS		0		66.2		6.7
Δ SRM DRY	5.5		0		5.0	
Δ SRM PROPELLANT	48.1		0		41.2	
Δ SRM GROSS		53.6		0		46.2
Δ SYSTEM GROSS (GLOW)		55.2		67.9		54.5

Figure 4.6-5 Fixed Capability Weight Sensitivity Breakdown

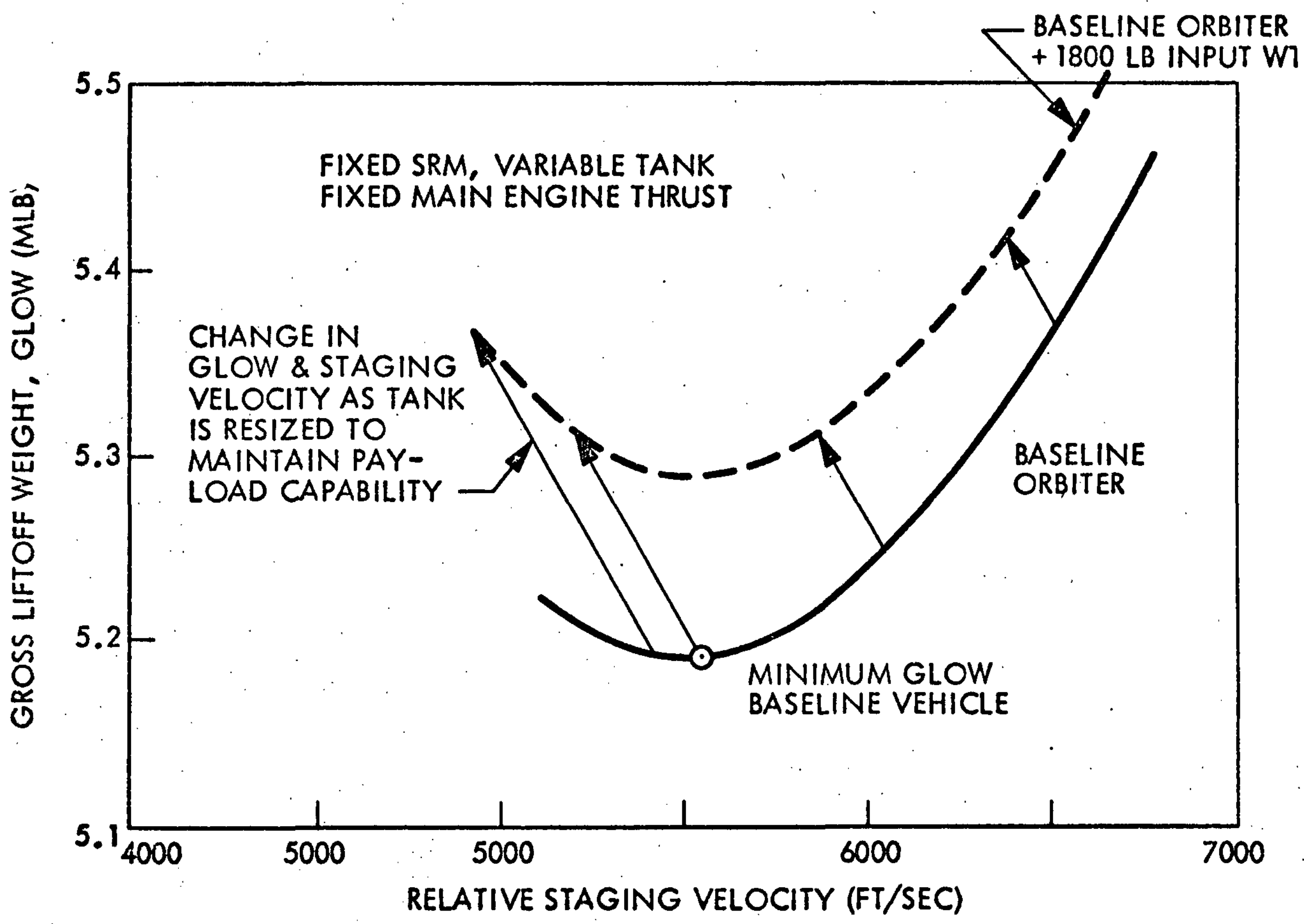

Figure 4.6-6 Fixed-Capability Sensitivities Vary With Selection of Point Design

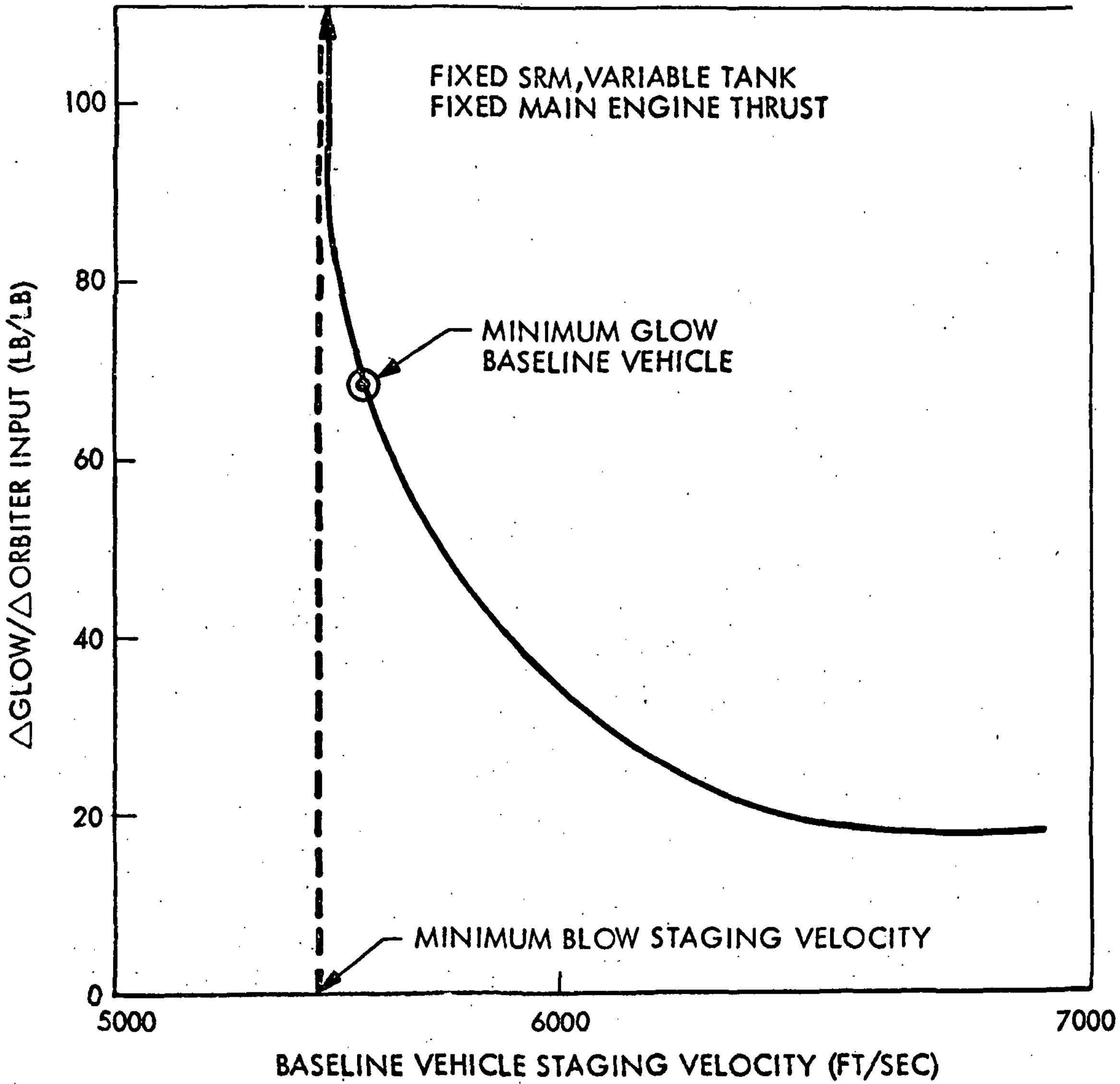

Figure 4.6-7 Fixed-Capability Sensitivity Variation With Baseline Vehicle Staging Velocity

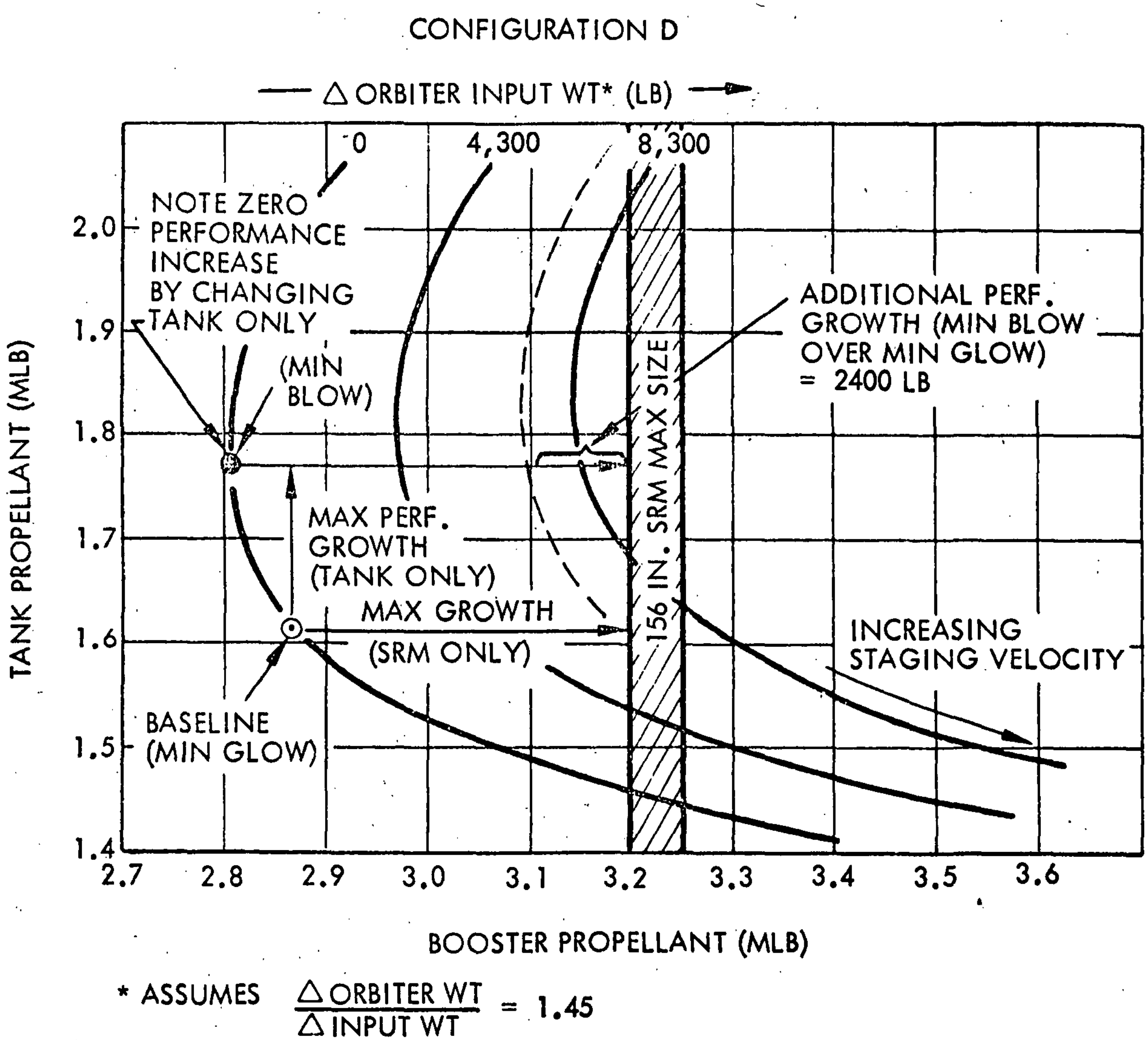

Figure 4.6-8 System Performance for Various SRM and Tank Sizes

4.6.4 Weight-Scaling Relationships

As the ratio of inert weight increase to propellant weight increase ($\Delta W_{IN}/\Delta W_P$) increases for an element in the system, fixed capability weight sensitivities also increase, assuming that that element is resized when the sensitivity is generated. As an example of this effect, consider Configuration D when only the external tank is resized to maintain payload capability. Figure 4.6-9 shows the variation of the launch weight-orbiter input weight sensitivity as the weight-scaling law of the external tank is changed. As the structural efficiency of the tank is reduced, the sensitivity increases rapidly. At a certain structural efficiency, this sensitivity will approach infinity because the additional tank weight carried to orbit decreases the ideal velocity supplied more than the increased propellant increases the velocity. Similar effects could be shown for the SRMs or for elements of the other configurations.

4.7 ANALYTICAL DERIVATION

Fixed capability weight sensitivities can be derived analytically, but the derivations are long and complex because of the large number of variables that change. Some sensitivities, such as for minimum launch weight, are virtually impossible to express analytically. The analytical procedure is to differentiate the basic performance relation (Equation (1) in Section 4.2) and to substitute in the derivatives of inert weight, ideal velocity, and specific impulse with respect to propellant load, thrust, and dependent functions of these variables (such as burn time for the SRMs, staging velocity for the heat-sink booster, etc.). Thus, very complicated equations will result.

As a very elementary example of how this might be done, this process has been carried out for a single-stage vehicle with a constant ideal velocity requirement (which is unrealistic, as has been shown in Section 4.22-1). Furthermore, specific impulse is assumed to be constant, the thrust effect on weight is not considered, and a linear scaling law of the stage inert weight with propellant is assumed. With this assumptions,

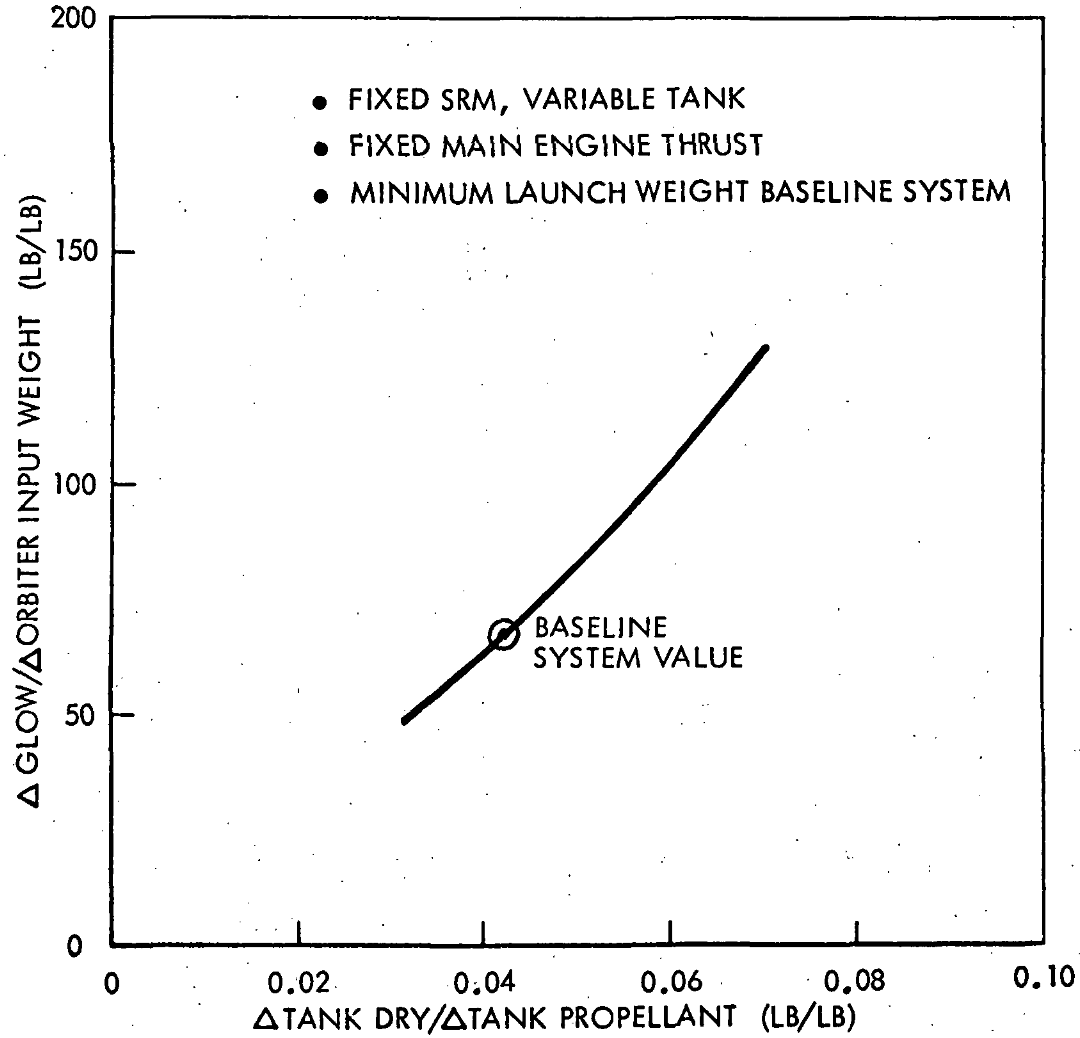

Figure 4.6-9 Fixed-Capability Sensitivities Increase As Structural Efficiency Decreases

a derivation of the sensitivity (ΔLaunch Weight/ΔOrbiter Input Weight) is given in Figure 4.7-1. An expression for the launch weight (W_{IGN}) is first derived in terms of the inert weight (W_{IN}), ideal velocity required (ΔV) and specific impulse (I_{SP}). The stage inert weight-scaling law is given as

$$W_{IN} = C + K = W_p$$

The launch weight is then differentiated, giving the sensitivity. Note that the sensitivity increases as the ideal velocity and the factor K increases.

DYNAMIC EQUATIONS

$$F = M\dot{V}$$

$$F = T = I_{SP}\dot{W}_P$$

$$\Delta V = g_0 I_{SP} \operatorname{Log} \frac{W_{IGN}}{W_{BO}}$$

WEIGHT SCALING RELATIONS AND WEIGHT BUILDUP

$$W_{IN} = C + K W_P$$

$$W_{BO} = W_{IN} + W_{PL}$$

$$W_{IGN} = W_{BO} + W_P$$

$$W_{IGN} = (1 + K)(W_{PL} + C)\frac{(e^{\Delta V/g I_{SP}} - 1)}{(1 + K - Ke^{\Delta V/g I_{SP}})} + W_{PL} + C$$

DERIVATIVE:

$$\frac{dW_{IGN}}{d W_{PL}} = e^{\Delta V/g I_{SP}} \frac{(1 - K/(1 + K))}{(1 - \frac{K}{1+K} e^{\Delta V/g I_{SP}})}$$

$$(1)\ \frac{dW_{IGN}}{dW_{PL}} \rightarrow \infty \text{ AS } \Delta V \rightarrow \infty \ ; \quad (2)\ \frac{dW_{IGN}}{dW_{PL}} \rightarrow \infty \text{ AS } K \rightarrow 1/(e^{\Delta V/g I_{SP}} - 1)$$

Figure 4.7-1 Example - Simplified Fixed Capability Weight Sensitivity Derivation

Section 5

STUDY RESULTS

5.1 SUMMARY OF RESULTS

The principal results of this study are the values of the sensitivities tabulated in this section. Tables 5.1-1, 5.1-2, and 5.1-3 summarize the most significant results, as follows:

- Table 5.1-1 summarizes system weight sensitivities to orbiter structure weight for each configuration and program phase.
- Table 5.1-2 summarizes the most significant program cost sensitivities to structure weight of each vehicle element of each configuration for the three program phases studied.
- Table 5.1-3 provides a complete listing of fixed-vehicle sensitivity results. These are payload capability and program cost sensitivities applicable in the test/operations phase when it is assumed that the vehicle can no longer be resized to maintain capability.

A final table in this section, Table 5.1-4, provides an index to the 48 tables with detailed listing of fixed-capability sensitivities (Tables 5.2-1 through 5.6-6).

The preliminary design and detailed design weight sensitivities in Table 5.1-1 are fixed-capability sensitivities.* That is they reflect vehicle redesign to maintain fixed system performance capability. The pattern of values may be understood by first observing the indirect dry weight sensitivities or the orbiter, detailed design phase, which range from 0.23 to 0.45 pounds per pound. These values involve similar changes to different configurations to

*Also referred to as fixed-performance sensitivities.

TABLE 5.1-1 SUMMARY OF WEIGHT SENSITIVITIES TO ORBITER STRUCTURAL WEIGHT

SYSTEM WEIGHT EFFECTS OF ORBITER WEIGHT CHANGES (POUNDS PER POUND)

		PREDESIGN PHASE			DETAILED DESIGN PHASE			TEST/OP PHASE
		IND DRY*	TOT DRY	LIFTOFF	IND DRY	TOT DRY	LIFTOFF	PAYLOAD
A. STAGE-AND-ONE-HALF	ORBITER	1.35	2.35	2.36	0.23	1.23	1.23	
	DROPTANK	1.12	1.12	27.59	1.06	1.06	26.37	
	TOTAL			29.95			27.60	1.00
B. STAGE-AND ONE-HALF WITH EXTERNAL TANK	ORBITER	1.23	2.23	2.43	0.25	1.25	1.33	
	EXTERNAL TANK	0.28	0.28	7.03	0.98	0.98	24.32	
	DROPTANK	0.67	0.67	16.60	0	0	0	
	TOTAL			26.06			25.65	1.00
C. DELTA-BODY ROCKET-ASSISTED ORBITER	ORBITER	0.30	1.30	1.37	0.30	1.30	1.37	
	EXTERNAL TANK	0.20	0.20	4.97	0	0	0	
	SRM	3.72	3.72	35.32	3.97	3.97	40.36	
	TOTAL			41.61			41.73	1.00
D. DELTA-WING ROCKET-ASSISTED ORBITER	ORBITER	0.45	1.45	1.63	0.45	1.45	1.63	
	EXTERNAL TANK	0.27	0.27	6.66	0	0	0	
	SRM	5.04	5.04	46.24	5.47	5.47	53.57	
	TOTAL			54.53			55.20	1.00
E. TWO-STAGE EXT ORB TANK HEAT-SINK BOOSTER	ORBITER	0.85	1.85	2.04	0.43	1.43	1.58	
	EXTERNAL TANK	0.50	0.50	10.52	2.04	2.04	42.82	
	BOOSTER	6.78	6.78	28.66	0	0	-2.06	
	TOTAL			41.22			42.34	1.00

*INDIRECT WEIGHT IS GROWTH NOT INCLUDING INPUT WEIGHT AND REFLECTS THE CHANGES WHICH PRODUCE THE "COST OF A FREE POUND".

TABLE 5.1-2 SYSTEM COST SENSITIVITY SUMMARY

DOLLARS PER POUND (FREE INPUT WEIGHT)

CONFIG	FLTS IN PROG	COST CATEGORY	EFFECTS OF ORBITER WEIGHT CHANGES			EFFECTS OF EXTERNAL TANK WEIGHT CHANGES			EFFECTS OF BOOSTER (OR DROPTANK) WEIGHT CHANGES		
			PRE-DES PHASE	DET DES PHASE	TEST/OPS PHASE	PRE-DES PHASE	DET DES PHASE	TEST/OPS PHASE	PRE-DES PHASE	DET DES PHASE	TEST/OPS PHASE
STAGE-AND-ONE-HALF A	250	TOT	13,523	6,824					4,446	1,943	
	500	NREC	8,839	1,664					2,872	259	
		REC	10,977	9,603		N/A	N/A	N/A	3,681	3,073	
		TOT	19,816	11,267	85,000				6,553	3,332	43,500
	750	TOT	26,305	15,428					8,725	4,618	
STAGE-AND-ONE-HALF EXT TANK B	500	TOT	18,500	11,381	85,000	13,037	7,719	80,000	3,119	1,663	20,600
RAO, Δ BODY C	500	TOT	26,459	27,743	85,000	18,120	19,025	80,000	1,574	1,675	7,100
RAO, Δ WING D	250	TOT	20,684	21,760		11,727	12,412		1,053	1,134	
	500	NREC	4,395	4,410		1,408	1,420		128	130	
		REC	30,519	32,515		19,027	20,294		1,705	1,854	
		TOT	34,914	36,925	85,000	20,435	21,714	78,000	1,833	1,984	8,400
	750	TOT	48,817	51,797		28,882	30,774		2,590	2,812	
TWO STAGE (WITH EXTERNAL TANK)	500	TOT	41,647	27,546	85,000	25,073	13,785	78,000	4,259	3,623	7,800

TABLE 5.1-3 FIXED-VEHICLE SENSITIVITIES

CONFIGURATION	INPUT WEIGHT IN	PAYLOAD CAPABILITY LOSS (LB/LB) EAST MISSION	RESUPPLY MISSION	POLAR MISSION	PROGRAM COST CHANGE,* 500 FLTS ($/LB)
A: STAGE-AND-ONE-HALF	ORBITER DROPTANK	1.00 0.446	1.00 0.455	1.00 0.512	85,000 39,500
B: STAGE-AND-ONE-HALF WITH EXTERNAL TANK	ORBITER EXTERNAL TANK DROPTANK	1.00 0.925 0.215	1.00 0.885 0.218	1.00 0.943 0.242	85,000 78,200 18,900
C: DELTA BODY ROCKET-ASSISTED ORBITER	ORBITER EXTERNAL TANK SRM	1.00 0.924 0.073	1.00 0.883 0.074	1.00 0.944 0.084	85,000 78,100 6,460
D: DELTA WING ROCKET-ASSISTED ORBITER	ORBITER EXTERNAL TANK SRM	1.00 0.892 0.086	1.00 0.834 0.087	1.00 0.918 0.099	85,000 75,100 7,620
E: TWO-STAGE EXTERNAL TANK ORBITER, HEAT SINK BOOSTER	ORBITER EXTERNAL TANK BOOSTER	1.00 0.893 0.081	1.00 0.836 0.083	1.00 0.919 0.092	85,000 75,300 7,170

*ASSUMES 50 PERCENT EAST MISSIONS, 25 PERCENT RESUPPLY, 25 PERCENT POLAR

TABLE 5.1-4 INDEX TABLE FOR FIXED-CAPABILITY SENSITIVITY TABLES

CONFIGURATION A TABLES

	Preliminary Design Phase		Detailed Design Phase	
	Orbiter	Drop Tank	Orbiter	Drop Tank
500-flt. prog.	5.2-1	5.2-2	5.2-3	5.2-4
250-flt. prog.	5.2-5	5.2-6	5.2-7	5.2-8
750-flt. prog.	5.2-9	5.2-10	5.2-11	5.2-12

CONFIGURATION B TABLES (500-flt. Prog. only)

P-D Phase			D-D Phase		
Orbiter	External Tank	Drop Tank	Orbiter	External Tank	Drop Tank
5.3-1	5.3-2	5.3-3	5.3-4	5.3-5	5.3-6

CONFIGURATION C TABLES (500 flt. Prog. only)

P-D Phase			D-D Phase		
Orbiter	External Tank	SRM	Orbiter	External Tank	SRM
5.4-1	5.4-2	5.4-3	5.4-4	5.4-5	5.4-6

CONFIGURATION D TABLES

	P-D Phase			D-D Phase		
	Orbiter	Ext.Tank	SRM	Orbiter	External Tank	SRM
500 flts:	5.5-1	5.5-2	5.5-3	5.5-4	5.5-5	5.5-6
250 flts:	5.5-7	5.5-8	5.5-9	5.5-10	5.5-11	5.5-12
750 flts:	5.5-13	5.5-14	5.5-15	5.5-16	5.5-17	5.5-18

CONFIGURATION E TABLES (500-flt. Prog. only)

P-D Phase			D-D Phase		
Orbiter	External Tank	Booster	Orbiter	External Tank	Booster
5.6-1	5.6-2	5.6-3	5.6-4	5.6-5	5.6-6

maintain the same set of requirements: on-orbit maneuvers, reentry crossrange, flying and landing loads, and landing speed. The increase from 0.23 to 0.30 from Configuration A through B to C reflects the increasing sensitivity of smaller delta-body orbiters (a given change is a larger percentage of the basic structure). The fact that the delta wing orbiter is about 50 percent more sensitive than the delta-body (Configuration D versus Configuration C) is an unexpected result and is discussed in some detail in Section 4.2. It stems from three sources:

1. There is a significantly lower sensitivity of fin weight to landing weight (delta-body) than wing weight to landing weight (delta wing) to maintain landing speed.
2. The body structure of the delta-body requires less redesign for structural integrity because line loads are smaller (much of it is already minimum gage).
3. The auxiliary propulsion system (OMPS and RCS) for the delta-body uses a common supply system and has high specific impulse rather than having separate systems (and modularized) with moderate specific impulse so that the tankage growth is much less severe.

Other variations of the weight sensitivities arise from resizing groundrules (see Figure 2-2). For instance, the orbiter dry weight sensitivities for Configurations A, B, and E are higher in preliminary design than in detailed design because main engine thrust is changed in the one case and not in the other. (Configuration E orbiter engine is changed to maintain commonality with the booster engine.) Superimposed on these engine weight changes are, of course, greater changes in all the indirect effects.

Variations in the growth of lower stages also shows examples of the effects of resizing constraints. For instance, the stage-and-one-half systems (Configurations A and B) become less sensitive in the detailed design phase than in the preliminary design phase, while the other configurations become more sensitive. The former effect reflects a shift from constant thrust-to-weight sizing to constant thrust sizing. The orbiter growth becomes much less with no engine change which more than compensates for the effects of less efficient ascent (higher gravity losses at lower thrust-to-weight). The increasing sensitivity of Configurations C, D, and E arises from less effective resizing when only one lower stage is changed. The fact that these are small increases indicates that there is little penalty for freezing the size of the other stage.

The system cost sensitivities in Table 5.1-2 summarize the most important results of the study. Several aspects of the variations in these sensitivities are discussed in Section 1. One additional parameter shown in Table 5.1-2 is the effect of program size on cost sensitivities for Configurations A and D. Nonrecurring effects have a much bigger effect for A than D. and a 50 percent change in the operational program size has a somewhat smaller effect on Configuration A total program cost sensitivities than is the case for Configuration D.

Tables 5.2-1 through 5.6-6 provide detailed listings of fixed-capability sensitivity results. Each table gives:

(1) Performance sensitivities for all significant cost drivers and for each vehicle element*
(2) Direct cost sensitivities for each cost driver
(3) System cost sensitivities computed by summing the products of the appropriate performance and direct cost sensitivities.

These tables are numbered as shown in Table 5.1-4.

*The term "performance sensitivities" includes both weight sensitivities and sensitivities of other parameters, specifically engine thrust in these tables.

Table 5.2-1

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Orbiter in Prelim Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.296	4,822	1,181	82	6,085
	Plumbing Wt	0.145	4,155	1,560	1,025	6,740
	TPS Wt	0.187	2,363	353	1,713	4,429
	Lndg Gear Wt	0.096	2,693	1,002	178	3,873
	S.L. Thrust/Eng (W 11 Engines)	3.46	921	385	152	1,458
Tank	Dry Wt	1.12	761	–	7,482	8,243
	Prop Wt	26.47	–	–	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	1.35	2.35	3.26	5,917	2,070	1,036	9,023	15,108
Drop Tank	1.12	1.12	27.59	0	9,941	10,793	10,793	10,793
Total			29.95	6,764	2,070	10,977	19,816	25,590
				8,839				

Table 5.2-2

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Droptank in Prelim Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.086	4,822	1,181	82	6,085
	Plumbing Wt	0.039	4,155	1,560	1,025	6,740
	TPS Wt	0.034	2,363	353	1,713	4,429
	Lndg Gear Wt	0.018	2,693	1,002	178	3,873
	S. L. Thrust/Eng (W. 11 Engines)	1.29	921	385	152	1,458
Tank	Dry Wt	.38	761	–	7,482	8,243
	Prop. Wt	9.03	–	–	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter	0.420	0.420	.422	1,894	689	305	2,888		2,888	
Droptank	0.38	1.38	10.41	289	0	3,376	3,665		11,908	
Total			10.83	2,183	689	3,681	6,553		14,796	
				2,872						

Table 5.2-3

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Orbiter in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.068	4,822	1,181	82	6,085
	Plumbing Wt	0	4,155	1,560	1,025	6,740
	TPS Wt	0.096	2,363	353	1,713	4,429
	Lndg Gear Wt	0.051	2,693	1,002	178	3,873
	S.L. Thrust/Eng (W. 11 Engines)	0	921	385	152	1,458
Tank	Dry Wt	1.06	761	–	7,482	8,243
	Prop. Wt	25.31	–	–	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.23	1.23	1.23	692	165	179	1.036	7,121
Droptank	1.06	1.06	26.37	807	0	9,424	10,231	10,231
Total			27.60	1,499	165	9,603	11,267	17,352
				1,664				

Table 5.2-4

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Droptank in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0	4,822	1,181	82	6,085
Orbiter	Plumbing Wt	0	4,155	1,560	1,025	6,740
Orbiter	TPS Wt	0	2,363	353	1,713	4,429
Orbiter	Lndg Gear Wt	0	2,693	1,002	178	3,873
Orbiter	S. L. Thrust/Eng (W. 11 Engines)	0	921	385	152	1,458
Tank	Dry Wt	0.34	761	–	7,482	8,243
Tank	Prop. Wt	8.97	–	–	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt: Total Prog (Costed Input Wt)
Orbiter	0	0	0	0	0	0	0	0
Droptank	0.34	1.34	9.31	259	0	3,073	3,332	11,575
Total			9.31	259	0	3,073	3,332	11,575
				259 (Dev + Prod)				

Table 5.2-5

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Orbiter in Prelim. Design Phase for 250 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.296	4,822	432	75	5,329
	Plumbing Wt	0.145	4,155	570	558	5,283
	TPS Wt	0.187	2,363	129	866	3,358
	Lndg Gear Wt	0.096	2,693	366	117	3,176
	S.L. Thrust/Eng (W. 11 Engines)	3.46	921	135	83	1,139
Tank	Dry Wt	1.12	761	–	4,161	4,922
	Prop. Wt	26.47	–	–	30	30

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	1.35	2.35	2.36	5,917	712	538	7,217	12,546
Droptank	1.12	1.12	27.59	852	0	5,454	6,306	6,306
Total			29.95	6,520	712	5,992	13,523	18,852
				7,232 (Dev + Prod)				

Table 5.2-6

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Droptank in Prelim. Design Phase for 250 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb)			
			Nonrecur		Recur	Total Prog
			Dev	Prod		
Orbiter	Structure Wt	0.086	4,822	432	75	5,329
Orbiter	Plumbing Wt	0.039	4,155	570	558	5,283
Orbiter	TPS Wt	0.034	2,363	129	866	3,358
Orbiter	Lndg Gear Wt	0.018	2,693	366	117	3,176
Orbiter	S. L. Thrust/Eng (W. 11 Engines)	1.29	921	135	83	1,139
Tank	Dry Wt	0.38	761	–	4,161	4,922
Tank	Prop. Wt	9.03	–	–	30	30

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb)				
				Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.420	0.420	0.422	1,826	238	161	2,305	2,305
Droptank	0.38	1.38	10.41	289	0	1,852	2,141	7,063
Total			10.83	2,115	238	2,013	4,446	9,368
				2,353				

Table 5.2-7

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Orbiter in Detailed Design Phase for 250 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.068	4,822	432	75	5,329
Orbiter	Plumbing Wt	0	4,155	570	558	5,283
Orbiter	TPS Wt	0.096	2,363	129	866	3,358
Orbiter	Lndg Gear Wt	0.051	2,693	366	117	3,176
Orbiter	S.L. Thrust/Eng (W. 11 Engines)	0	921	135	83	1,139
Tank	Dry Wt	1.06	761	–	4,161	4,922
Tank	Prop. Wt	25.31	–	–	30	30

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	Free Input Wt: System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Costed Input Wt: Total Prog
Orbiter	0.23	1.23	1.23	628	55	85	847	6,176
Droptank	1.06	1.06	26.37	807	0	5,170	5,977	5,977
Total			27.60	1,435	55	5,255	6,824	12,153
				1,490 (Dev + Prod)				

Table 5.2-8

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Droptank in Detailed Design Phase for 250 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0	4,822	432	75	5,329
	Plumbing Wt	0	4,155	570	558	5,283
	TPS Wt	0	2,363	129	866	3,358
	Lndg Gear Wt	0	2,693	366	117	3,176
	S. L. Thrust/Eng (W. 11 Engines)	0	921	135	83	1,139
Tank	Dry Wt	0.34	761	–	4,161	4,922
	Prop. Wt	8.97	–	–	30	30

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter	0	0	0	0	0	0	0		0	
Droptank	0.34	1.34	9.31	259	0	1,684	1,943		6,865	
Total			9.31	259	0	1,684	1,943		6,865	
				259						

Table 5.2-9

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Orbiter in Prelim. Design Phase for 750 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities — ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.296	4,822	2,190	89	7,101
	Plumbing Wt	0.145	4,155	2,892	1,493	8,540
	TPS Wt	0.187	2,363	654	2,559	5,576
	Lndg Gear Wt	0.096	2,693	1,859	238	4,790
	S. L. Thrust/Eng (W. 11 Engines)	3.46	921	741	222	1,884
Tank	Dry Wt	1.12	761	—	10,526	11,287
	Prop. Wt	26.47	—	—	87	87

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities — ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	1.35	2.35	2.36	5,668			11,361	18,462
Droptank	1.12	1.12	27.59	852			14,944	14,944
Total			29.95	6,520			26,305	33,406

Table 5.2-10

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Droptank in Prelim. Design Phase for 750 Flight Programs

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.086	4,822	2,190	89	7,101
	Plumbing Wt	0.039	4,155	2,892	1,493	8,540
	TPS Wt	0.034	2,363	654	2,559	5,576
	Lndg Gear Wt	0.018	2,693	1,859	238	4,790
	S. L. Thrust/Eng (W. 11 Engines)	1.29	921	741	222	1,884
Tank	Dry Wt	0.38	761	–	10,526	11,287
	Prop. Wt	9.03	–	–	87	87

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.420	0.420	0.422	1,826			3,650	3,650
Droptank	0.38	1.38	10.41	289			5,075	16,362
Total			10.83	2,115			8,725	20,012

Table 5.2-11

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Orbiter in Detailed Design Phase for 750 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.068	4,822	2,190	89	7,101
	Plumbing Wt	0	4,155	2,892	1,493	8,540
	TPS Wt	0.096	2,363	654	2,559	5,576
	Lndg Gear Wt	0.051	2,693	1,859	238	4,790
	S. L. Thrust/Eng (W. 11 Engines)	0	921	741	222	1,884
Tank	Dry Wt	1.06	761	–	10,526	11,287
	Prop. Wt	25.31	–	–	87	87

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.23	1.23	1.23	628			1,262	8,363
Droptank	1.06	1.06	26.37	807			14,166	14,166
Total			27.60	1,435			15,428	22,529

Table 5.2-12

CONFIGURATION A SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Droptank in Detailed Design Phase for 750 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0	4,822	2,190	89	7,101
	Plumbing Wt	0	4,155	2,892	1,493	8,540
	TPS Wt	0	2,363	654	2,559	5,576
	Lndg Gear Wt	0	2,693	1,859	238	4,790
	S.L. Thrust/Eng (W. 11 Engines)	0	921	741	222	1,884
Tank	Dry Wt	0.34	761	–	10,526	11,287
	Prop. Wt	8.97	–	–	87	87

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter	0	0	0	0	0	0	0		0	
Droptank	0.34	1.34	9.31	259			4,618		15,905	
Total			9.31	259			4,618		15,905	

Table 5.3-1

CONFIGURATION B SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Orbiter in Prelim. Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.272	4,996	1,222	85	6,303
Orbiter	Plumbing Wt	0.143	4,095	1,541	1,011	6,647
Orbiter	TPS Wt	0.189	2,741	368	1,790	4,898
Orbiter	Lndg Gear Wt	0.096	2,771	1,009	183	3,963
Orbiter	S.L. Thrust/Eng (W. 9 Engines)	3.62	849	304	121	1,274
	Ext. Tank Dry Wt	0.281	769	0	8,290	9,059
	Ext. Tank Prop Wt	6.75	0	0	59	59
	Droptank Dry Wt	0.666	769	0	8,290	9,059
	Droptank Prop. Wt	15.93	0	0	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter	1.23	2.23	2.43	5,802	1,820	961	8,583		14,886	
External Tank	0.28	0.28	7.03	216	0	2,728	2,944		2,944	
Droptank Set	0.67	0.67	16.60	512	0	6,461	6,973		6,973	
Total			26.06	6,530	1,820	10,150	18,500		24,803	
				8,550						

Table 5.3-2

CONFIGURATION B SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Ext. Tank in Prelim. Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.150	4,996	1,222	85	6,303
Orbiter	Plumbing Wt	0.107	4,095	1,541	1,011	6,647
Orbiter	TPS Wt	0.066	2,741	368	1,790	4,898
Orbiter	Lndg Gear Wt	0.032	2,771	1,009	183	3,963
Orbiter	S. L. Thrust/Eng (W. 9 Engines)	2.72	849	304	121	1,274
	Ext. Tank Dry Wt	0.212	769	0	8,290	9,059
	Ext. Tank Prop. Wt	5.08	0	0	59	59
	Droptank Dry Wt	0.501	769	0	8,290	9,059
	Droptank Prop. Wt	11.98	0	0	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog (Costed Input Wt)
Orbiter	0.74	0.74	0.81	3,766	1,232	574	5,572		5,572
External Tank	0.21	1.21	6.29	163	0	2,057	2,220		11,279
Droptank Set	0.50	0.50	12.48	385	0	4,860	5,245		5,245
Total			19.58	4,314	1,232	7,491	13,037		22,096
				5,546					

Table 5.3-3

CONFIGURATION B SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Droptank in Prelim. Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.039	4,996	1,222	85	6,303
Orbiter	Plumbing Wt	0.028	4,095	1,541	1,011	6,647
Orbiter	TPS Wt	0.017	2,741	368	1,790	4,898
Orbiter	Lndg Gear Wt	0.008	2,771	1,009	183	3,963
Orbiter	S.L. Thrust/Eng (W. 9 Engines)	0.720	849	304	121	1,274
	Ext. Tank Dry Wt	0.047	769	0	8,290	9,059
	Ext. Tank Prop.	1.13	0	0	59	59
	Droptank Dry Wt	0.111	769	0	8,290	9,059
	Droptank Prop. Wt	2.66	0	0	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.19	0.19	0.21	990	324	151	1,465	1,465
External Tank	0.05	0.05	1.18	36	0	456	492	492
Droptank Set	0.11	1.11	3.77	85	0	1,077	1,162	10,221
Total			5.16	1,111	324	1,684	3,119	12,178
				1,435				

Table 5.3-4

CONFIGURATION B SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Orbiter in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.066	4,996	1,222	85	6,303
Orbiter	Plumbing Wt	0	4,095	1,541	1.011	6,647
Orbiter	TPS Wt	0.098	2,741	368	1,790	4,898
Orbiter	Lndg Gear Wt	0.049	2,771	1,009	183	3,963
Orbiter	S. L. Thrust/Eng (W. 9 Engines)	0	849	304	121	1,274
	Ext. Tank Dry Wt	0.984	769	0	8,290	9,059
	Ext. Tank Prop. Wt	23.34	0	0	59	59
	Droptank Dry Wt	0	769	0	8,290	9,059
	Droptank Prop. Wt	0	0	0	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.25	1.25	1.33	734	166	190	1,090	7,393
External Tank	0.98	0.98	24.32	757	0	9,534	10,291	10,291
Droptank Set	0	0	0	0	0	0	0	0
Total			25.65	1,491	166	9,724	11,381	17,684
				1,657				

Table 5.3-5

CONFIGURATION B SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Ext. Tank in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb)			
			Nonrecur		Recur	Total Prog
			Dev	Prod		
Orbiter	Structure Wt	0	4,996	1,222	85	6,303
Orbiter	Plumbing Wt	0	4,095	1,541	1,011	6,647
Orbiter	TPS Wt	0	2,741	368	1,790	4,898
Orbiter	Lndg Gear Wt	0	2,771	1,009	183	3,963
Orbiter	S. L. Thrust/Eng (W. 9 Engines)	0	849	304	121	1,274
	Ext. Tank Dry Wt	0.738	769	0	8,290	9,059
	Ext. Tank Prop. Wt	17.51	0	0	59	59
	Droptank Dry Wt	0	769	0	8,290	9,059
	Droptank Prop. Wt	0	0	0	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb)				
				Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0	0	0	0	0	0	0	0
External Tank	0.74	1.74	19.25	567	0	7,152	7,719	16,778
Droptank Set	0	0	0	0	0	0	0	0
Total			19.25	567	0	7,152	7,719	16,778

Table 5.3-6

CONFIGURATION B SENSITIVITIES

(Stage-And-One-Half)

Effects of One Pound of Droptank in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0	4,996	1,222	85	6,303
Orbiter	Plumbing Wt	0	4,095	1,541	1,011	6,647
Orbiter	TPS Wt	0	2,741	368	1,790	4,898
Orbiter	Lndg Gear Wt	0	2,771	1,009	183	3,963
Orbiter	S. L. Thrust/Eng (W. 9 Engines	0	849	304	121	1,274
	Ext. Tank Dry Wt	0.159	769	0	8,290	9,059
	Ext. Tank Prop. Wt	3.77	0	0	59	59
	Droptank Dry Wt	0	769	0	8,290	9,059
	Droptank Prop. Wt	0	0	0	59	59

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0	0	0	0	0	0	0	0
External Tank	0.16	0.16	3.93	122	0	1,541	1,663	1,663
Droptank Set	0	1.00	1.00	0	0	0	0	9,059
Total			4.93	122	0	1,541	1,663	10,722

Table 5.4-1

CONFIGURATION C SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Body Orbiter)

Effects of One Pound of Orbiter in Prelim. Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.106	5,441	1,309	92	6,842
Orbiter	Plumbing Wt	0	7,317	2,027	1,327	10,671
Orbiter	TPS Wt	0.111	3,011	439	2,137	5,587
Orbiter	Lndg Gear Wt	0.055	2,953	1,100	194	4,247
Orbiter	S.L. Thrust/ Eng	0	766	112	44	922
	Ext. Tank Dry Wt	0.197	1,153	0	8,399	9,552
	Ext. Tank Prop. Wt	4.72	0	0	59	59
	SRM Booster Dry Wt	N/A				(7,500)*
	SRM Booster Prop. Wt	31.60	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.30	1.30	1.37	1,073	248	258	1,579	8,421
External Tank	0.20	0.20	4.92	227	0	1,933	2,160	2,160
SRM Booster	3.72	3.72	35.32	1,485	0	21,235	22,720	22,720
Total			41.61	2,785	248	23,426	26,459	33,301

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellent weight.

Table 5.4-2

CONFIGURATION C SENSITIVITIES

(Rocket-Assisted Orbiter with ΔBody Orbiter)

Effects of One Pound of Ext. Tank in Prelim. Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0	5,441	1,309	92	6,842
	Plumbing Wt	0	7,317	2,027	1,327	10,671
	TPS WT	0	3,011	439	2,137	5,587
	Lndg Gear Wt	0	2,953	1,100	194	4,247
	S. L Thrust/Eng	0	766	112	44	922
	Ext. Tank Dry Wt	0.141	1,153	0	8,399	9,552
	Ext. Tank Prop. Wt	3.38	0	0	59	59
	SRM Booster Dry Wt	N/A				(7,500)*
	SRM Booster Prop. Wt	23.05	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0	0	0	0	0	0	0	0
External Tank	0.14	1.14	4.52	163	0	1,384	1,547	11,099
SRM Booster	2.71	23.05	25.76	1,083	0	15,490	16,573	16,573
Total			30.28	1,246	0	18,874	18,120	27,672

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellent weight.

Table 5.4-3

CONFIGURATION C SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Body Orbiter)

Effects of One Pound of Booster in Prelim Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities — ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0	5,441	1,309	92	6,842
Orbiter	Plumbing Wt	0	7,317	2,027	1,327	10,671
Orbiter	TPS Wt	0	3,011	439	2,137	5,587
Orbiter	Lndg Gear Wt	0	2,953	1,100	194	4,247
Orbiter	S. L. Thrust/Eng	0	766	112	44	922
	Ext. Tank Dry Wt	0.017	1,153	0	8,399	9,552
	Ext. Tank Prop. Wt	0.40	0	0	59	59
	SRM Booster Dry Wt	N/A				(7,500)*
	SRM Booster Prop. Wt	1.93	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities — ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0	0	0	0	0	0	0	0
External Tank	0.02	0.02	0.42	20	0	166	186	186
SRM Booster	0.28	1.28	3.21	91	0	1,297	1,388	8,888
Total			3.63	111	0	1,463	1,574	9,074

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellent weight.

Table 5.4-4

CONFIGURATION C SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Body Orbiter)

Effects of One Pound of <u>Orbiter</u> in <u>Detailed</u> Design Phase for <u>500</u> Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0.106	5,441	1,309	92	6,842
Orbiter	Plumbing Wt	0	7,317	2,027	1,327	10,671
Orbiter	TPS Wt	0.111	3,011	439	2,137	5,587
Orbiter	Lndg Gear Wt	0.055	2,953	1,100	194	4,247
Orbiter	S.L. Thrust/Eng	0	766	112	44	922
	Ext. Tank Dry Wt	0	1,153	0	8,399	9,552
	Ext. Tank Prop. Wt	0	0	0	59	59
	SRM Booster Dry Wt	N/A				(7,500)*
	SRM Booster Prop. Wt	36.39	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.30	1.30	1.37	1,073	248	258	1,579	8,421
External Tank	0	0	0	0	0	0	0	0
SRM Booster	3.97	3.97	40.36	1,710	0	24,454	26,164	26,164
Total			41.73	2,783	248	24,712	27,743	34,585

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellent weight.

Table 5.4-5

CONFIGURATION C SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Body Orbiter)

Effects of One Pound of Ext. Tank in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0	5,441	1,309	92	6,842
Orbiter	Plumbing Wt	0	7,317	2,027	1,327	10,671
Orbiter	TPS Wt	0	3,011	439	2,137	5,587
Orbiter	Lndg Gear Wt	0	2,953	1,100	194	4,247
Orbiter	S.L. Thrust/Eng	0	766	112	44	922
	Ext. Tank Dry Wt	0	1,153	0	8,399	9,552
	Ext. Tank Prop. Wt	0	0	0	59	59
	SRM Booster Dry Wt	N/A				(7,500)*
	SRM Booster Prop. Wt	26.46	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Costed Input Wt Total Prog
Orbiter	0	0	0	0	0	0	0		0
External Tank	0	1.00	1.00	0	0	0	0		9,552
SRM Booster	2.89	2.89	29.35	1,244	0	17,781	19,025		19,025
Total			30.35	1,244	0	17,781	19,025		28,577

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellent weight.

Table 5.4-6

CONFIGURATION C SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Body Orbiter)

Effects of One Pound of Booster in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt	0	5,441	1,309	92	6,842
Orbiter	Plumbing Wt	0	7,317	2,027	1,327	10,671
Orbiter	TPS Wt	0	3,011	439	2,137	5,587
Orbiter	Lndg Gear Wt	0	2,953	1,100	194	4,247
Orbiter	S. L. Thrust/Eng	0	766	112	44	922
	Ext. Tank Dry Wt	0	1,153	0	8,397	9,552
	Ext. Tank Prop. Wt	0	0	0	59	59
	SRM Booster Dry Wt	N/A				(7,500)*
	SRM Booster Prop. Wt	2.33	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Costed Input Wt Total Prog
Orbiter	0	0	0	0	0	0	0		0
External Tank	0	0	0	0	0	0	0		0
SRM Booster	0.30	1.30	3.63	110	0	1,565	1,675		9,175
Total			3.63	110	0	1,565	1,675		9,175

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellent weight.

Table 5.5-1

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter)

Effects of One Pound of Orbiter in Prelim. Design Phase for 500 Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	.255	5,250	1,263	89	6,602
Orbiter	Plumbing Wt (lb)	0	7,302	2,028	1,328	1,058
Orbiter	TPS Wt (lb)	.070	3,105	464	2,250	5,819
Orbiter	Lndg Gear Wt (lb)	.060	2,890	1,077	194	4,161
Orbiter	S. L. Thrust/Eng (lb)	0	766	112	44	922
	Ext Tank Dry Wt	.27	1,150	0	8,381	9,531
	Ext Tank Prop Wt	6.39	0	0	59	59
	SRM Booster Dry Wt (lb)	N/A				(7,500)*
	SRM Booster Prop Wt (lb)	41.20	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter	0.45	1.45	1.63	1,730	419	192	2,341		8,943	
External Tank	.27	.27	6.66	310	0	2,640	2,950		2,950	
SRM Booster	5.04	5.04	46.24	1,936	0	27,687	29,623		29,623	
Total			54.53	3,976	419	30,519	34,914		41,516	
				4,395						

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-2

CONFIGURATION D SENSITIVITIES

(Rocket Assisted ΔWing Orbiter)

Effects of One Pound of Ext Tank in Prelim Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0	5,250	1,263	89	6,602
	Plumbing Wt (lb)	0	7,302	2,028	1,328	1,058
	TPS Wt (lb)	0	3,105	464	2,250	5,819
	Lndg Gear Wt (lb)	0	2,890	1,077	194	4,161
	S.L. Thrust/Eng (lb)	0	766	112	44	922
	Ext Tank Dry Wt	0.166	1,150	0	8,381	9,531
	Ext Tank Prop Wt	3.92	0	0	59	59
	SRM Booster Dry Wt (lb)	N/A				(7,500)*
	SRM Booster Prop Wt (lb)	25.90	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Closed Input Wt
Orbiter	0	0	0	0	0	0	0		0	
External Tank	0.17	1.17	5.09	191	0	1,622	1,813		11,344	
SRM Booster	3.17	3.17	29.07	1,217	0	17,405	18,622		18,622	
Total			34.16	1,408	0	19,027	20,435		29,966	
				1,408 (Dev + Prod)						

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-3

CONFIGURATION D SENSITIVITIES

(Rocket Assisted Orbiter With ΔWing Orbiter)

Effects of One Pound of Booster in Prelim Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0	5,250	1,263	89	6,602
Orbiter	Plumbing Wt (lb)	0	7,302	2,028	1,328	1,058
Orbiter	TPS Wt (lb)	0	3,105	464	2,250	5,819
Orbiter	Lndg Gear Wt (lb)	0	2,890	1,077	194	4,161
Orbiter	S.L. Thrust/Eng (lb)	0	766	112	44	922
	Ext Tank Dry Wt	.019	1,150	0	8,381	9,531
	Ext Tank Prop Wt	0.46	0	0	59	59
	SRM Booster Dry Wt (lb)	N/A				(7,500)*
	SRM Booster Prop Wt (lb)	2.26	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Costed Input Wt Total Prog
Orbiter	0	0	0	0	0	0	0		0
External Tank	.02	.46	.47	22	0	186	208		208
SRM Booster	0.33	1.33	3.59	106	0	1,519	1,625		9,125
Total			4.06	128	0	1,705	1,833		9,333
				128					

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-4

CONFIGURATION D SENSITIVITIES

(Rocket Assisted Orbiter With Δ Wing Orbiter)

Effects of One Pound of Orbiter in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	.255	5,250	1,263	89	6,602
Orbiter	Plumbing Wt (lb)	0	7,302	2,028	1,328	1,058
Orbiter	TPS Wt (lb)	.070	3,105	464	2,250	5,819
Orbiter	Lndg Gear Wt (lb)	.060	2,890	1,077	194	4,161
Orbiter	S.L. Thrust/Eng (lb)	0	766	112	44	922
	Ext Tank Dry Wt	0	1,150	0	8,381	9,531
	Ext Tank Prop. Wt	0	0	0	59	59
	SRM Booster Dry Wt (lb)	N/A				(7,500)*
	SRM Booster Prop. Wt (lb)	48.10	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Costed Input Wt Total Prog
Orbiter	0.45	1.45	1.63	1,730	419	192	2,341		8,943
External Tank	0	0	0	0	0	0	0		0
SRM Booster	5.47	5.47	53.57	2,261	0	32,323	34,584		34,584
Total			55.20	3,991	419	32,515	36,925		43,527
				4,410					

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-5

CONFIGURATION D SENSITIVITIES

(Rocket Assisted Orbiter With ΔWing Orbiter)

Effects of One Pound of Ext Tank in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sersitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0	5,250	1,263	89	6,602
Orbiter	Plumbing Wr (lb)	0	7,302	2,028	1,328	1,058
Orbiter	TPS Wt (lb)	0	3,105	464	2,250	5,819
Orbiter	Lndg Gear Wt (lb)	0	2,890	1,077	194	4,161
Orbiter	S.L. Thrust/Eng (lb)	0	766	112	44	922
	Ext Tank Dry Wt	0	1,150	0	8,381	9,531
	Ext Tank Prop. Wt	0	0	0	59	59
	SRM Booster Dry Wt (lb)	N/A				(7,500)*
	SRM Booster Prop. Wt (lb)	30.20	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities –($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog (Costed Input Wt)
Orbiter	0	0	0	0	0	0	0		0
External Tank	0	1.00	1.00	0	0	0	0		9,531
SRM Booster	3.44	3.44	33.64	1,420	0	20,294	21,714		21,714
Total			34.64	1,420	0	20,294	21,714		31,245
				1420					

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-6

CONFIGURATION D SENSITIVITIES

(Rocket Assisted Orbiter With ΔWing Orbiter)

Effects of One Pound of Booster in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb)			
			Nonrecur		Recur	Total Prog
			Dev	Prod		
Orbiter	Structure Wt (lb)	0	5,250	1,263	89	6,602
Orbiter	Plumbing Wt (lb)	0	7,302	2,028	1,328	1,058
Orbiter	TPS Wt (lb)	0	3,105	464	2,250	5,819
Orbiter	Lndg Gear Wt (lb)	0	2,890	1,077	194	4,161
Orbiter	S.L. Thrust/Eng (lb)	0	766	112	44	922
	Ext Tank Dry Wt	0	1,150	0	8,381	9,531
	Ext Tank Prop. Wt	0	0	0	59	59
	SRM Booster Dry Wt (lb)	N/A				(7,500)*
	SRM Booster Prop. Wt (lb)	2.76	47	0	672	719

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb)				Free Input Wt	Costed Input Wt
				Dev	Prod	Recur	Total Prog		Total Prog
Orbiter	0	0	0	0	0	0	0		0
External Tank	0	0	0	0	0	0	0		0
SRM Booster	0.36	1.36	4.12	130	0	1,854	1,984		9,484
Total			4.12	130	0	1,854	1,984		9,484
				130					

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-7

CONFIGURATION D SENSITIVITIES

(Rocket Assisted Orbiter With ΔWing Orbiter)

Effects of One Pound of Orbiter in Prelim Design Phase for 250 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	.255				5,792
Orbiter	Plumbing Wt (lb)	0				8,776
Orbiter	TPS Wt (lb)	.070				4,414
Orbiter	Lndg Gear Wt (lb)	.060				3,417
Orbiter	S.L. Thrust/Eng (lb)	0				828
	Ext Tank Dry Wt	27				5,810
	Ext Tank Prop. Wt	6.39				30
	SRM Booster Dry Wt (lb)	N/A				(5,250)*
	SRM Booster Prop. Wt (lb)	41.20				411

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter	0.45	1.45	1.63				1,991	7,793
External Tank	.27	.27	6.66				1,760	1,760
SRM Booster	5.04	5.04	41.24				16,933	16,933
Total			54.53				20,684	26,476

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-8

CONFIGURATION D SENSITIVITIES

(Rocket Assisted Orbiter With ΔWing Orbiter)

Effects of One Pound of Ext Tank in Prelim Design Phase for 250 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur			
			Dev	Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0				5,792
Orbiter	Plumbing Wt (lb)	0				8,776
Orbiter	TPS Wt (lb)	0				4,414
Orbiter	Lndg Gear Wt (lb)	0				3,417
Orbiter	S. L. Thrust/Eng (lb)	0				828
	Ext Tank Dry Wt	.166				5,810
	Ext Tank Prop. Wt.	3.92				30
	SRM Booster Dry Wt (lb)					(5,250)*
	SRM Booster Prop. Wt (lb)	25.90				411

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter							0		0	
External Tank							1,082		6,892	
SRM Booster							10,645		10,645	
Total							11,727		17,537	

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-9

CONFIGURATION D SENSITIVITIES

(Rocket Assisted Orbiter With ΔWing Orbiter)

Effects of One Pound of <u>Booster</u> in <u>Prelim</u> Design Phase for <u>250</u> Flight Program

	Cost Driver	Preliminary Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)					5,792
Orbiter	Plumbing Wt (lb)					8,776
Orbiter	TPS Wt (lb)					4,414
Orbiter	Lndg Gear Wt (lb)					3,417
Orbiter	S.L. Trhsut/Eng (lb)					828
	Ext Tank Dry Wt	.019				5,810
	Ext Tank Prop. Wt	0.46				30
	SRM Booster Dry Wt (lb)					(5,250)*
	SRM Booster Prop Wt (lb)	2.26				411

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total	Free Input Wt	Costed Input Wt Total
Orbiter							0		0
External Tank							124		124
SRM Booster							929		6,179
Total							1,053		6,303

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-10

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter)

Effects of One Pound of Orbiter in Detailed Design Phase for 250 Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb)			
			Nonrecur		Recur	Total Prog
			Dev	Prod		
Orbiter	Structure Wt (lb)	.255				5,792
Orbiter	Plumbing Wt (lb)	0				8,776
Orbiter	TPS Wt (lb)	.070				4,414
Orbiter	Lndg Gear Wt (lb)	.060				3,417
Orbiter	S. L. Thrust/Eng (lb)	0				828
	Ext Tank Dry Wt	0				5,810
	Ext Tank Prop Wt	0				30
	SRM Booster Dry Wt (lb)	N/A				(5,250)*
	SRM Booster Prop Wt (lb)	48.10				411

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb)				
				Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter							1,991	7,783
External Tank							0	0
SRM Booster							19,769	19,769
Total							21,760	27,552

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-11

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter)

Effects of One Pound of Ext Tank in Detailed Design Phase for 250 Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)					5,792
Orbiter	Plumbing Wt (lb)					8,776
Orbiter	TPS Wt (lb)					4,414
Orbiter	Lndg Gear Wt (lb)					3,417
Orbiter	S. L. Thrust/Eng (lb)					828
	Ext Tank Dry Wt					5,810
	Ext Tank Prop Wt					30
	SRM Booster Dry Wt (lb)	N/A				(5,250)*
	SRM Booster Prop Wt (lb)	30.20				411

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter							0	0
External Tank							0	5,810
SRM Booster							12,412	12,412
Total							12,412	18,222

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-12

CONFIGURATION D SENSITIVITES

(Rocket-Assisted Orbiger with Δ Wing Orbiter)

Effects of One Pound of Booster in Detailed Design Phase for 250 Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)					5,792
	Plumbing Wt (lb)					8,776
	TPS Wt (lb)					4,414
	Lndg Gear Wt (lb)					3,417
	S. L. Thrust/Eng (lb)					828
	Ext Tank Dry Wt					5,810
	Ext Tank Prop Wt					30
	SRM Booster Dry Wt (lb)	N/A				(5,250)*
	SRM Booster Prop Wt (lb)	2.76				411

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter				0	0	0	0		0	
External Tank				0	0	0	0		0	
SRM Booster							1,134		6,384	
Total							1,134		6,384	

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-13

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter)

Effects of One Pound of Orbiter in Prelim Design Phase for 750 Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivites – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	.255				7,688
Orbiter	Plumbing Wt (lb)	0				13,019
Orbiter	TPS Wt (lb)	.070				7,328
Orbiter	Lndg Gear Wt (lb)	.060				5,152
Orbiter	S. L. Thrust/Eng (lb)	0				1,049
	Ext Tank Dry Wt	.27				12,945
	Ext Tank Prop Wt	6.39				87
	SRM Booster Dry Wt (lb)	N/A				(9,750)*
	SRM Booster Prop Wt (lb)	41.20				1,019

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter							2,783		10,471	
External Tank							4,051		4,051	
SRM Booster							41,983		41,983	
Total							48,817		56,505	

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-14

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter)

Effects of One Pound of <u>Ext Tank</u> in <u>Prelim</u> Design Phase for <u>750</u> Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0				7,688
Orbiter	Plumbing Wt (lb)	0				13,019
Orbiter	TPS Wt (lb)	0				7,328
Orbiter	Lndg Gear Wt (lb)	0				5,152
Orbiter	S. L. Thrust/Eng (lb)	0				1,049
	Ext Tank Dry Wt	.166				12,945
	Ext Tank Prop Wt	3.92				87
	SRM Booster Dry Wt (lb)	–				(9,750)*
	SRM Booster Prop Wt (lb)	25.90				1,019

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter							0	
External Tank							2,490	15,435
SRM Booster							26,392	26,392
Total							28,882	41,827

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-15

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter)

Effects of One Pound of Booster in Prelim Design Phase for 750 Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog	
Orbiter	Structure Wt (lb)					7,688	
Orbiter	Plumbing Wt (lb)					13,019	
Orbiter	TPS Wt (lb)					7,328	
Orbiter	Lndg Gear Wt (lb)					5,152	
Orbiter	S. L. Thrust/Eng (lb)					1,049	
	Ext Tank Dry Wt	.019				12,945	
	Ext Tank Prop Wt	0.46				87	
	SRM Booster Dry Wt (lb)	-				(9,750)*	
	SRM Booster Prop Wt (lb)	2.26				1,019	

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog	Free Input Wt	Total Prog	Costed Input Wt
Orbiter							0		0	
External Tank							286		286	
SRM Booster							2,304		12,054	
Total							2,590		12,340	

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-16

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter)

Effects of One Pound of Orbiter in Detailed Design Phase for 750 Flt Program

Cost Driver		Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	.255				7,688
	Plumbing Wt (lb)	0				13,019
	TPS Wt (lb)	.070				7,328
	Lndg Gear Wt (lb)	.060				5,152
	S. L. Thrust/Eng (lb)	0				1,049
	Ext Tank Dry Wt	0				12,945
	Ext Tank Prop Wt	0				87
	SRM Booster Dry Wt (lb)					(9,750)*
	SRM Booster Prop Wt (lb)	48.10				1,019

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter							2,783	10,471
External Tank							0	0
SRM Booster							49,014	49,014
Total							51,797	59,485

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-17

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter

Effects of One Pound of Ext Tank in Detailed Design Phase for 750 Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0				7,688
Orbiter	Plumbing Wt (lb)	0				13,019
Orbiter	TPS Wt (lb)	0				7,328
Orbiter	Lndg Gear Wt (lb)	0				5,152
Orbiter	S. L. Thrust/Eng (lb)	0				1,049
	Ext Tank Dry Wt	0				12,945
	Ext Tank Prop Wt	0				87
	SRM Booster Dry Wt (lb)	N/A				(9,750)*
	SRM Booster Prop Wt (lb)	30.20				1,019

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter							0	0
External Tank							0	12,945
SRM Booster							30,774	30,774
Total							30,774	43,719

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.5-18

CONFIGURATION D SENSITIVITIES

(Rocket-Assisted Orbiter with Δ Wing Orbiter)

Effects of One Pound of Booster in Detailed Design Phase for 750 Flt Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities – ($/lb) Nonrecur Dev	Nonrecur Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0				7,688
Orbiter	Plumbing Wt (lb)	0				13,019
Orbiter	TPS Wt (lb)	0				7,328
Orbiter	Lndg Gear Wt (lb)	0				5,152
Orbiter	S. L. Thrust/Eng (lb)	0				1,049
	Ext Tank Dry Wt	0				12,945
	Ext Tank Prop Wt	0				87
	SRM Booster Dry Wt (lb)	N/A				(9,750)*
	SRM Booster Prop Wt (lb)	2.76				1,019

System Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities – ($/lb) Dev	Prod	Recur	Total Prog (Free Input Wt)	Total Prog (Costed Input Wt)
Orbiter							0	0
External Tank							0	0
SRM Booster							2,812	12,562
Total							2,812	12,562

*Needed only for costed pound of booster dry weight growth since all solid booster costing is based on propellant weight.

Table 5.6-1

CONFIGURATION E SENSITIVITIES

Effects of One Pound of Orbiter in Preliminary Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivites ($/lb) Nonrecurring Dev	Nonrecurring Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0.452	5,425	1,311	90	6,826
	Plumbing Wt (lb)	0.056	7,420	2,040	1,300	10,760
	TPS Weight (lb)	0.090	3,168	477	2,213	5,858
	Landing Gear Wt (lb)	0.076	3,350	1,230	110	4,690
Tank	Ext Tank Dry Wt	0.498	1,400	0	9,582	10,982
	Ext Tank Prop Wt	10.02	0	0	59	59
Booster	Structure Wt	4.94	2,956	323	285	3,564
	Plumbing Wt	0.311	4,421	931	756	6,108
	TPS Wt	0.064	527	580	440	1,547
	Landing Gear Wt	0.331	2,642	535	452	3,629
	Main Eng Thrust/Eng*	4.30	1,091	284	235	1,610
	ABES Eng Thrust/Eng	0.268	4,347	2,042	1,054	7,443
	Booster Prop Wt	21.88	0	0	59	59

Vehicle Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities ($/lb) Dev	Prod	Recur	Total Prog (Free Input Weight)	Total Prog (Costed Input Weight)
Orbiter	0.852	1.852	2.04	3,407	843	322	4,572	11,398
External Tank	0.498	0.498	10.52	697	0	5,363	6,060	6,060
Booster	6.78	6.78	28.66	22,742	3,868	4,405	31,015	31,015
Total			41.22	26,846	4,711	10,090	41,647	48,473

*Both Orbiter and Booster Engine Thrust changed together

Table 5.6-2

CONFIGURATION E SENSITIVITIES

Effects of One Pound of External Tank in Preliminary Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities ($/lb) Nonrecurring Dev	Nonrecurring Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0.095	5,425	1,311	90	6,826
	Plumbing Wt (lb)	0.035	7,420	2,040	1,300	10,760
	TPS Weight (lb)	0.012	3,168	477	2,213	5,858
	Landing Gear Wt (lb)	0.011	3,350	1,230	110	4,690
Tank	Ext Tank Dry Wt	0.319	1,400	0	9,582	10,982
	Ext Tank Prop Wt	6.41	0	0	59	59
Booster	Structure Wt	3.12	2,956	323	285	3,564
	Plumbing Wt	0.197	4,421	931	756	6,108
	TPS Weight	0.041	527	580	440	1,547
	Landing Gear Wt	0.209	2,642	535	452	3,629
	Main Eng Thrust/Eng*	2.72	1,091	284	235	1,610
	ABES Eng Thrust/Eng	0.167	4,547	2,042	1,054	7,443
	Booster Prop Wt	13.83	-	-	59	59

Vehicle Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities ($/lb) Dev	Prod	Recur	Total Prog (Free Input Weight)	Total Prog (Costed Input Weight)
Orbiter	0.27	0.27	0.29	850	215	82	1,147	1,147
External Tank	0.32	1.32	7.73	447	0	3,881	4,328	15,310
Booster	4.29	4.29	18.12	14,369	2,445	2,784	19,598	19,598
Total			26.14	15,666	2,660	6,747	25,073	36,055

*Both Orbiter and Booster Engine Thrust changed together

Table 5.6-3

CONFIGURATION E SENSITIVITIES

Effects of One Pound of Booster in Preliminary Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities ($/lb) Nonrecurring Dev	Nonrecurring Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0.0175	5,425	1,311	90	6,826
	Plumbing Wt (lb)	0.0065	7,420	2,040	1,300	10,760
	TPS Weight (lb)	0.0023	3,168	477	2,213	5,858
	Landing Gear Wt (lb)	0.0020	3,350	1,230	110	4,690
Tank	Ext Tank Dry Wt	0.0657	1,400	0	9,582	10,982
	Ext Tank Prop Wt	1.32	0	0	59	59
Booster	Structure Wt	0.352	2,956	323	285	3,564
	Plumbing Wt	0.0362	4,421	931	756	6.108
	TPS Weight	0.0074	527	580	440	1,547
	Landing Gear Weight	0.0775	2,642	535	452	3,629
	Main Eng Thrust/Eng*	0.566	1,091	284	235	1,610
	ABES Eng Thrust/Eng	0.0629	4,347	2,042	1,054	7,443
	Booster Prop Wt	1.705	-	-	59	59

Vehicle Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities ($/lb) Dev	Prod	Recur	Total Prog (Free Input Weight)	Total Prog (Costed Input Weight)
Orbiter	0.05	1.05	0.05	157	40	15	212	212
External Tank	0.07	0.07	1.39	92	0	707	799	799
Booster	0.66	1.66	3.36	2,300	482	466	3,248	6,812
Total			4.80	2,549	522	1,188	4,259	7,823

*Both Orbiter and Booster Engine Thrust changed together

Table 5.6-4

CONFIGURATION E SENSITIVITIES

Effects of One Pound of Orbiter in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities ($/lb) Nonrecurring Dev	Nonrecurring Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0.302	5,425	1,311	90	6,826
	Plumbing Wt (lb)	0	7,420	2,040	1,300	10,760
	TPS Weight (lb)	0.070	3,168	477	2,213	5,858
	Landing Gear Wt (lb)	0.059	3,350	1,230	110	4,690
Tank	Ext Tank Dry Wt	2.039	1,400	0	9,582	10,982
	Ext Tank Prop Wt	40.78	0	0	59	59
Booster	Structure Wt	0	2,956	323	285	3,564
	Plumbing Wt	0	4,421	931	756	6,108
	TPS Weight	0	527	580	440	1,547
	Landing Gear Wt	0	2,642	535	452	3,629
	Main Eng Thrust/Eng	0	1,091	284	235	1,610
	ABES Eng Thrust/Eng	0	4,347	2,042	1,054	7,443
	Booster Prop Wt	-2.06*				

Vehicle Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities ($/lb) Dev	Prod	Recur	Total Prog (Free Input Weight)	Total Prog (Costed Input Weight)
Orbiter	0.43	1.43	1.58	2,058	502	188	2,748	9,574
External Tank	2.04	2.04	42.82	2,855	0	21,943	24,798	24,798
Booster	0	0	-2.06*	0	0	0	0	0
Total			42.34	4,913	502	22,131	27,546	34,372

*Reduced JP-4 flyback fuel because of lower staging velocity, cost savings neglected

Table 5.6-5

CONFIGURATION E SENSITIVITIES

Effects of One Pound of External Tank in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities ($/lb) Nonrecurring Dev	Nonrecurring Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0	5,425	1,311	90	6,826
	Plumbing Wt (lb)	0	7,420	2,040	1,300	10,760
	TPS Weight (lb)	0	3,168	477	2,213	5,858
	Landing Gear Wt (lb)	0	3,350	1,230	110	4,690
Tank	Ext Tank Dry Wt	1.281	1,400	0	9,582	10,982
	Ext Tank Prop Wt	25.60	0	0	59	59
Booster	Structure Wt	0	2,956	323	285	3,564
	Plumbing Wt	0	4,421	931	756	6,108
	TPS Weight	0	527	580	440	1,547
	Landing Gear Wt	0	2,642	535	452	3,629
	Main Eng Thrust/Eng	0	1,091	284	235	1,610
	ABES Eng Thrust/Eng	0	4,347	2,042	1,054	7,443
	Booster Prop Wt	-1.29*				

Vehicle Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Snesitivities ($/lb) Dev	Prod	Recur	Total Prog (Free Input Weight)	Total Prog (Costed Input Weight)
Orbiter	0	0	0	0	0	0	0	0
External Tank	1.28	2.28	27.88	1,793	0	13,785	15,578	26,560
Booster	0	0	-1.29*	0	0	0	0	0
Total			26.59	1,993	0	13,785	15,578	26,560

*Reduced JP-4 flyback fuel because of decreased staging velocity, cost savings neglected

Table 5.6-6

CONFIGURATION E SENSITIVITIES

Effects of One Pound of Booster in Detailed Design Phase for 500 Flight Program

	Cost Driver	Performance Sensitivities (lb/lb)	Direct Cost Sensitivities ($/lb) Nonrecurring Dev	Nonrecurring Prod	Recur	Total Prog
Orbiter	Structure Wt (lb)	0	5,425	1,311	90	6,826
	Plumbing Wt (lb)	0	7,420	2,040	1,300	10,760
	TPS Weight (lb)	0	3,168	477	2,213	5,858
	Landing Gear Wt (lb)	0	3,350	1,230	110	4,690
Tank	Ext Tank Dry Wt	0.148	1,400	0	9,582	10,982
	Ext Tank Prop Wt	2.951	0	0	59	59
Booster	Structure Wt	0.325	2,956	323	285	3,564
	Plumbing Wt	0	4,421	931	756	6,108
	TPS Weight	0	527	580	440	1,547
	Landing Gear Wt	0.0727	2,642	535	452	3,629
	Main Eng Thrust/Eng	0	1,091	284	235	1,610
	ABES Eng Thrust/Eng	0.054	4,347	2,042	1,054	7,443
	Booster Prop Wt	-0.10*				

Vehicle Element	Indir Dry Wt	Total Dry Wt	Total Wet Wt	System Cost Sensitivities ($/lb) Dev	Prod	Recur	Total Prog (Free Input Weight)	Total Prog (Costed Input Weight)
Orbiter	0	0	0	0	0	0	0	0
External Tank	0.15	0.15	3.10	207	0	1,592	1,799	1,799
Booster	0.49	1.49	1.38	1,388	254	182	1,824	5,388
Total			4.48	1,595	254	1,774	3,623	7,187

*Reduced JP-4 flyback fuel because of decreased staging velocity, cost savings neglected.