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SPACE SHUTTLE SYSTEM PROGRAM DEFINITION

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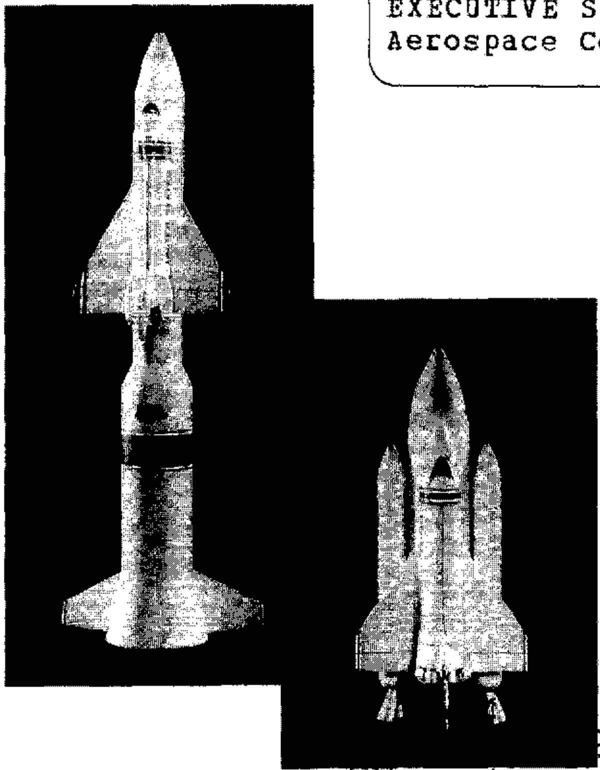

PHASE B EXTENSION
FINAL REPORT

EXECUTIVE SUMMARY

Volume I

CONTRACT: NAS 9-11160
MODIFICATIONS NO. 11S
B35-43 RP-33
15 March 1972

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SPACE SHUTTLE SYSTEM PROGRAM DEFINITION

PHASE B EXTENSION FINAL REPORT

EXECUTIVE SUMMARY

Volume I

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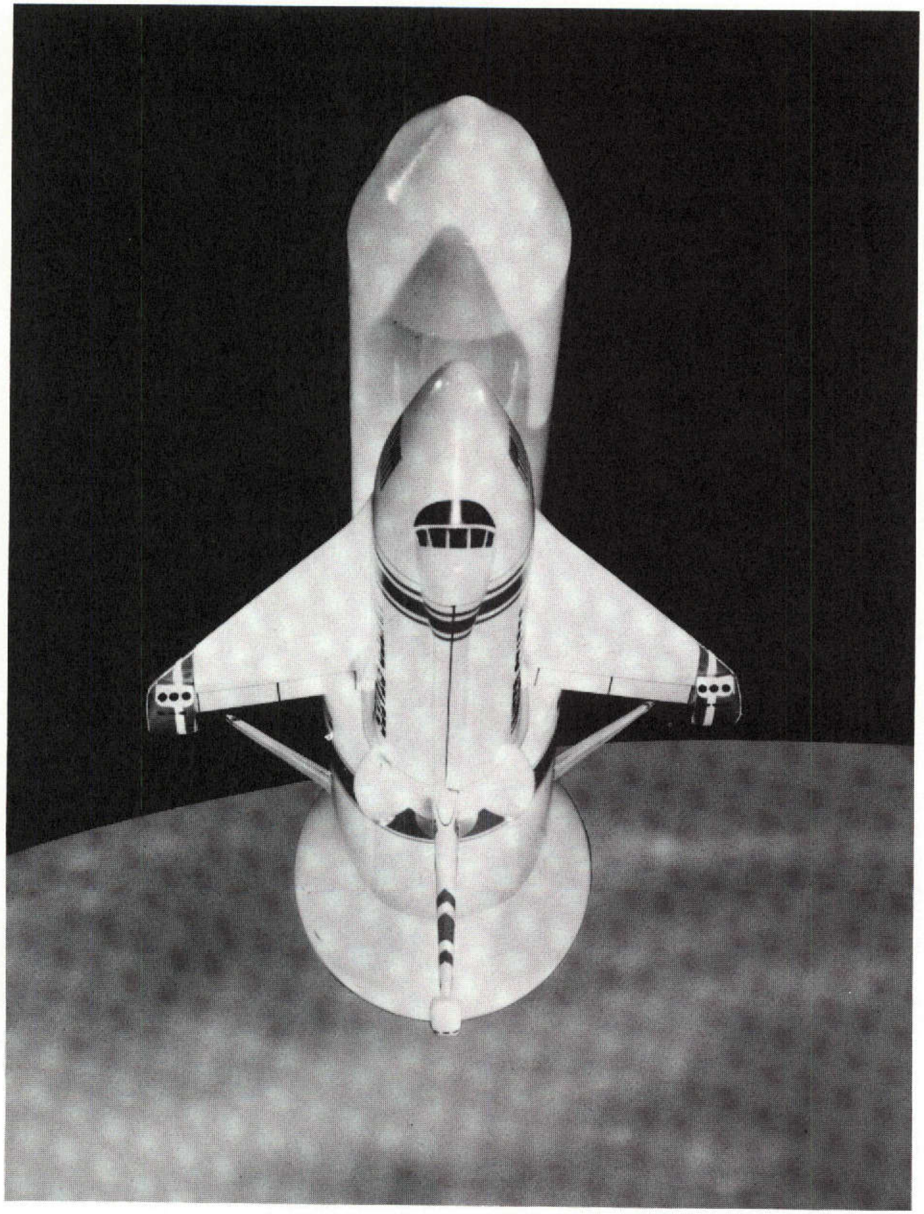

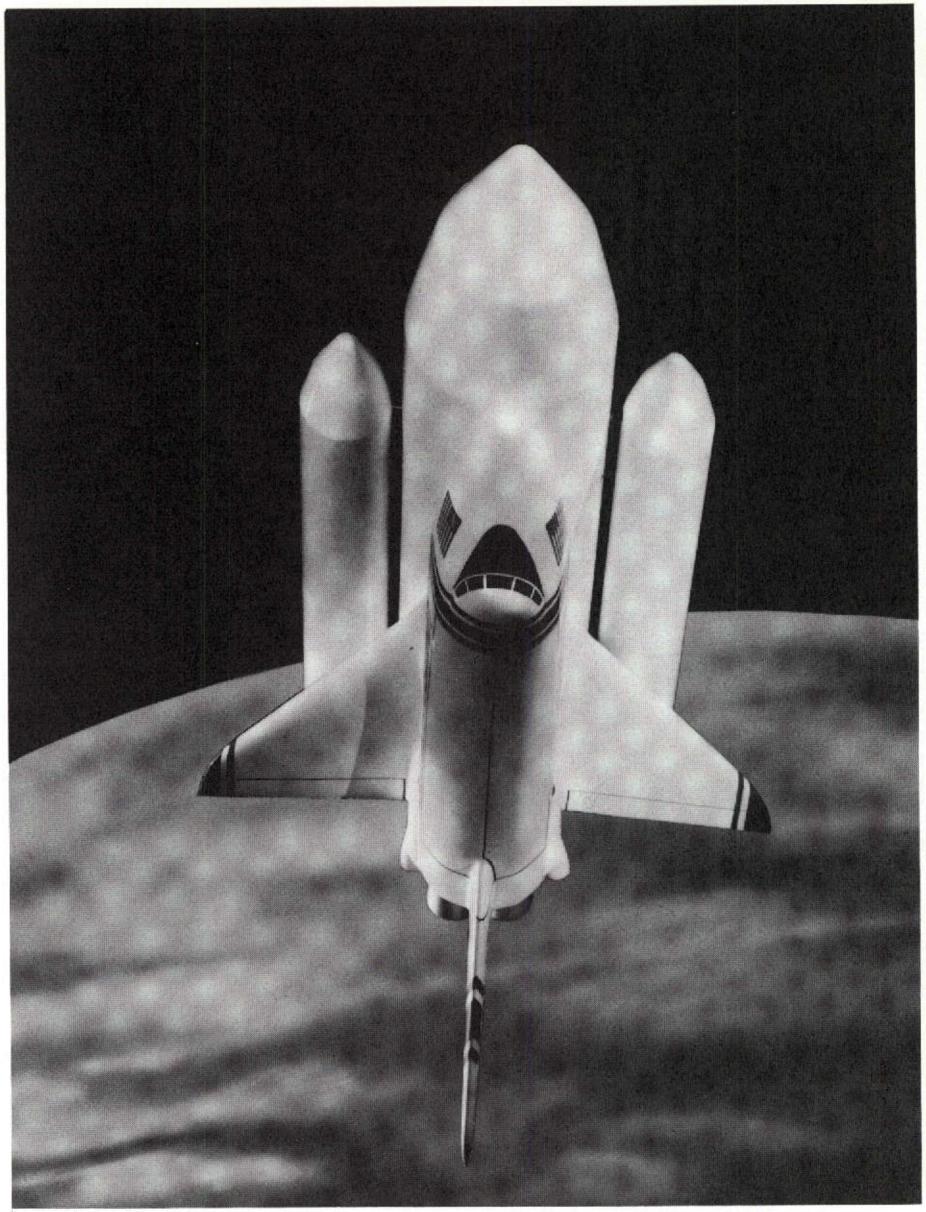

FOREWORD

This document is one of a series prepared in accordance with Contract NAS9-11160, Modification No. 11S, Space Shuttle System Program Definition Study, Phase B Extension Contract. Listed below are the documents required by Data Line Items 6, 8 and 9 of the contract.

CONTRACT DOCUMENTS

Volume I – Executive Summary
Volume II – Technical Report
Volume IIa – Payload Impact Analysis On Orbiter Subsystems (NASA TD GAC-11)
Volume III – Mass Properties Report
Volume IV – Cost and Schedule Report

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STUDY PARTICIPATION

The Space Shuttle System Program Definition Study, Phase B Extension Contract was conducted jointly by the Grumman Aerospace Corporation, the Boeing Company, and their associates. To assure consistency of results, Grumman and Boeing worked together closely, particularly in the areas of system design, costing, and concept/configuration evaluation. In addition to the overall study management, Grumman concentrated on Orbiter and HO Tank design, analysis of mated configuration, and development test planning. Boeing concentrated on Booster design, ground operations, and maintenance planning.

In concurrence with NASA's encouragement for the inclusion of Shuttle applicable technologies from the international industrial community, Grumman continued its working relationship with two major European aerospace firms, Dassault of France and Dornier of W. Germany.

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CONTENTS

ILLUSTRATIONS

ILLUSTRATIONS (Cont'd)

INTRODUCTION AND SUMMARY

This Phase B Extension Study final report is submitted to NASA to aid in the selection of a low cost shuttle system for design and development. The objective of the final study period was to examine and penetrate the major technical and cost issues affecting the choice of:

- Liquid propulsion recoverable or solid propulsion expendable booster
- Parallel burn/parallel mount or series burn/tandem mount configurations and
- Payload weight and payload bay size of the orbiter

In accordance with NASA direction (MSC telegraphic message LV-10482-7, dated 7 January 1972) Grumman baselined a series burn system comprised of an orbiter with a standard (15x60) payload bay and a recoverable liquid propulsion pressure-fed booster, and used this system as the standard to which we compared the cost and performance of all other configurations studied. The complete matrix of program options evaluated in this final study period is shown in Figure 1, with the above described baseline in heavy outline on the left of the figure. Of the alternates to the baseline, the major study emphasis was placed on a parallel burn configuration employing solid rocket boosters (both 120" and 156" solids were considered) and an orbiter resized for a 14x45 payload bay. Considerable effort was also devoted to a series configuration utilizing a pump-fed booster and parallel/SRM configuration with a standard payload bay size orbiter. A variation of the baseline system, incorporating swing engines rather than fixed engines on the orbiter, was considered primarily in conjunction with a study of pad abort capability and will be discussed in the section dealing with that subject.

NASA direction also modified several of the system requirements and characteristics used during previous study periods. The requirements baseline for the final study period is compared to the December 15 set of requirements and characteristics in Figure 2. Of particular interest are:

- Elimination of the phase development (Mark I/Mark II) concept
- Introduction of a 14x45 payload bay orbiter to deliver 45K due east payload up, 25K down
- Specification of three space shuttle main engines (SSME's) at 472K vacuum thrust each for the orbiter
- Introduction of reusable space insulation (RSI) on the first production orbiter

Our approach to the study and evaluation of the 16 configurations comprising the base of the pyramid of Figure 1 was as follows:

- In order to assess the relative merits of liquid propellant/recoverable versus solid propellant/expendable boosters, the technical issues affecting this appraisal were explored using the 15x60 payload bay orbiter in conjunction with (a) a liquid propellant, pressure-fed booster in a series burn, tandem arrangement and (b) 156" and 120" solid rocket boosters in parallel burn, parallel mounting arrangements

- The effect of payload weight and payload bay size reduction was examined in a three-step process:

  - We first designed a small payload bay orbiter with aerodynamic performance characteristics comparable to our baseline 15 x60 payload bay orbiter

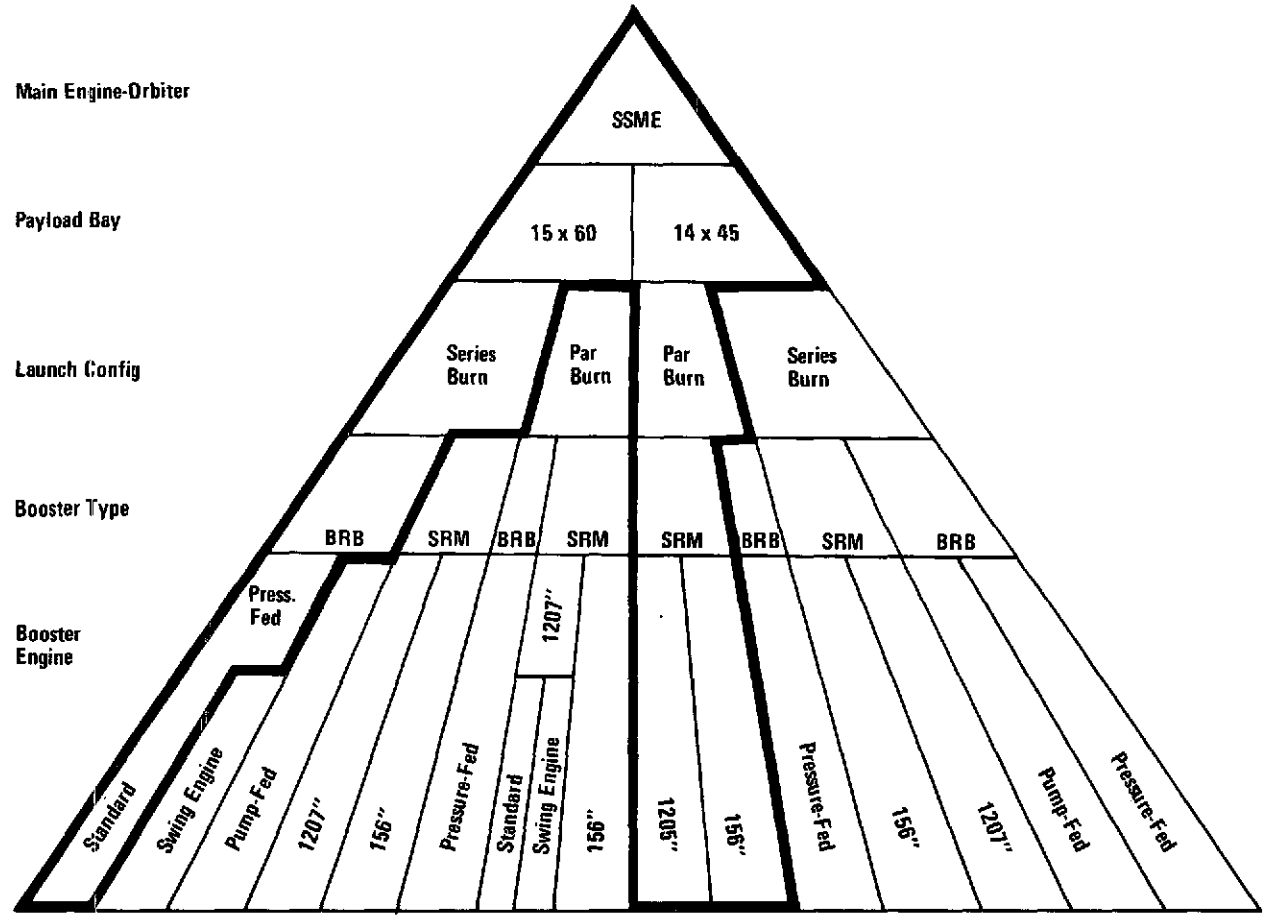


Figure 1 Program Options

- We then developed parallel burn/SRM (solid rocket motor) stacks using the above core orbiter and 156" and 120" motors and
- Then compared both the orbiter as well as the full-up stacks to their baseline counterparts in weight and cost

	WAS		IS NOW	
	Mk I	Mk II		
Orbiter Payload	15 x 60	15 x 60	15 x 60	14 x 45
Payload Up-East/Polar/55^{0}	?/25/?	65/40/?	65K/40K/25K	45K/?/25K
Payload Down	25K	40K	40K	25K
V_{Stage}, fps	6000 ± 1000	6000 ± 1000	>4000	>4000
Main Engine Type/T_{vac}	J-2S/265K	SSME/TBD	SSME/472K	SSME/472K
TPS	Ablative	RSI	RSI	RSI
Avionics	Low Cost	Upgraded	Low Cost/ Evolutionary	Low Cost/ Evolutionary
OMS/RCS	Storable	Storable	Storable	Storable
OMS ΔV, fps	650/1000	650/1000	650/1000/ 1400	650/1000/ 1400
Cross Range, N Mi	1100	1100	1100	1100
Abort	Intact (Not Pad)	Intact (Not Pad)	Intact-All Phases	Intact-All Phases

Figure 2 System Characteristics

Other major issues studied were the:

- Relative merits of pressure-fed versus pump-fed liquid propellant boosters
- Impact of providing pad abort capability and
- Effect on the environment as a disciminator between the various configurations studied

The Study Key Issues are summarized in Figure 3.

STUDY KEY ISSUES

- What Are Technical & Cost Differences Between Series/BRB & Parallel SRM?
  - How Much Weight & Cost Reduction for Smaller-Payload-Bay-Size Orbiter?
    - What Is Booster Design & Cost Status?
      - What Is Orbiter Design Status?
        - How Can We Achieve Pad Abort Capability?
          - What Are Implications of National Environmental Policy Act On Shuttle?

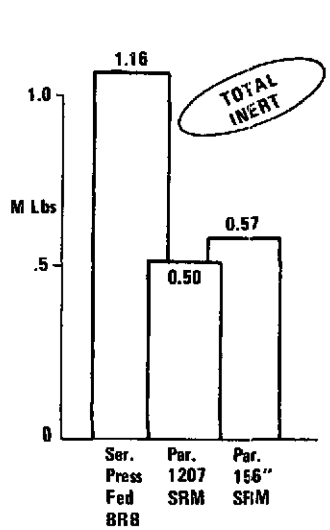
Figure 3 Study Key Issues

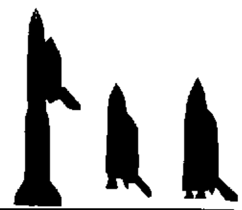
In summary, these were the conclusions of our studies during this final contract period:

- The comparison of a series/BRB (Ballistic Recoverable Booster) to a parallel/SRM system, using the same orbiter in each case, reaffirmed what we had learned during previous study phases - namely, that solid propellant boosters significantly lower DDT&E costs, since they are more efficient than liquids and require less development, but increase cost per flight, since they are expendable. The weight comparison is shown in Figure 4, the cost comparison in Figure 5. Note that the SRM versions of the shuttle have about half the weight of total inerts of BRB systems and nearly 2M lb lower GLOW (gross liftoff weight). The corresponding DDT&E cost decrease for SRM configurations is nearly one billion dollars, but the "unamortized" cost per flight is about twice as high for the expendable SRM's than liquid propellant recoverable boosters.

  As far as other technical issues affecting the selection of series/BRB or parallel/SRM systems are concerned, we found that:

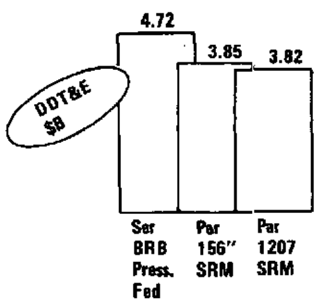
  - The HO tank is more efficient (has a lower structural fraction) but generally heavier for a parallel configuration, thus contributing to the higher cost/flight

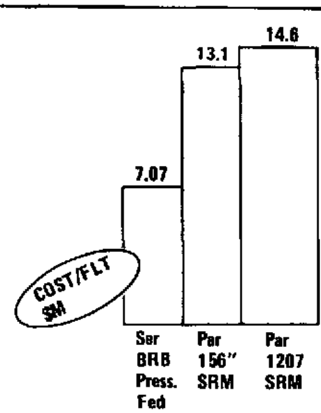




	15 x 60		
	Ser BRB	Par 120"	Par 156"
GLOW, M Lb	6.396	4.580	4.553
BLOW, M Lb	5.119	2.788	2.854
OLOW, M Lb	1.277	1.792	1.699
Inert, M Lb	1.158	0.499	0.568
V_{Stage}, fps	4879	5067	5892
Payload Margin, Lb	7500	10,700	7600

Figure 4 Launch Configurations Characteristic Comparison





PAF, $B	1.12	.90	.88
Total Prog $B	9.35	10.46	11.06

Figure 5 15 x 60 Orbiter Series BRB vs Parallel SRM

- Although use of orbiter engines and aero surfaces permit ascent control of the parallel stack, the control authority is so marginal that we decided to assume booster TVC in all sizing and costing. Booster TVC can be eliminated by canting the booster nozzle and providing fins on the tank
- The thermal and acoustic environment induced on the parallel configuration is more severe than that for a series stack, imposing about 1500 lb orbiter and 1000 lb HO tank weight penalty
- The inflight abort capability of the series and parallel systems are essentially equivalent if a high thrust (10,000 lb) OMS engine is used
- The parallel/SRM configuration is somewhat more sensitive to environmental issues because of the HCl generated during booster burn.

• On the issue of orbiter payload bay size and payload weight, we found that:

- A 14x45 payload bay orbiter could not meet all performance requirements without either changing engine thrust (from 3 x 472 K to 3 x 380 K) or increasing the payload bay by a minimum of five feet beyond the specified 45 ft length
- The orbiter dry weight decreased by about 8000 to 16,000 lb depending upon which of the above two options were exercised
- The GLOW's of the small payload bay orbiter configurations were about 2.5M lb lower and the total inerts about half that of the baseline series/BRB stack. DDT&E was approximately one billion dollars lower and cost per flight approximately double that of the baseline system
- Relative to a parallel/SRM stack using the 15x60 payload bay orbiter, the overall DDT&E reduction of the small orbiter configuration was about $43M, 70% of which was attributable to payload weight, rather than payload bay size, reduction

A summary configuration comparison chart is presented in Figure 6. A checkmark designates the configuration which performs best relative to the evaluation parameter to which it applies.

- A comparative evaluation of pressure-fed versus pump-fed boosters led to the conclusion that the pump-fed booster would present a lower development risk, and would cost less to develop (by about $500M) than the pressure-fed
- We studied the feasibility of providing pad abort capability and the performance and cost impact of implementing it. We concluded that:
  - Successful pad abort from the worst case situation (tank or booster explosion) requires 5 to 7 sec warning time, depending upon the axial loading and blastwave overpressure we can tolerate
  - A swing engine orbiter configuration (the orbiter engines located on the HO tank during thrusting and then transferred to the orbiter) offers significant advantages for pad abort
  - About $250M DDT&E and about $300K per flight would be the cost of providing pad abort capability

Our overall conclusions developed during this final study period can be summarized as follows:

- On the subject of solid versus liquid propellant boosters
  - SRM's have lower DDT&E but significantly higher cost per flight than liquid boosters
  - All SRM applications make the program potentially more sensitive to environmental issues (primarily because of HCl pollution).

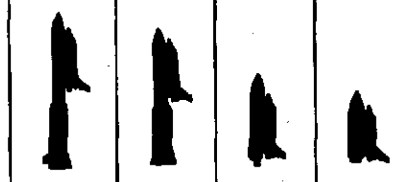
	Ser/BRB 15 x 60 Orbiter		Par/SRM 15 x 60 Orb.	Par.SRM 14 x 45 Orb.
	Press. Fed	Pump Fed	156"	1205
				
Lowest Total Program Cost, $B	9.35 ☐	8.66 ☑	10.46 ☐	10.39 ☐
Lowest DDT&E, $B	4.72 ☐	4.23 ☐	3.85 ☑	3.79 ☑
Lowest Cost/Flight, $M	7.07 ☑	6.62 ☑	13.1 ☐	13.1 ☐
Least Complex Design • Least Acoustic Impact	☑	☑	☐	☐
• Easiest Ascent Control	☑	☑	☐	☐
Least Environmental Impact	☑	☑	HCL ☐	☐

Figure 6 Configuration Comparison Summary

Recommendation

Since cost per flight, in our view, is vital to the future of the shuttle program, we recommend the recoverable liquid booster in a series configuration for shuttle development. Of the liquid boosters, we prefer the pump-fed system because of its better performance, lower risk and lower cost.

- On the subject of payload bay size on the orbiter:

– The cost benefit of the smaller orbiter derives primarily from payload weight rather than payload bay size reduction

– Small payload bays make orbiter balance difficult. Bay must be lengthened or engine thrust reduced

Recommendation

If we must minimize DDT&E, reduce payload weight requirement first, but hold on to the 60 ft bay.

- On the subject of pad abort:

  – The capability can be achieved but, as on previous programs, will compound the design effort
  – Increases development and per flight costs

Recommendation

Let's make sure we understand all implications before we proceed to implement. Swing engine arrangement should be seriously considered if pad abort is a requirement.

STUDY OBJECTIVES

During the first half of this second four-month study extension, we concentrated on the generic technical and cost issues of series vs parallel burn shuttle systems, with primary emphasis on configurations employing ballistic recoverable liquid propellant boosters. Solid rocket motors were treated as backup to the liquids and were sized to operate with the same orbiter/HO tank combinations that resulted from optimization with BRB's. Our conclusions at the mid-point of the present study period were that:

- The series/BRB system was technically simpler than a parallel system but that:
- The parallel/BRB configuration provided a backoff potential to solid propellant boosters at lower cost per flight

For the second half of the study, the objectives were somewhat different. Primary emphasis was still to be placed on the series/BRB system, but NASA now desired to consider the SRM's as candidate shuttle boosters in their own right rather than as a backup to liquids. Major interest centered on the use of solids in conjunction with parallel burn stacks. It was also required that the effect of orbiter payload bay size and payload weight reduction on such a parallel/SRM configuration be evaluated.

Major study emphasis was to be placed on the following areas of technical and programmatic concern:

- Parallel burn technical issues (control, inflight abort, induced environment)
- Capability for pad abort
- Solid booster technical problems
- Liquid booster recovery technology
- Program costs

Study effort was also to be devoted to a number of other configurations (see Figure 1 for the complete matrix of study configurations), but, in general, at a somewhat lower level of emphasis than the "baseline" series/BRB and the parallel/SRM systems.

Our analysis of NASA's objectives and concerns resulted in the formulation of the set of "key issues" of Figure 3, and of a subset of technical and programmatic questions for each key issue, the answers to which would provide the data for arriving at a conclusion and recommendation of the particular "key issue".

In the sections to follow, we will treat each key issue in terms of the sub-issues or questions relevant to it, present our conclusions for each sub-issue and then summarize these as the basis for our overall conclusion and recommendation on the "key issue".

WHAT ARE THE PHYSICAL CONFIGURATION CHARACTERISTICS?

In selecting the specific design points for all configurations studied, we used the approach of choosing that booster staging velocity which yielded 5% potential margin on orbiter inert weight. We define potential margin (or payload margin) as that amount of inert weight increase in the orbiter (or payload) which can be accommodated by simply expanding the HO tank while leaving all other elements of the system unchanged. This margin is over and above the 10%-2%-10% growth allowance built into the orbiter/tank/booster design.

The design point selection based on potential margin is only applicable if the booster can be "rubberized", i.e., sized for any given orbiter/tank weight. In the case of 120" solids, the maximum total impulse is a fixed quantity and we are forced to accept whatever tank size falls out when we tailor the SRM thrust profile to meet max q and max g constraints. When we cite a potential payload margin for a 120" SRM configuration, it must be realized that the accommodation of such an increase in inert weight involves not only resizing the tank but also retailoring the SRM thrust profile.

Figure 7 shows the result of applying the 5% potential payload margin concept to the selection of the design point for both the baseline series/BRB as well as the parallel/156" SRM configurations. The payload margin is zero at or near the GLOW bucket and increases as we move towards the higher staging velocities. At the 7.5K lb margin point (5% of the approximately 150K orbiter inert weight), we pay about 300K lb and 200K lb GLOW penalty for the series/BRB and parallel/SRM cases, respectively. Figure 8 shows the cost penalty for this departure from the optimum weight design point (about $8M DDT&E) and indicates that this is a relatively low cost

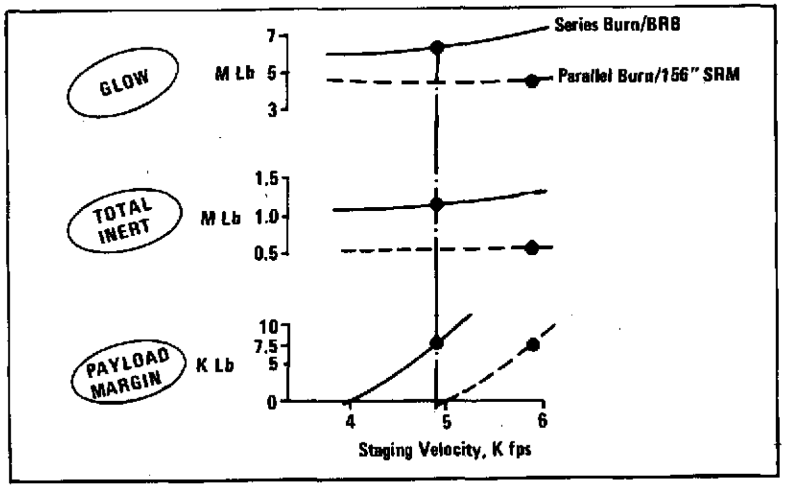


Figure 7 15 x 60 Orbiter Launch Configuration Performance

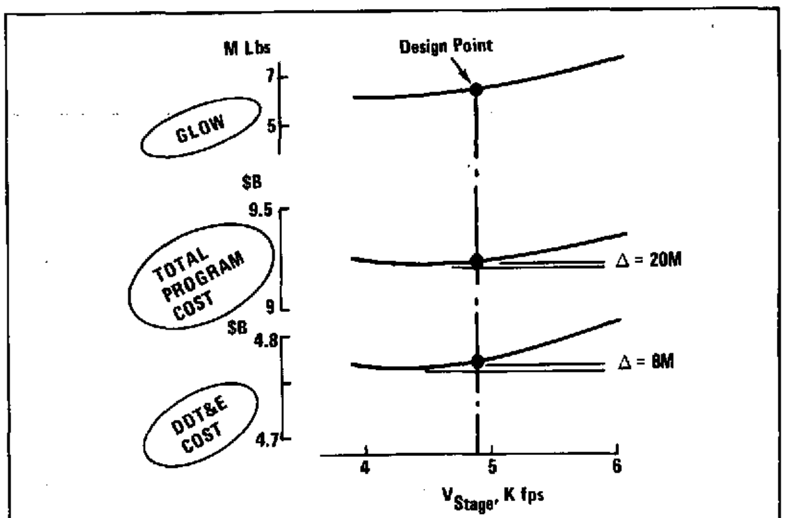


Figure 8 Cost of Payload Margin – Series BRB

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method of providing insurance against unexpected weight growth. The delta's shown represent, however, only the costs associated with the initial sizing for a non-optimum staging velocity, and do not reflect the additional costs resulting from redesigning the tank if the potential weight increase actually occurs.

We have made an extensive study of the cost implications of providing an inherent allowance for growth versus margin for a potential growth which may or may not realize, but a discussion of these results is not possible within the limitations of this summary. Our general conclusion is that a judicious mix of growth allowance and margin is the best method of achieving payload assurance, and that the extent of total contingency provided and the percentages thereof to be allocated to allowance and margin are a function of the weight confidence level which we wish to impose.

The launch configurations corresponding to the design points selected from the trending data are shown in Figures 9, 10 and 11. A comparison of their major characteristics is presented in Figure 12. Typically, the greater structural efficiency of the solid propellant boosters results in the parallel/SRM configuration exhibiting a decrease in GLOW of about 2.0M lb relative to the series/BRB case. Of greater interest, as being a stronger cost driver, is the fact that the total inert weight of the former is less than half that of the baseline. The more efficient SRM also tends to drive the staging velocity of the parallel/156" SRM stack to near 6000 fps, which is typically about 1000 ft higher than that of the series system. This does tend to penalize the parallel configuration in cost per flight, since generally the minimum in the launch cost trends occur between 4000 fps and 4500 fps staging velocity.

WHAT IS THE DIFFERENCE BETWEEN SERIES AND PARALLEL HO TANKS?

One of the major reasons for the GLOW difference between parallel and series configurations is the greater structural efficiency of the parallel HO tank. This efficiency is most readily quantified in terms of pounds of

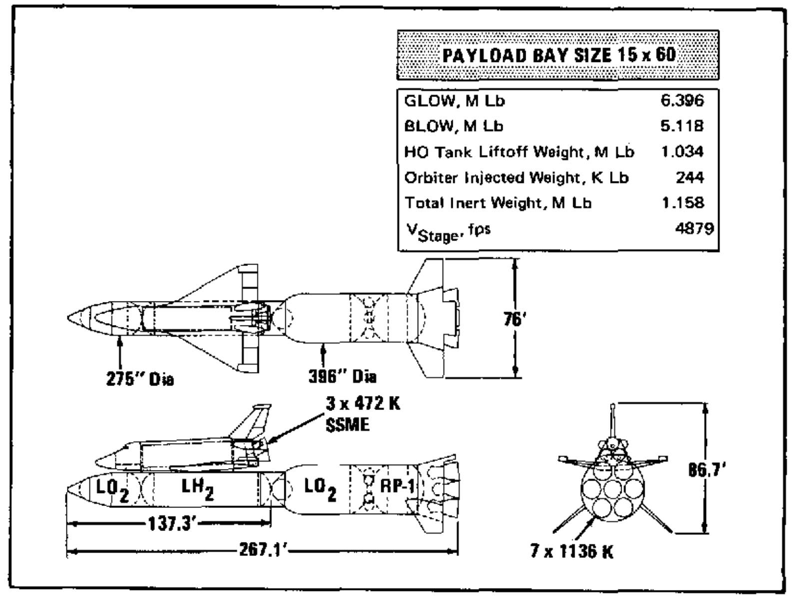


Figure 9 Launch Configuration – Series BRB

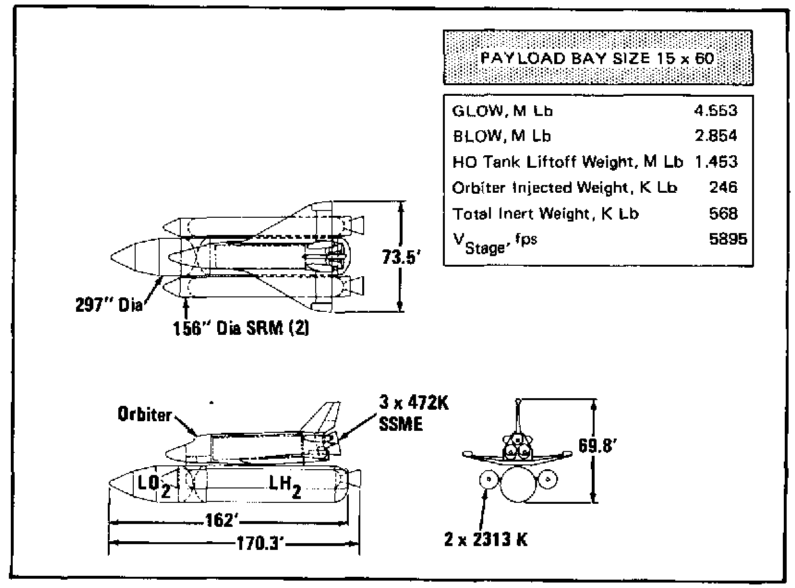


Figure 10 Launch Configuration – Parallel 156" SRM

dry tank weight per pound of loaded propellant. We designate this ratio the "structural fraction" (SF) of the tank, with a lower SF indicating a more efficient structure. As shown in Figure 13 the parallel tanks generally exhibit a lower SF (or higher propellant fraction (PF) which is the number of pounds of propellant per pound of total loaded tank weight) than series tanks. The reason for the higher efficiency of the parallel tanks becomes apparent if the tank design criteria and loading conditions for the series and parallel stack are compared. In order to clearly demonstrate the weight differences resulting from these loading conditions, we have taken a series tank at the design point propellant loading of one of our study configurations (14x45 payload bay orbiter/BRB) and compared its weight to a parallel tank designed for the same amount of propellant.

The design criteria for the two tanks are shown in Figure 14. Comparable elements of these tanks are, of course, designed by the same loading conditions, but the actual loads are quite different in the two cases. Figure 15 shows the results of an analysis of applied axial loads and bending

PAYLOAD BAY SIZE 15 x 60	
GLOW, M Lb	4.580
BLOW, M Lb	2.788
HO Tank Liftoff Weight, M Lb	1.546
Orbiter Injected Weight, K Lb	246
Total Inert Weight, K Lb	598
V_{Stage}, fps	5067

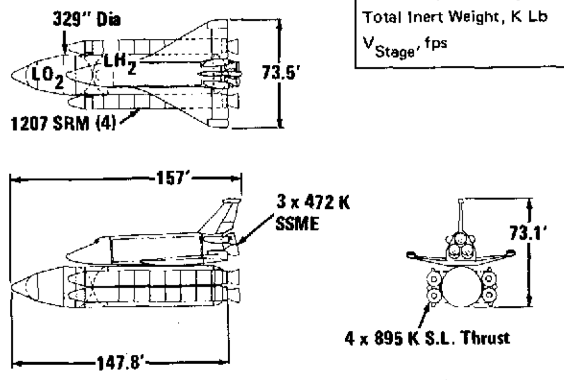
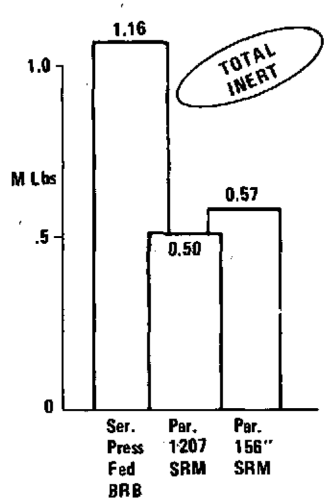
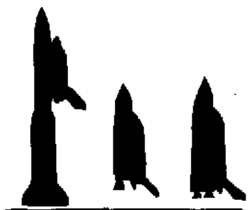


Figure 11 Launch Configuration – Parallel 1207





	15 x 60		
	Ser BRB	Par 120"	Par 156"
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Inert, M Lb	1.158	0.499	0.568
V_{Stage}, fps	4879	5067	5892
Payload Margin, Lb	7500	10,700	7500

Figure 12 Launch Configurations Characteristic Comparison

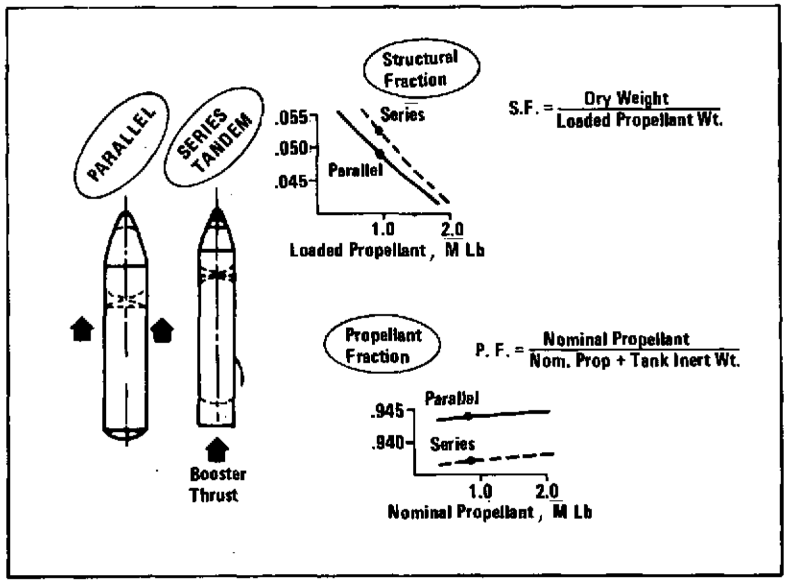

Figure 13 HO Tank Mass Fractions

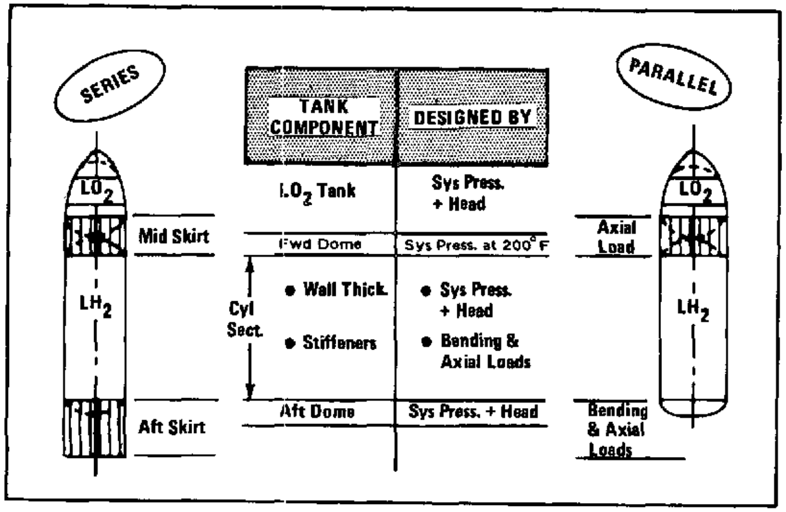

Figure 14 HO Tank Structural Design Criteria

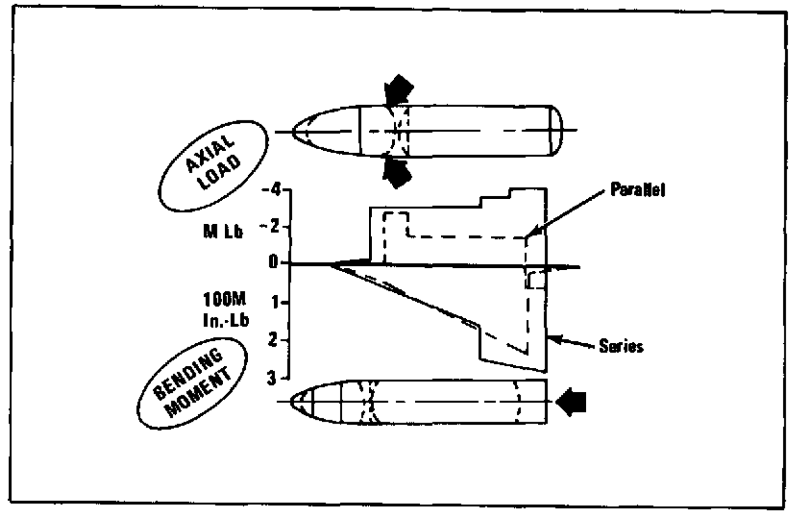

Figure 15 HO Tank Design Limit Loads Envelope

moments for the case of a series and parallel tank. These loads partially design the cylindrical sections and aft domes of the LH_2 tank (see Figure 13).

The series tank experiences significantly higher axial loads over most of the length of LH_2 tank and higher bending moments at the aft section of that tank than does the parallel one. The higher axial loads result from the difference in the manner in which booster thrust loads (indicated by the arrows in Figure 15) are applied to the HO tank. In the series stack, booster thrust is transmitted to the tank via the aft tank skirt, thus applying axial compression loads over the entire length of LH_2 tank and over part of the LO_2 tank up to the forward tank frame where the major load carrying orbiter attachment structure is located. By contrast, the parallel booster thrust loads are carried into the tank at the forward intertank area, so that the only axial loads seen by the parallel LH_2 tank are the orbiter thrust loads transmitted by the aft orbiter/tank attachment structure. Bending moments experienced by the series tank are higher in the aft section, since it is a cantilevered, end supported structure, while the parallel tank is forward and aft supported via the attachment structure to the booster.

Figure 16 summarizes the design conditions for the two tanks and Figure 17 shows the weight differential resulting therefrom. The LO_2 and

Tank Component	Designed By	Design Condition: Series	Design Condition: Parallel
LO_2 Tank	Sys Press. + Head	500°F at 24 psi 3.25g Axial Load, Full Tank	500°F at 24 psi 1.25g at Liftoff 3.2g at 70% Full Tank
Mid Skirt	Axial Load, Primarily	3.25g Axial Load	3.2g + Axial Load
LH_2 Tank	Sys Press.		
Wall Fwd Sect		200°F at 36 psi	200°F at 36 psi
Wall Mid Sect		3.25g at Bstr BO	1.25g at Liftoff
Wall Aft Sect	+ Head	3.25g at Bstr BO	3.0g + at Bstr BO
LH_2 Tank Stiffeners	Bending & Axial Loads	Full-Length Cantilever, Bstr BO: $N_z = 0.05 \pm 0.1$ $N_y = \pm 0.1$ $N_x = 3.25$	Two-Point Support: $N_z = -0.14$, $N_x = 3.0+$ } Max Accel $N_z = -0.27$, } After Sep

Figure 16 HO Tank Structural Design Conditions

BASED ON PROPELLANT WEIGHT OF 925,450 Lb (Nominal)

	Series	Parallel	Δfrom Series
LO_2 Tank	9274	7336	−1938
Mid Skirt	4807	3641	−1166
LH_2 Tank	22,320	20,718	−1602
Miscellaneous	620	820	+200
Aft Skirt (Series)	1218		−1218
Booster Attach (Par)		314	+314
Nose Cone	732	732	−
TPS/Insulation	3997	4937	+940
Systems	7200	6645	−555
Dry Weight	50,168	45,143	−5025
Dry Weight / Loaded Propellant	.05351	.04842	

Figure 17 Parallel – Series Comparison

part of the LH_2 tank weight decrease of the parallel tank comes from the lower pressure head seen by these tanks at the high g levels (near booster burnout) and the decrease in tank wall thickness that it allows. The reason for the pressure head being lower is that, in a parallel burn configuration, the HO tank is being depleted during booster burn so that, at staging, the tanks are only about 70% full and the static pressure head due to liquid column height is commensurately lower.

Comparing the actual tanks for the configurations of specific interest in this section, we find that the series/BRB tank dry weight is 52K lb and that of the parallel/SRM (120") tank is 66K lb. However, although the parallel tank is significantly heavier, its SF is 0.0445 as compared to 0.0525 for the series tank. This is typically the case whenever we compare parallel to series configurations. Since the parallel burn orbiter engines fire during the entire ascent-to-orbit flight, the HO tank must carry more propellant and thus becomes heavier than for a comparable series configuration - but, although heavier, it is more efficient.

WHAT IS THE SSME EPL FOR NO ABORT GAP?

In this section we will consider the abort capability of the parallel series configurations relative to all in flight abort regimes. We will, at this point, exclude pad abort capability considerations, since those are treated in some detail in a later section. The abort regimes discussed herein are illustrated in Figure 18. Of major concern is the ability of the configuration to avoid having to use an alternate site when aborting during ascent flight. This alternate landing site requirement arises from the inability to either abort back to the launch site or abort to orbit as a result of a failure, primarily that of an orbiter engine, occurring during the ascent thrust phase. The time period during ascent flight in which a failure of the orbiter engine

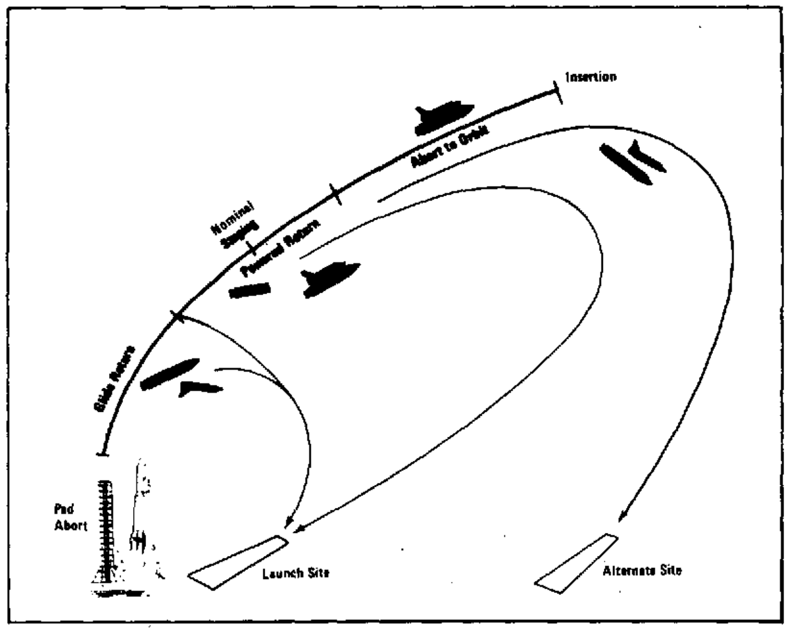

Figure 18 Abort Regimes

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requires landing at a site other than the original launch site is designated as the "abort gap". It is important to minimize or eliminate this abort gap since alternate landing sites are either not available or, if there are possibilities for landing at such alternate sites, the problem of ferrying the orbiter back to the launch site may become exceedingly complex.

Since the failure of an orbiter engine can be partially compensated for by increasing the thrust level of the remaining orbiter engines, (going, in other words, to the so-called "emergency power level" (EPL)), the extent to which such EPL capability is available on the SSME's, or the extent to which it is required to eliminate the abort gap is of considerable interest.

Our studies have shown that it is possible to close this abort gap for both configurations and all missons by applying various techniques such as increasing the flight performance reserves (FPR), or increasing the thrust level of the OMS engines. This latter approach has the dual benefit of increasing the rate at which OMS propellants are being depleted, as well as increasing the thust level itself, both of which increase orbiter thrust to weight and improve abort performance. As shown in Figure 19, a zero abort gap at zero or very small main engine EPL can be achieved for both configurations on both due-east and south-polar missions by the use of the 9700 lb thrust LM descent engine for OMS.

DO LAUNCH ACOUSTICS AND INTERFERENCE HEATING PENALIZE PARALLEL SRM?

Some of the features characteristic to a parallel burn configuration – namely, the simultaneous firing at all engines at liftoff and the close conjunction of these engines to the base of the HO tank – tend to induce thermal and acoustic environments on that configuration which are more severe than what would be experienced on a series burn stack. This is illustrated on Figure 20, in which the launch acoustic levels at liftoff for both configurations are indicated by the straight dash lines. Note that, in general, the parallel burn configuration experiences eight to nine db higher liftoff acoustic levels at the aft end of the configuration than does the series

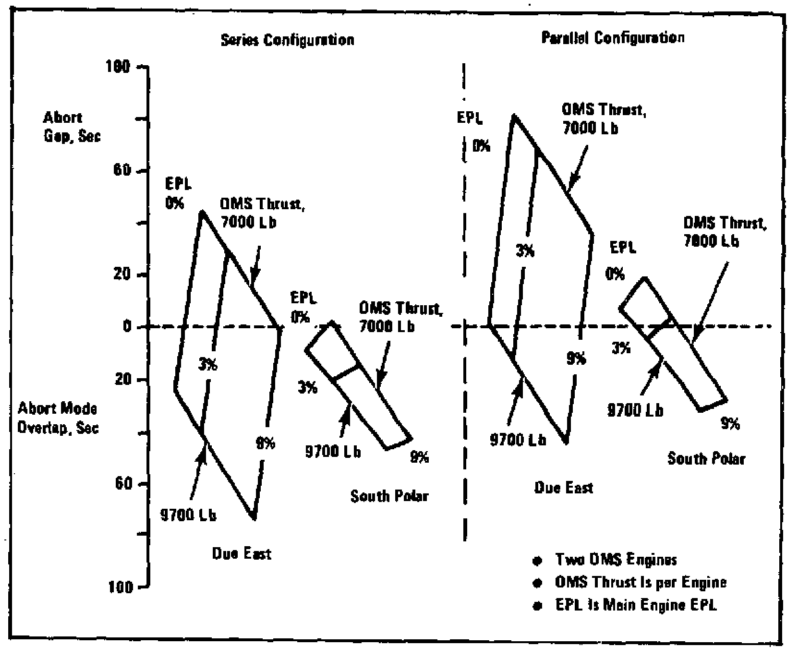

Figure 19 Abort Gap

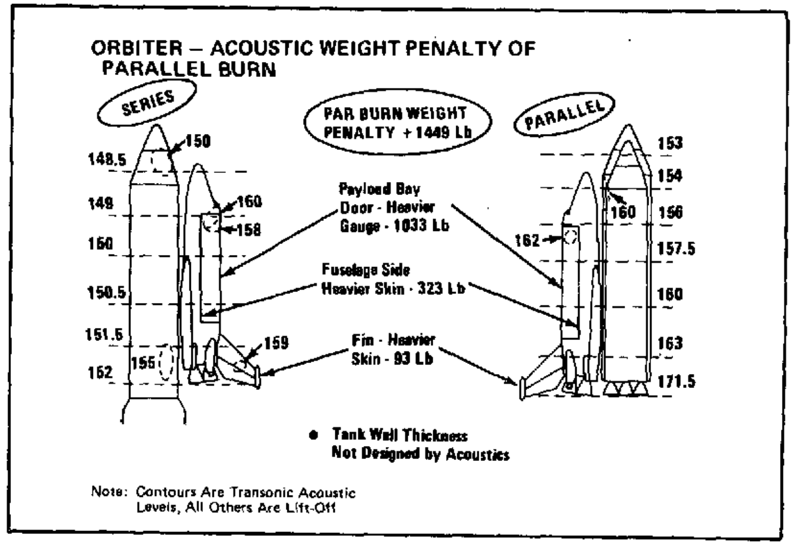

Figure 20 Orbiter – Acoustic Weight Penalty of Parallel Burn

stack. This results from the simultaneous firing of booster and orbiter engines and from the amplification of the ground reflection wave of the pad at or near liftoff. During transonic flight, there are localized areas on both configurations which experience higher vibration or acoustic levels than would normally be expected. These are indicated by the closed contours shown on Figure 20. The weight penalties imposed on the parallel burn configuration by these higher acoustic levels are shown on the figure. The total weight penalty is approximately 1500 lb which goes toward increasing the gage of the payload bay door skins, fuselage side skins, and part of the vertical fin. There is no weight penalty attached to the tank structure per se for these higher acoustic levels since the tank wall thickness is designed by pressure considerations and is adequate to withstand the predicted acoustic levels. Additional weight penalties may result for the parallel system as a result of the increased vibration environment seen by the orbiter equipments, particularly those in the aft sections. This will require the imposition of higher vibration qualification levels or a stronger structural design for these equipments at a cost and weight penalty that it is not possible to assess at this point.

The other area of more severe induced environment on a parallel burn configuration is caused by booster plume impingement. Radiation from the metallic particles contained in the SRM exhaust plume impose a high heat flux on the bottom of the HO tank and requires additional thermal protection to keep the tank temperature within design limits. Figure 21 shows the effect of this plume radiation on the HO tank base in terms of TPS penalty. The heat flux shown on the figure is essentially the additional flux generated by the SRM plume and the additional ablative protection, shown on the right hand side of the figure, results in a total weight penalty on the order of 1000 lb. We have also examined other potential sources of thermal environment penalties on the parallel configurations, such as interference heating between the tank and booster and plume induced recirculation heating near the aft section of the orbiter. We have found however, that neither of those phenomena have a severe enough effect to cause any additional weight penalty on the orbiter, tank or booster.

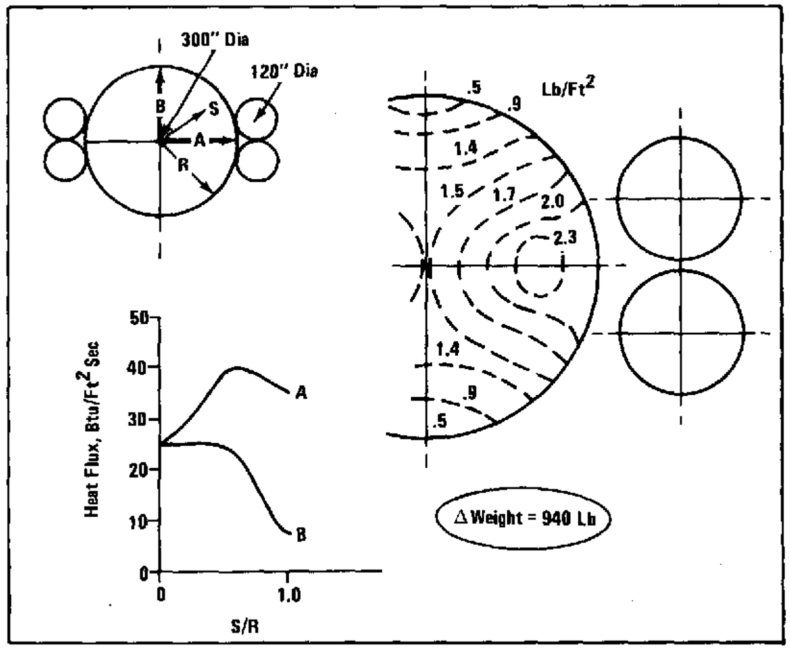

Figure 21 Tank Base Heating Penalty from Plume Radiation

CAN TVC AND THRUST TERMINATION BE ELIMINATED ON THE PARALLEL SRM?

One of the original attractions of a parallel burn configuration was the possibility that booster thrust vector control might not be required and that thus the cost and weight of the booster could be significantly reduced. During the first half of the second extension study period, considerable preliminary control studies were performed to determine whether such an approach was feasible. Our conclusions at the mid-term briefing were that there was sufficient uncertainty about the ability to control the configuration with orbiter engines and orbiter control surfaces alone to warrant the recommendation that booster thrust vector control be included in all further studies of booster size and cost.

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During the final half of the study period, we continued and extended these control studies by using a 6-degree-of-freedom digital simulation to explore all possible avenues of approach to the control of the combined configuration. Our studies included examination of control authority requirements due to orbiter/booster roll-yaw coupling and due to aero disturbances generated by worst case wind shear conditions at various altitudes in the trajectory. We then studied several possible methods of providing the control authority required. We looked at the possibility of using orbiter engines alone, then at coupling the engine control capability with those of the orbiter aero surfaces and finally at the combination of orbiter and booster engine control capability to provide control authority.

The results of these studies are summarized in Figures 22, 23 and 24. The figures show, in each case, the required control torque for each configuration axis and the amount of control authority available under conditions of all orbiter engines firing and of one orbiter engine out. Figure 22 shows clearly that the torque available from orbiter engines alone, even when

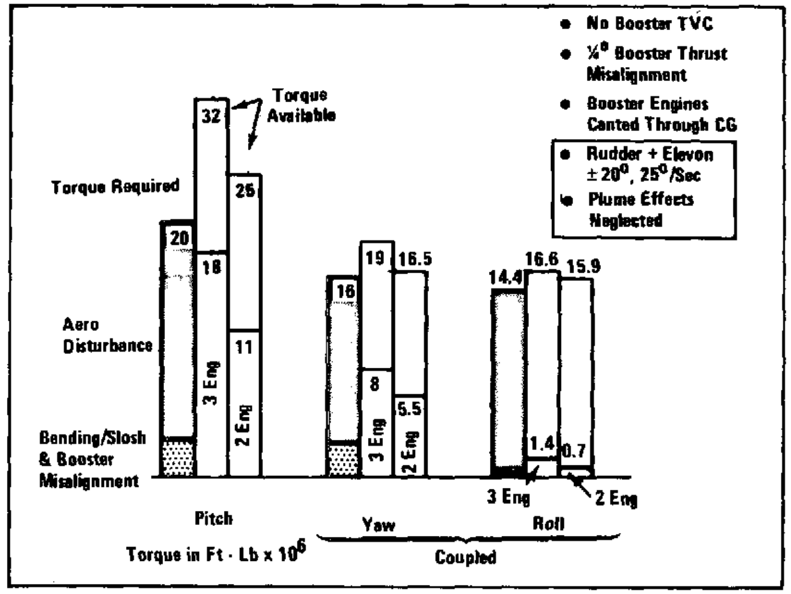


Figure 23 Aero Disturbance – Aero Surface Control – 156's

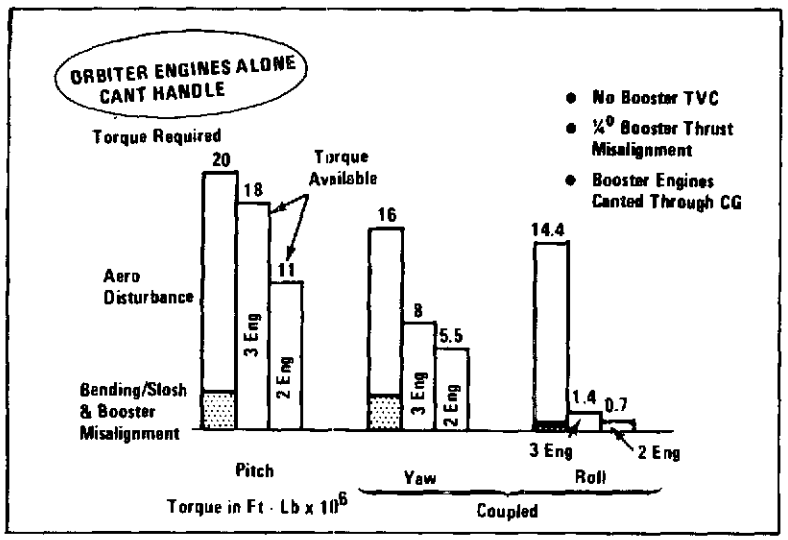


Figure 22 Aero Disturbance – Control Requirements – 156's

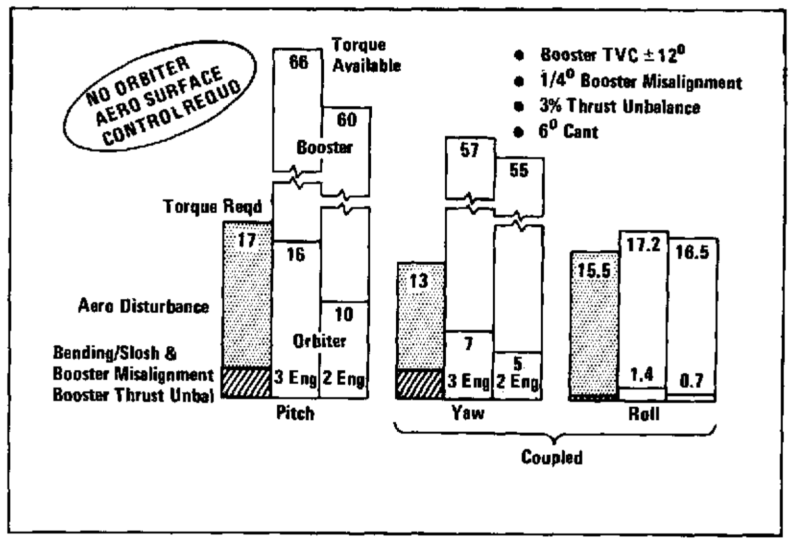


Figure 24 Aero Disturbance – SRM TVC Control

all engines are firing, is insufficient to provide the requisite control authority. The combination of orbiter engines and orbiter aero surface control comes close to meeting control requirements as shown in Figure 23. For the case of one orbiter engine out, however, this situation becomes sufficiently marginal in the pitch and yaw axes to still retain the mid-term conclusion that booster TVC is required. Note also that the data shown for aero surface control capability is based on ±20° deflection angle and 25 degree per second rate capability for both rudder and elevons which exceeds the normal design requirements for reentry and supersonic aerodynamic control. Thus, a design and weight penalty must be paid even for the somewhat marginal control capability provided.

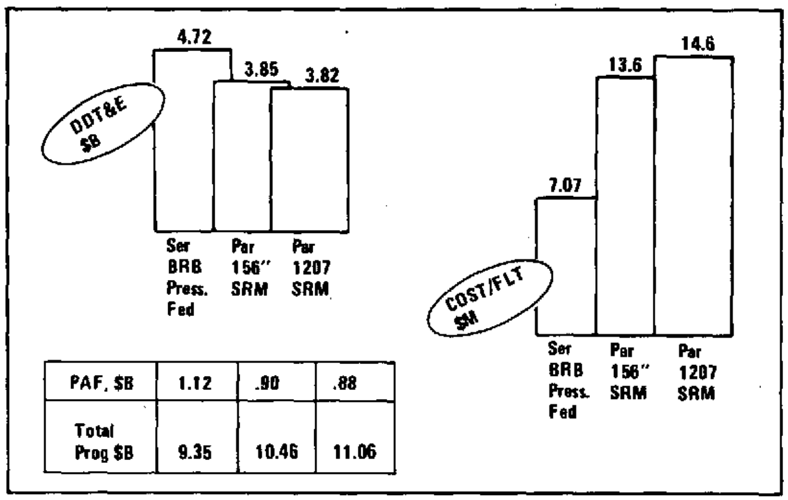
A better control margin can be provided by the use of fins on the underside of the tank. Approximately 410 sq ft of fin area is required to provide a 20% excess of control torque available over that required. The use of such a fin would impose weight penalty of approximately 2500 lb on the tanks. Additional fin surface could be added to reduce the aero surface deflection requirements closer to present design capability. This, however, would increase the weight penalty on the tanks. If booster thrust vector control capability is provided, the combination of booster and orbiter engine control authority is sufficient to provide the control torque requirements, Figure 24. Booster thrust vector control capability with an approximately ±12° gimballing range would eliminate the necessity for the use of orbiter aero surfaces.

In summary then, the question of "can booster TVC be eliminated?" can be answered as follows: "Yes it can, by the proper combination of orbiter engine gimballing, use of orbiter elevon and rudder deflection capability and by providing a relatively large fixed fin under the tank." However, all our weight and cost numbers are based on the assumption of TVC in the booster.

The question of whether SRM thrust termination can be eliminated however, must be answered in the negative. For the situation of a mission abort contingency arising during the early phase of the ascent boost flight, the orbiter must be capable of separating from the booster, which can only be accomplished if booster thrust is neutralized. Thus, the requirement for thrust termination of the SRM cannot be waived unless the probability of early mission abort is considered to be too small to design for.

HOW DO THESE CONFIGURATIONS COMPARE ON COSTS?

The comparative DDT&E cost, cost per flight, and peak annual funding data for the three configurations considered in this section are presented in Figure 25. Note that, typically, the development cost of the parallel/SRM configurations is about $900M less than that of a series burn/BRB system, but that the cost per flight of the expendable SRM configuration is nearly double that of the recoverable liquid propellant booster system. Peak annual funding for the liquid propellant booster system is on the order of $100M to $200M higher than that of the SRM configurations, but total program cost is higher for the solids since, for a 455 flight standard traffic model, 900 solid boosters must be manufactured as opposed to only 12 of the liquid propellant recoverable boosters for the series burn system.



PAF, $B	1.12	.90	.88
Total Prog $B	9.35	10.46	11.06

Figure 25 Series BRB vs Parallel SRM 15 x 60 Orbiter

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TO SUM UP

The results of the studies pertinent to the question of series BRB versus parallel SRM configurations are summarized as follows:

- Cost and Weight – Series BRB higher DDT&E by $870M but lower cost per flight by $6M
- Ascent Control -- Booster TVC requirement assumed weight and cost. Can be eliminated by use of fins and engine canting.
- Inflight Abort
  - High thrust OMS can close abort gap for all missions and configurations at zero or very low SSME EPL
- Induced Environment – For parallel burn:

  Added – 1450 lb to orbiter for acoustics
  - 940 lb of TPS on HO tanks
  - 7 db higher acoustic level at cargo bay
  - Higher equipment vibration levels
- HO Tank– Parallel tank more efficient – .005 to .008 difference in structural fraction

HOW HAS THE 15x60 BAY ORBITER CHANGED SINCE DECEMBER 1971?

The recent orbiter weight history is depicted graphically on Figure 26. Of particular interest is the weight growth shown from the 161,000 lb landed weight of the December 1971 version to the 190,000 lb target weight presently used in our weight reporting. This weight increase has resulted primarily from the changing requirements and groundrules imposed on the study by NASA. Figure 27 summarizes the changes from the December 1971 mid-term report. Up to that point, all orbiter design and performance requirements had been based on the Mark I version of the system and these are shown on the first column on Figure 27. Instead of four J-2S engines previously required for the Mark I orbiter, three 472K SSME's were now specified. Instead of the 25K lb up payload for the south-polar mission, 40K lb up-payload was now specified and similarly the requirement for down payload went from 25,000 to 40,000 lb. This increased the orbiter landed weight to 184,000 lb.

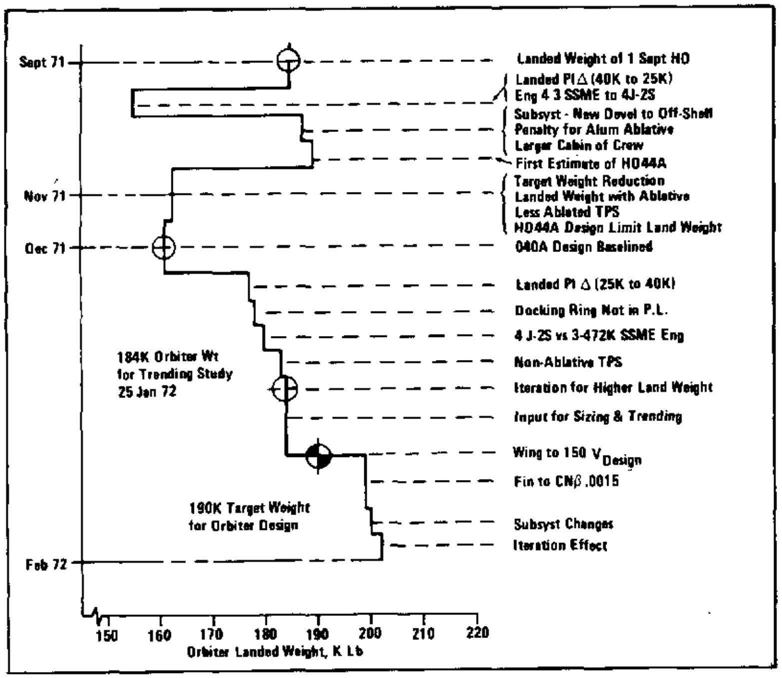


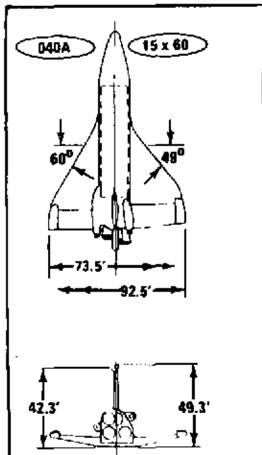
Figure 26 15 x 60 Orbiter Landed Weight History

The most significant performance requirement change was the reduction in design speed from 156 to 150 knots. It was this particular performance specification which had the greatest impact on the orbiter configuration and weight. Considerable aerodynamic studies were performed to examine

	Sept 71 - Dec 71 — 040A —		Feb 72 - 15 x 60 Orb
	Mk I	Mk II	
Main Engines	4 x J-2S	4 x 265K	3 x 472K
Payload Up - Polar	≤ 25K	40K	40K
Payload Down	25K	40K	40K
V_{Design}	156 Kts		150 Kt
Subsonic Long. Stability		2%	
C.G. Range		2%	
Hypersonic α Range		20^0 - 35^0	
External Shape	Communal 040A		Bigger Wing Less Sweep Bigger Fin & Rudder
TPS	Ablative	RSI & Carbon/ Carbon	RSI & Carbon/Carbon

• Orbiter Getting Heavier

Figure 27 Changing Requirements and Ground Rules Orbiter Status all the configuration options in terms of wing area, wing cross section, wing sweep angle, fin area, etc., which would allow us to meet the design speed condition at a minimum weight penalty. The lowest weight solution, which still, however, imposed about 6,000 lb orbiter landed weight penalty, involved a change in the wing reference area from 3150 to 3440 sq ft, a change in the landing edge wing sweep from 60° to 49° and in trailing edge sweep from 0 to -5°, a change in the wing cross section from symmetrical to twisted cambered and a change in the tail area from 354 to 550 sq ft. The orbiter evolution is summarized on Figure 28 which compares the February 1972 orbiter characteristics to those presented at the December 1971 mid-term briefing. Figure 29 shows the 15x60 payload bay orbiter configuration which corresponds to the present target weight of 190,000 lb. This configuration incorporates such recent baseline changes as nose docking rather than hood docking and the change from LM ascent engines to LM descent engines in the OMS. All sizing and trending data, however, is based on the 184,000 lb landed weight orbiter which was the version in existence at the time when we had to finalize our input to the trending programs.



	Dec. 71 040A		Feb. 72 15 x 60
	Mk I	Mk II	
Engines	4 x J2S	4 x 265K	3 x 472K
Dry Wt Lb	136K	133K	145K
P.L. Down, Wt Lb	25K	40K	40K
Landed Wt, Lb	161K	178K	190K
V_{Design}, Kt	167	175	150
C.G. Landed	65-67%	65-67%	66.1-68.1%
S_{Ref}, Ft^2	3155	3155	3440
L.E. Sweep	60°	60°	49°
T.E. Sweep	0°	0°	−6°
Wing Section	8% Sym	8% Sym	9% Camb
Body Chine	Soft	Soft	Soft
$C_n\beta$	.0007	.0007	.0015
Tail Area, Ft^2	354	354	550

Figure 28 Orbiter Evolution

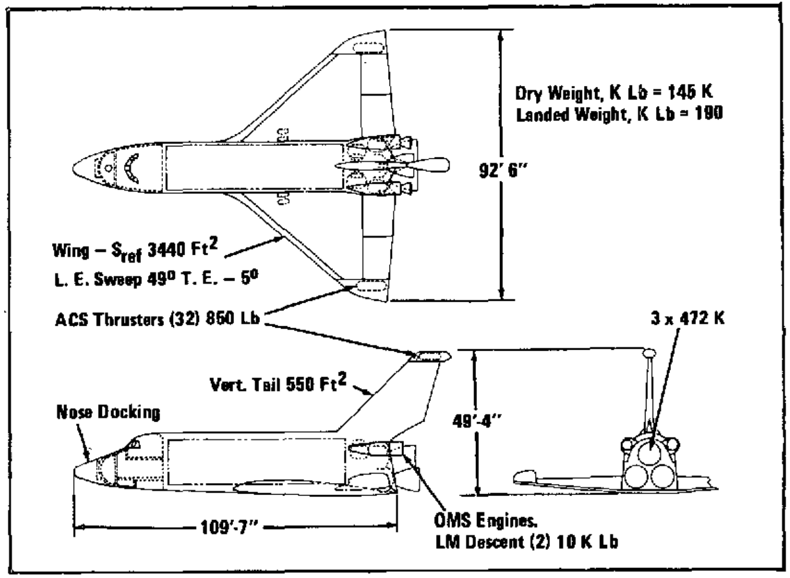


Figure 29 15 x 60 Orbiter

IS THE 14x45 PAYLOAD BAY ORBITER FEASIBLE?

The major problem encountered in arriving at an aerodynamically acceptable configuration for a 14x45 payload bay orbiter was the fact that the fuselage was reduced in length and diameter but the engine weight remained the same, thus causing the cg to shift too far aft for acceptable aerodynamic performance. Two options were open to us to achieve the desired performance as shown in Figure 30. In one option, the payload bay was extended from 45 to 50 ft in length and in the other option the total thrust of the engine system was reduced from the 1.4M lb of the three 472K engines to 1.14M lb corresponding to three 380K engines. In both cases, the RCS pod on the tail fin had to be moved to the forward section of the fuselage and the APU's from the aft section to the mid-body in order to obtain acceptable cg locations. We developed an orbiter configuration meeting all aerodynamic design requirements for each of these two approach options and their characteristics are presented in Figure 31 where they are compared to each other as well as to the corresponding characteristics of the 15x60

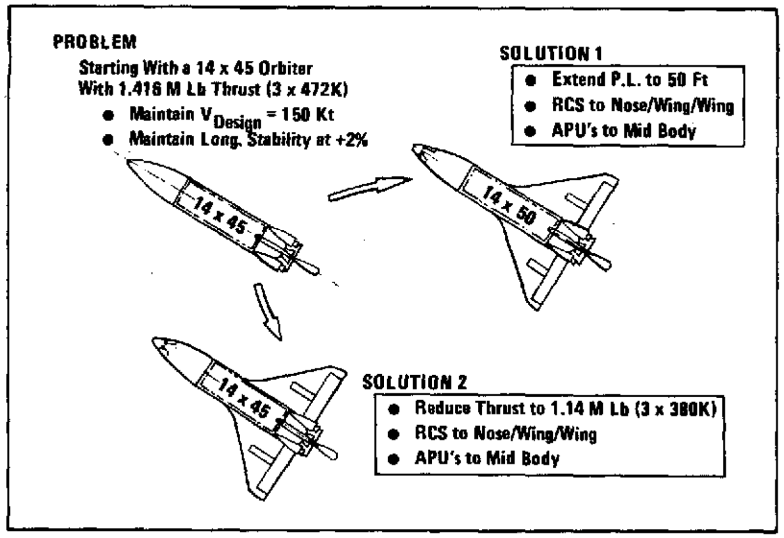
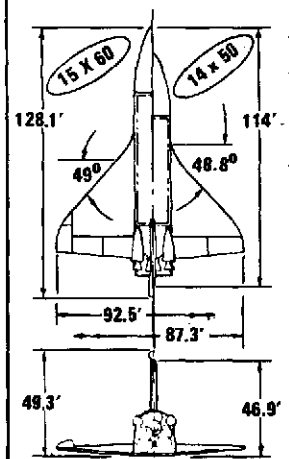


Figure 30 14 x 45 Orbiter Aero Options



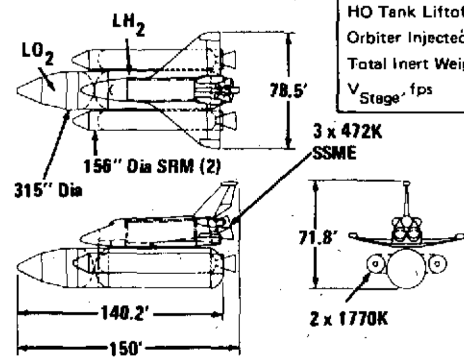
	15X60	14X50	14X45
P.L. Bay Vol, Ft^3	10,600	7,660	6,927
Press. Cabin Vol, Ft^3	3,550	3,180	3,180
Dry Weight, Lb	145K	137K	129K
Landed Weight, Lb	190K	167K	158K
P.L. Down Weight, Lb	40K	25K	25K
C.G. Aft, % B.L.	68.1	67	66.2%
Wings Ref Ft^2	3,440	3,060	2,950
Vert Tail, Ft^2	550	500	500
Total S_{Wet}, Ft^2	11,070	9,640	9,134
Approach Drag, Lb	32,000	27,000	26,300
T/W Staging	1.11	1.10	1.13
V_{Des}, Kt	150	150	150
Long Stab. Margin, %	2	2	2
Orbiter Engines	3 x 472 K	3 x 472 K	3 x 380K
Length, Ft	128.1	114.0	108.0
Span, Ft	92.5	87.2	84.5
Height, Ft	49.3	46.8	46.1

Figure 31 Orbiter Comparison

payload bay orbiter. Note that the 14x45 orbiter version with the smaller engine size results in lower dry weight for the orbiter itself, but, as will be seen later, it increases the overall configuration weight because of the lower thrust to weight resulting from the lower engine thrust.

WHAT ARE THE PHYSICAL CHARACTERISTICS OF A 14x45/ PARALLEL/SRM CONFIGURATION?

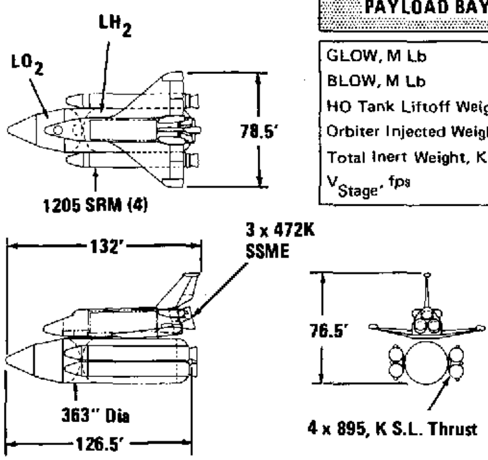
The launch configurations of 14x45 payload bay orbiter/SRM stacks using either four 1205's or two156" SRM's are shown in Figure 32 and 33. A comparison of major configuration characteristics is presented in Figure 34 in which the baseline system characteristics are also included for reference. Again we see that the SRM configurations show a significant reduction in total inert as well as overall liftoff weight relative to the liquid propellant baseline. It should be noted that the use of low thrust engines, which as shown in the previous section, resulted in the lower dry weight in the orbiter, did however increase the stack weight by anywhere from 200K to



PAYLOAD BAY SIZE 14 x 45

Glow, M Lb	3.705
BLOW, M Lb	2.144
HO Tank Liftoff Weight, M Lb	1.350
Orbiter Injected Weight, K Lb	211
Total Inert Weight, K Lb	476
V_{Stage}, fps	5,399

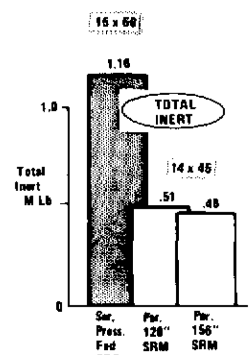
Figure 33 Launch Configuration – Parallel 156" SRM



PAYLOAD BAY SIZE 14 x 45

GLOW, M Lb	3.738
BLOW, M Lb	2.032
HO Tank Liftoff Weight, M Lb	1.495
Orbiter Injected Weight, K Lb	211
Total Inert Weight, K Lb	513
V_{Stage}, fps	4045

Figure 32 Launch Configuration – Parallel 1205



	15 x 60	14 x 45	14 x 45
	Series BRB	Par. 1205 SRM	Par. 156 SRM
GLOW, M Lb	6.396	3.738	3.705
BLOW, M Lb	5.119	2.032	2.144
OLOW, M Lb	1.277	1.706	1.561
Inert, M Lb	1.158	.513	.476
V_{Stage}, fps	4879	4045	5399
Payload Margin, Lb	7500	11000	6900

Figure 34 Launch Configuration Characteristics Comparison

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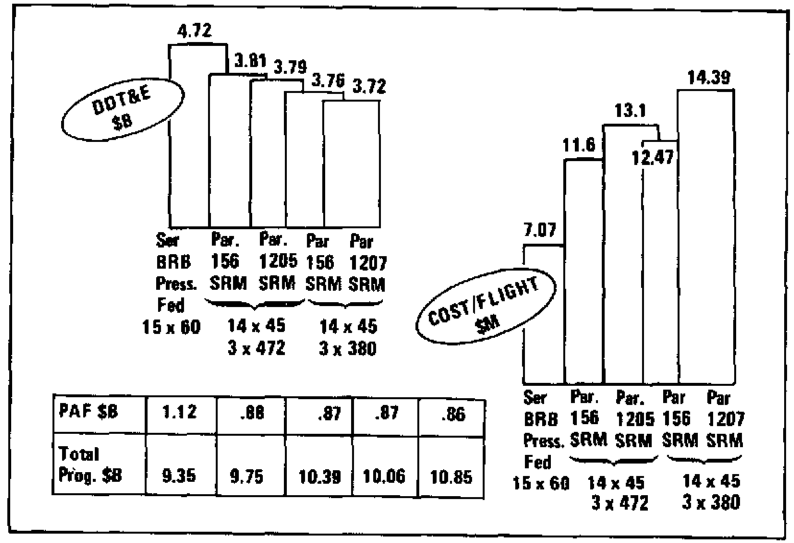
400K lb because of the lower performance capability at the low engine thrust.

HOW DO PAYLOAD AND BAY SIZE WEIGHT REDUCTIONS EFFECT COST?

The comparative costs of the small payload bay orbiter configurations relative to the baseline series/liquid propellant booster system are shown in Figure 35. The general cost relationships shown on that figure follow the same trends previously evidenced whenever a liquid propellant and solid propellant booster configuration were compared. That is, the development cost of the solid system is lower, but cost per flight and total program cost of the solid system is considerably higher than that of the liquid propellant booster configuration. In comparing the configurations options for the small payload bay orbiters, it may be noted that the low thrust versions show some reduction in development costs relative to the standard size engine version but as might be expected, the cost per flight increases since the increase in stack and in tank weight more than compensates for the lower refurbishment costs of the orbiter itself. Compared to the baseline, the 14x45/parallel/SRM system costs between $900M to $1B less to develop, but $4.5M to $5.5M more per flight. Relative to the standard payload bay size version of the parallel/SRM configuration, the small orbiter results in an approximately $40M saving in development cost, nearly $37M of which is the result of development cost savings in the orbiter itself. We found that 70% of the saving in development cost accrued from the reduction in payload weight rather than in size of the payload bay.



PAF $B	1.12	.88	.87	.87	.86
Total Prog. $B	9.35	9.75	10.39	10.06	10.85

Figure 35 Series BRB 15 x 60 vs Parallel SRM 14 x 45

TO SUM UP

The conclusions of our study relative to the effect of reducing the payload bay size and payload weight of the orbiter are summarized as follows:

- Design Evolution

	WAS	IS
Payload Up/Down	25K lb	40K
V_{Design}	167/175 Kt	150 Kt
Landed Weight	161/178K lb	190K lb
LE Sweep	60°	49°

- Small Payload Bay
  - Feasible – but 5 ft longer or use lower thrust engines
- Weight and Size
  - Small payload bay orbiter/parallel/SRM stack about 2.4 to 2.6M lb lower GLOW and 600K lb lower inerts than 15x6 orbiter/series/BRB
  - 14x45 payload bay orbiter lighter than 14x50 orbiter, but GLOW higher by 200K lb and total inerts higher by 30K lb

- Cost

Relative to baseline system:
  - Small bay orbiter/parallel/SRM saves nearly $1B development cost but
  - Increases cost per flight by about $5M

Relative to 15x60 payload bay orbiter/parallel/SRM system:
  - Small payload bay orbiter saves $43M DDT&E ($37M in orbiter) for 14x50 version and $92M DDT&E ($62M in orbiter) for 14x45 version
  - 70% of DDT&E saving is due to payload weight reduction
  - Cost per flight is $1.6M less for 14x50 orbiter, $0.5M less for 14x45 orbiter

SOLID PROPELLANT BOOSTERS

The solid propellant booster configurations considered in this study period are shown on Figure 36.

Our study of solid propellant boosters concentrated on the resolution of these key issues:

- The best method of providing booster thrust vector control capability
- Booster separation technique for a parallel burn system
- Choice between 120" and 156" diameter SRM'S
- Choice between parallel and series configurations
- Detailed evaluation of solid booster cost buildup from the motor to the complete stage

On the subject of ascent control, we had concluded that booster thrust vector control should be baselined for all configurations. We then performed a trade study to determine if liquid injection or mechanical nozzle gimballing should be employed as the SRM TVC method. We compared a gimballed nozzle with $\pm 7.5^{\circ}$ thrust vectoring capability to a liquid injection system capable of $\pm 5^{\circ}$ thrust deflection. Our study showed that the gimballed nozzle offered significant weight (by over 100K lb) and cost (by about $8M DDT&E and $800K per flight) savings as well as providing greater flexibility in accommodating changes in vehicle design and flight conditions. Although the study was done specifically for a parallel system, the general results are equally applicable to a series configuration. We thus baselined the gimballed nozzle as the thrust vectoring mechanism for all solid boosters.

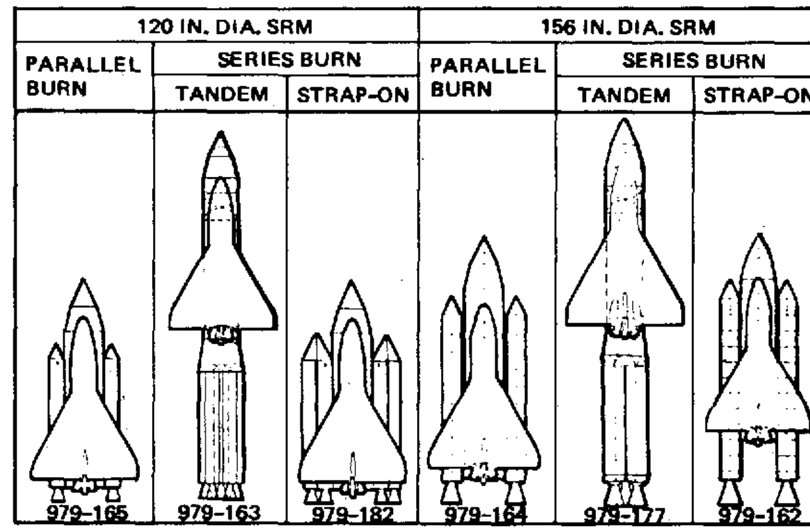
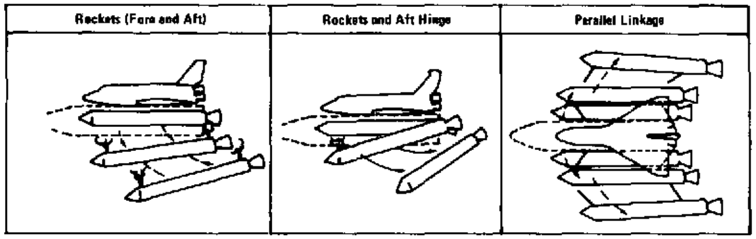


Figure 36 SRM Booster Configurations Candidates

For booster separation, we considered separation-rockets-only, mechanical-linkage-only and combination separation-rocket-forward/links aft system, as shown on Figure 37. The all-rocket system provided weight and cost advantages relative to the other approaches considered. (See Figure 37). In addition the lower development risk provided by previous Titan experience and the negligible load interaction with the orbiter warranted our baselining the rocket-only system as the booster separation approach.

We examined the factors relating to the choice of SRM diameter in some detail. Again, our studies were specifically oriented towards a parallel configuration, but the conclusions would apply equally well to a series system.



Discriminators	Rockets (Fore and Aft)	Rockets and Aft Hinge	Parallel Linkage
ΔWeight, Lb			
Booster	Baseline	+8180	-4030
Orbiter	Baseline	0	+5770
Payload	Base	- 600	-5000
ΔSystem Cost - ($$10^6$)			
DDT&E	–	+ 4.2	+ 0.6
*Cost/Flight	–	+ 0.26	0.12
Program	–	+ 119	+ 40.0
Experience	Titan III C	None	None
Plume Impingment on Orbiter	Worst	Moderate	None
Reaction Forces on Orbiter	Negligible	Worst	Moderate

*Does Not Include Transportation Cost

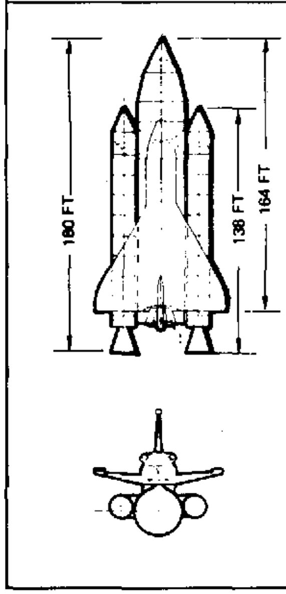
Figure 37 Separation Approach

The results of the trade study of 120" vs 156" SRM's (the only two diameters evaluated in detail) are presented in Figure 38. Clearly, the experience factor favors the SRM's since they have been used in operational Titan flights, whereas the 156" solids have only been test fired. This operational background reflects itself in a somewhat lower DDT&E cost, but the shuttle application requires sufficient additional motor and stage development on the 120" SRM's to make the development cost advantage relative to the 156" SRM's insignificant (about $30M). On the other hand, because the 120" SRM configurations require generally twice as many motors than do 156" systems, the cost/flight increases by about $1.5M for the case of a parallel configuration, thus adding approximately $600M to the total program cost at the standard traffic model. The significant decrease in cost/flight coupled with the greater reliability because of the fewer components and lower stage complexity makes 156" the preferred solid booster diameter.

DISCRIMINATORS	120 INCH DIAMETER	156 INCH DIAMETER
EXPERIENCE	TITAN III C 5 PFRT 17 FLIGHTS TITAN III M 1 PFRT	9 TEST FIRINGS THIOKOL AND LOCKHEED
TRANSPORT	UNRESTRICTED RAIL ROUTES	RAIL O.K. RESTRICTED ROUTES
HANDLING	EASIER	-
CONFIGURATION COMPLEXITY	GREATER	-
RELIABILITY (BOOSTER)	0.98	0.99
MOTOR QUANTITY (PRODUCTION)	1,780	890
BLOW	2,825 M LB.	2,767 M LB.
COST, BOOSTER (M)		
DDT&E	340	369
PRODUCTION	3,949	3,419
OPERATIONS	488	394
TOTAL	4,777	4,182

Figure 38 What is Preferred SRM Diameter? (Parallel Burn)

Having explored the major technical and cost factors relating to SRM's, we compared series and parallel configurations employing these solids (specifically 156" SRM's) as booster stages. The results of this comparative evaluation are summarized in Figure 39. We prefer the parallel system primarily because the lower GLOW and weight of total inerts of the parallel configuration (by about 300K lb in GLOW, and 100K lb in total inerts) and the reduction in number of SRM's required (from three for series to two for parallel) results in a $2M saving in cost per flight without penalizing the development cost. From the booster point of view, the technical problems of



PARALLEL BURN 979-164	DISCRIMINATORS	SERIES BURN STRAP ON 979-162
4.574	GLOW (M LBS)	4.627
2.767	BLOW	3.478
1.807	OLOW	1.149
	BOOSTER COST ($ M)	
369	DDT&E	379
8.2	COST/FLT	8.8
	ABORT	WORSE REQ. ENG. START UP
WORSE @ IGNITION	LOADS	WORSE DURING FLIGHT
WORSE PLUME & INTERFACE	HEATING	
WORSE	ACOUSTICS	
	ENGINE OPERATION	REQUIRES ALT. START
22	GROWTH SENSITIVITY $\left(\frac{\partial\ \text{GLOW}}{\partial\ \text{PAYLOAD}}\ \frac{\text{LB}}{\text{LB}}\right)$	25

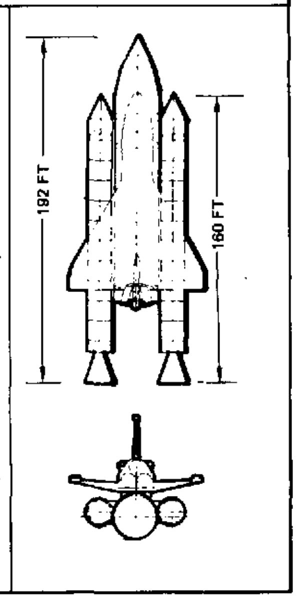


Figure 39 What is the Best SRM Booster?

integrating three SRM's into a tandem stage for a series configuration overshadow the attachment and separation problems of parallel mounted boosters, thus further adding to our preference for the parallel version of the SRM booster.

Considerable effort was devoted to estimating the cost of developing and producing a solid booster. Cost inputs were received from SRM manufacturers and the stage build-up costs developed from detailed manpower, material and subcontract estimates. Figure 40 summarizes the buildup of costs from the basic SRM to a fully integrated and tested stage for the case of a parallel burn 156" solid. We concluded that the motor itself represents a relatively small fraction of the total development cost (about 20%), which accounts for the minor difference in development costs between 120" and 156" solids, but constitutes the major proportion of production costs, which makes it imperative to minimize the number of solids required for the program.

From our studies of SRM boosters we concluded that:

- Parallel solid booster separation should be performed with separation rockets
- Booster thrust vectoring should be performed by gimballing the nozzles
- 156" solids are preferable to 120"

BOOSTER COST (MILLIONS)			
ELEMENTS	DDT&E	PROD	OPS
SRM	75.0	2,373.0	
STAGE HARDWARE			
STRUCTURE	19.0	593.7	
PROPULSION	11.3	74.3	
AVIONICS	2.7	21.1	
POWER	3.9	55.8	
SE&I	35.0	16.5	
FACILITIES	10.3		
SYSTEMS TEST	24.5		
GROUND TEST HARDWARE	14.5		
FLIGHT TEST HARDWARE	15.2		
SYSTEM SUPPORT	21.4	110.0	
MANAGEMENT	11.5	42.8	
FLT TEST OPS	110.0		
OPERATIONS			379.0
SUBTOTAL	354.3	3,287.0	379.0
TOTAL BOOSTER PROGRAM	4,020.3		

DOES NOT INCLUDE SHUTTLE MANAGEMENT

Figure 40 Stage Cost (Parallel Burn — 156" SRM Booster)

- The solid motor cost comprise a small portion of stage development costs but a major fraction of production costs
- From the booster point of view, parallel/SRM are preferred over series/SRM systems

LIQUID PROPELLANT BOOSTERS

The liquid propellant booster systems considered are shown in Figure 41. The study of liquid propellant boosters aimed primarily at:

- Refining the pressure-fed booster design with particular emphasis on ascent control, reentry and recovery as being the major configuration drivers
- Evaluating the comparative advantages of series vs. parallel configurations employing liquid propellant recoverable boosters and
- Providing the data required to make a selection between pressure-fed and pump-fed boosters

The optimum method of providing booster thrust vector control for the pressure-fed liquid propellant booster turned out to be liquid injection rather than mechanical gimballing. (For the pump-fed booster, which uses an existing engine, the mechanical gimballing capability of that engine would, of course, be retained). The major considerations

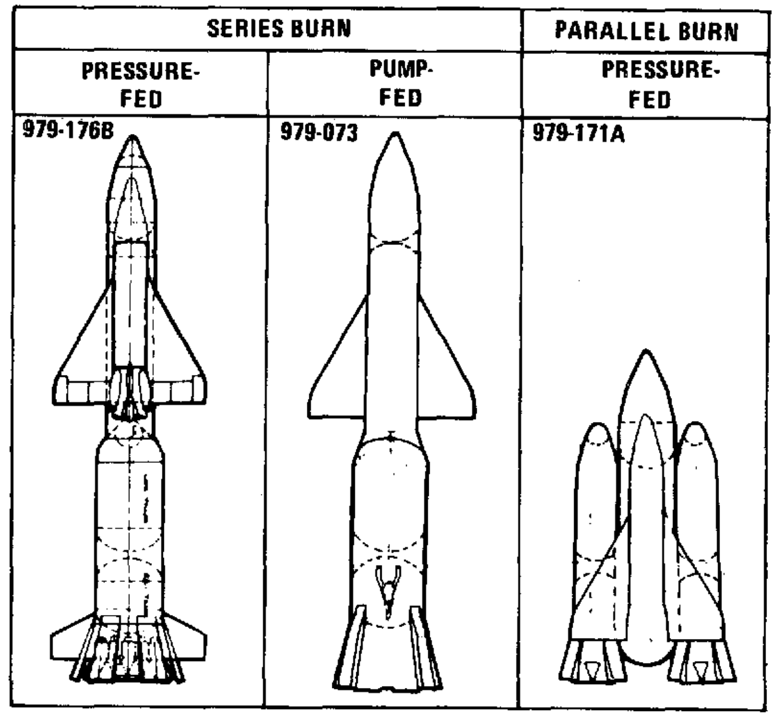


Fig. 41 Liquid Booster Configurations

for the case of the pressure-fed liquids were the additional weight and complexity that mechanical gimbals add to the base structure which would tend to compromise the capability for water impact survival and intact recovery. The control study also showed that a combination of orbiter control surface and booster engine control authority minimizes the deflection requirements and system weight (see Figure 42).

For reentry, we concluded that zero rather than high angle of attack was the preferred mode. This type of reentry assures aerodynamic stability without movable fins and active control systems. The recovery system selected is one consisting of parachutes only. As shown on Figure 43, the all parachute system is weight competitive with a combined retro-rocket/parachute system at the selected impact velocity of 100 fps, but is simpler and lower in cost than a combined system.

The summary comparison between series and parallel liquid propellant booster configurations is shown on Figure 44. From the booster point of view, the situation is very nearly a standoff in both development and per-flight costs, but when the overall system is considered, the reduction in HO tank weight and production cost results in a lower average cost/flight of the series configuration relative to the parallel burn system by about $300K. This cost advantage, coupled with the greater technical difficulties of integrating two parallel mounted boosters rather than a single tandem booster makes us prefer the series configuration in the case of liquid booster systems.

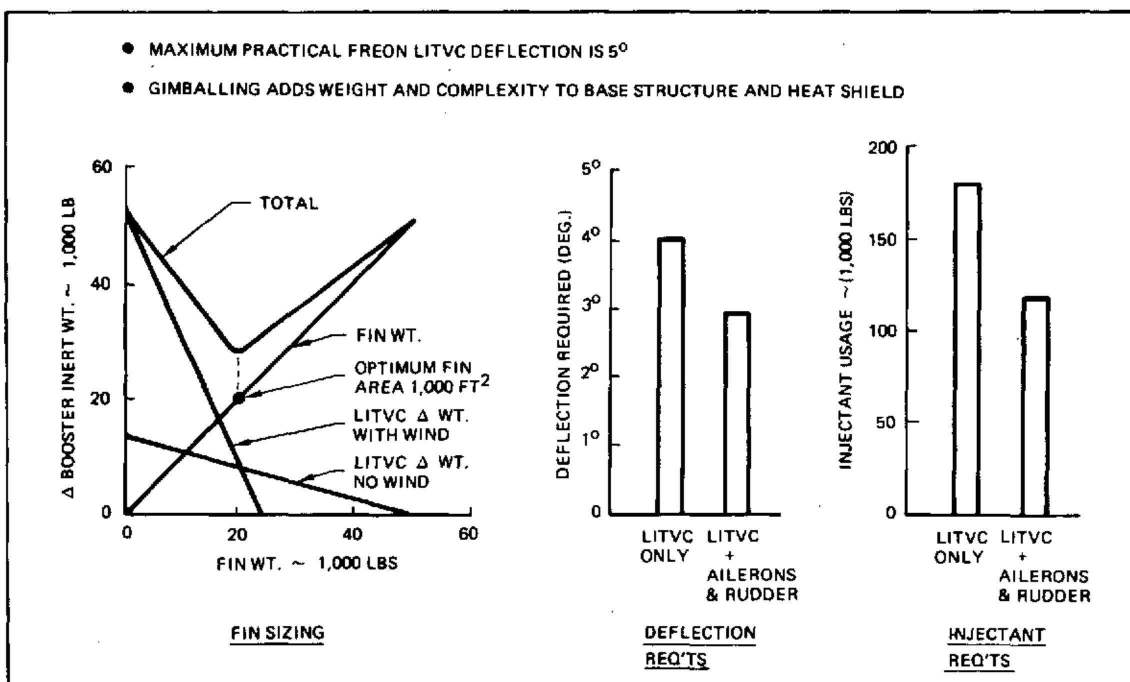


Figure 42 Optimization of Aero Surfaces/TVC Requirement Allows Effective Use of LITVC

- WHY ALL PARACHUTE SYSTEM ?
  - SIMPLICITY AND LOW COST
  - NEAR MINIMUM WEIGHT
  - LARGE PARACHUTE CLUSTERING WITHIN SOA
    - REQUIRED DEVELOPMENT PROCESS IS STRAIGHT FORWARD
  - NO MAJOR IDENTIFIABLE DESIGN OR DEVELOPMENT CONSTRAINTS
  - MINIMIZES VEHICLE STRUCTURAL PENALTY
- SELECTED SYSTEM
  - 3 – 70 FT DIAMETER DROGUES
  - 6 – 165 FT DIAMETER MAINS

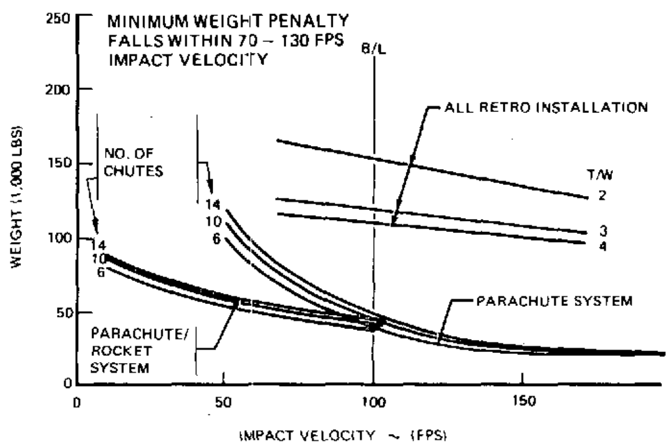
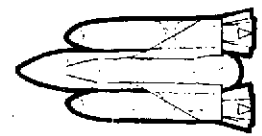


Figure 43 Recovery System Selection

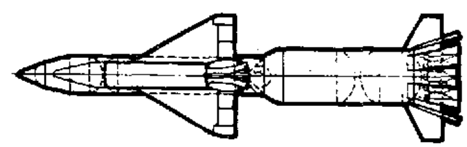
We investigated in some detail the design and cost aspects of a pump-fed booster for a series system for comparison with a pressure-fed stage. The pump-fed, ballistically recoverable booster is shown on Figure 45. Compared to the pressure-fed device, its inert weight is over 350K lb lower and its gross liftoff weight about 1.4M lb lower. One of the major advantages of the pump-fed booster is the decoupling between the engine and stage de-

PARALLEL BURN



• STAGING VELOCITY	5,500 FPS
• BOOSTER WT	
BLOW	4,276,200
INERT	773,400 (BOTH)
• BOOSTER COST ($M)	
DDT&E -	1,158
PER FLT -	2.6
• HO TANK	
INERT WT	85,300 LBS

SERIES BURN



• STAGING VELOCITY	5,000 FPS
• BOOSTER WT	
BLOW	5,032,900
INERT	881,800
• BOOSTER COST ($M)	
DDT&E -	1,259
PER FLT -	2.5
• HO TANK	
INERT WT	64,900 LBS

Figure 44 Series Burn vs Parallel Burn Liquid Boosters

GENERAL ARRANGEMENT

VEHICLE	
STAGING VELOCITY	5,300 FT/SEC
GLOW	4,870,200 LBS
BOOSTER	
INERT WEIGHT	520,400 LBS
BLOW	3,625,200 LBS
ORBITER	1,245,000 LBS

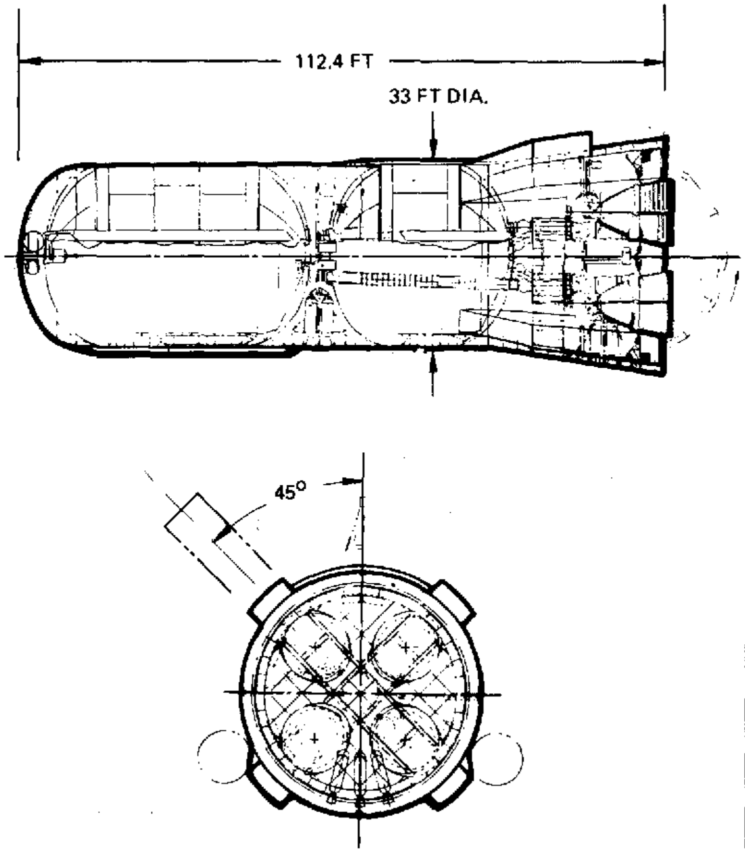


Figure 45 Ballistic Recoverable Booster — LOX/RP, Pump-Fed

velopment, since the turbopumps make the engine performance relatively independent of tank pressures. Furthermore, since we propose the existing F-1 engine for the pump-fed stage, an engine development program is not required and the development cost and risk is accordingly reduced.

The pump-fed versus pressure-fed issues are summarized on Figure 46, with the check marks indicating the preferred configuration relative to each of the evaluation parameters. We concluded that because of the lower development risk and cost (by about $500M) and the lower cost/flight (by about $500K), we prefer the pump-fed to the pressure-fed liquid propellant booster.

TO SUM UP

Our booster studies concluded that:

- For solid boosters, parallel/156" is the preferred configuration
- Comparing solids to liquids, the solids have the advantage of lower development cost, the liquids that of lower cost per flight

ENGINE TYPE IS THE FORCING ISSUE BEHIND:	PRESSURE FED	PUMP FED
• ENGINE/VEHICLE INTEGRATION	NEW ENGINE DEPENDENT DEVELOPMENT	✓ EXISTING ENGINE INDEPENDENT DEVELOPMENT
• THRUST VECTOR CONTROL	NEW DEVELOPMENT LIMITED GROWTH	✓ GIMBALLED ENGINE
• RECOVERY & WATER IMPACT	✓ LOW I_s AND LOW λ' REQUIRE HIGH PROPELLANT LOADS	✓ HIGH I_s AND HIGH λ' RESULT IN SMALLER SIZE
• TANK FABRICATION	HIGH PRESSURES REQUIRED THICK WALLS	✓ NORMAL TANK WALL SIZES
• BOOSTER COST ($ IN MILLIONS)		
RDT&E	$ 1,259	✓ $ 788
COST/FLT	$ 2.5	✓ $ 2.0

Figure 46 Pressure-Fed vs Pump-Fed Issue

- For liquid propellant boosters, the pump-fed represents the lower cost, lower risk approach is thus preferred over the pressure fed machine

The major cost and technical booster issues are summarized on Figure 47.

	PARALLEL SRM	BRB SERIES PRESSURE FED	BRB SERIES PUMP FED
BOOSTER DEVELOPMENT COST	0.4 B √	1.3 B	0.8 B
POTENTIAL LOW COST/FLIGHT		√	√
DEVELOPMENT RISK			
PROPULSION	√		√
RECOVERY	√		
SYSTEM FLEXIBILITY			√
ORBITER/BOOSTER INTERFACE COMPLEXITIES		√	√
ENVIRONMENTAL ISSUES		√	√

Figure 47 The Issues

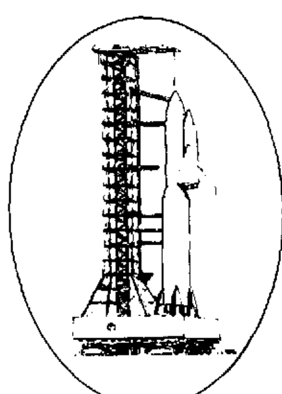
PAD ABORT

One of the major concerns of this final study period was the evaluation of the implications of providing pad abort capability. We considered the subject sufficiently important to devote a separate section to a discussion of what we did, why we did it and what we found out about pad abort.

WHAT REQUIREMENTS ARE WE TRYING TO MEET?

To determine the system requirements for pad abort capability, we systematically postulated all the failures which could require pad abort. We then evaluated the criticality of each of the failure conditions to establish which ones would impose the most severe requirements on the pad abort system. The results of this evaluation are summarized on Figure 48. We found that the most time critical failure would be an explosion of the booster/HO tank caused by either uncontrollable over-pressurization or by fire. This occurrence would generate a blast-wave having the characteristics shown on Figure 49, which depicts the overpressure (delta-p over atmospheric) conditions at the altitudes and times indicated. For the purpose of our pad abort studies, we assumed a 20% TNT equivalence of the baseline series/BRB combination of propellants. The delta-p = 3.0 psi dashed line represents the maximum overpressure the orbiter is considered to be capable of withstanding without sustaining damage that would prevent a successful glide return to the landing strip. (Later studies showed that this value might be increased to 4 psi with a small structural penalty.)

Since abort capability improves as the ability to accelerate away from the source of the blastwave increases, we looked into the maximum g



	Prior to Liftoff	Post Liftoff
BRB/HO Feed Sys/Eng - Fire/Explosion	**Most Critical**	**Most Critical**
Rupture of Propellant Lines	?	Rapid Escape
Overpressure of Booster or HO Tank	Non-Time Critical	Not Time Critical
Inadvertant Release of Restraint System	Time Critical at Engine Ignition	Time Critical at Engine Ignition
Loss of an Outboard Engine		Time Critical
Guidance and Control		Time Critical
Solid Booster Fails to Start	Non-Time Critical	
Solid Booster Burn Through	Time Critical	Time Critical
Solid Booster Nozzle Failure		Time Critical
TVC Failure (Solid)		Time Critical

Figure 48 Failure Criticalities

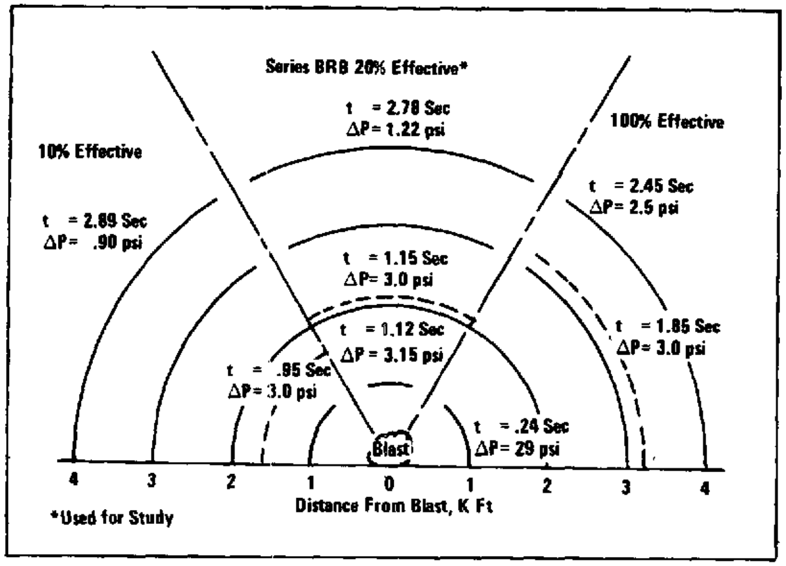


Figure 49 (5.0 x 10^6 Propellant Explosion) Blast Wave Characteristics

loading the orbiter could tolerate if designed in accordance with nominal requirements plus safety factors. This turned out to be between 4.2 and 4.5g acceleration. The constraint of maximum allowable vehicle acceleration established that we could not escape the wave front without experiencing catastrophic overpressure, unless there was some warning of the incipience of an explosion. The warning time required is plotted as a function of T/W (or acceleration in g's) on Figure 50. The warning time for "Max Payload" corresponds to the T/W that would be obtained if, at zero payload, the maximum allowable acceleration capability were provided. It is clear that between 5 and 7 seconds of warning time is essential if a pad abort capability is to exist. The bullet designated "SSME Start up Delay = 2 sec" illustrates the kind of warning time requirement necessary if the orbiter engines themselves were to be utilized to provide the escape thrust. This excessive warning requirement was the major reason why the attempt to use the SSME's in conjunction with orbiter cryogenic fuel storage as a pad abort system was quickly abandoned.

CONFIGURATION APPROACHES

The approaches considered for providing pad abort capability are shown on Figure 51. The configurations employing the main orbiter engines to provide abort thrust were eliminated after a brief study because:

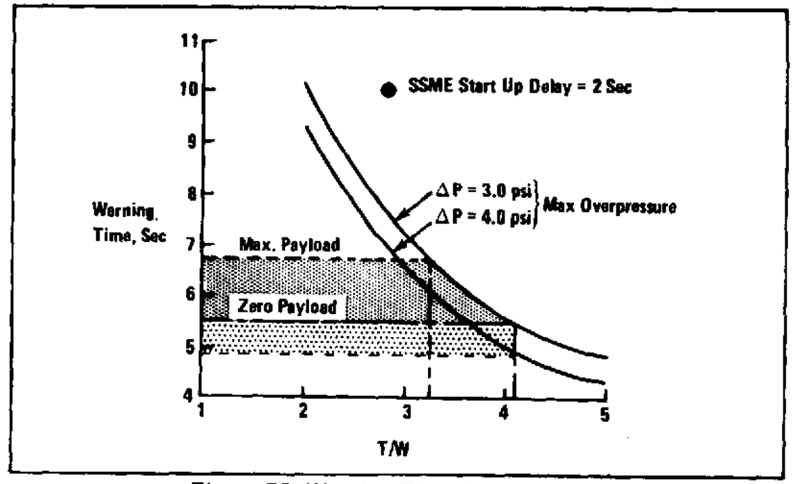


Figure 50 Warning Time Requirements

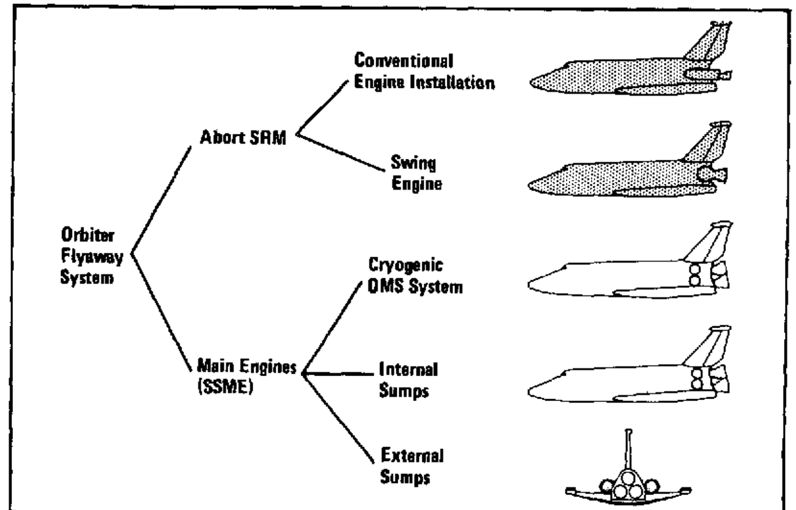
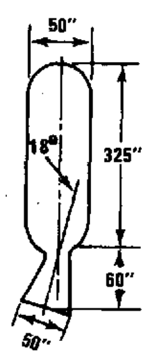


Figure 51 Configurations Considered for Pad Aborts

- Since only two of the three engines are usable for abort (three engines do not allow thrust vectoring away from the tank) the T/W is too low (T/W = 2.56) for effective abort

- The inert weight penalties imposed by the requirement for propellant storage on the orbiter are effective for the entire mission as opposed to the abort rocket system, in which the unused inert weight can be jettisoned at or before booster staging

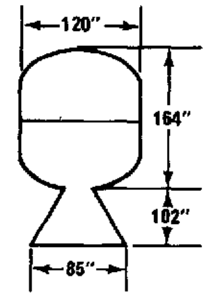
We selected for detailed study a series/BRB stack with two orbiter versions – one a conventional orbiter with two abort rockets strapped to the aft end of the fuselage above the wings – the other a swing engine orbiter with a single abort rocket mounted in the cavity in which the engines are normally stowed after orbit insertion. The abort rockets for these orbiters were sized to provide the maximum allowable T/W for a zero payload launch and to provide the impulse to impart sufficient energy to the orbiter for a glideback to the proposed new landing strip at KSC (825 fps at a burnout altitude of 6600 ft). The characteristics of the abort SRM's which meet these requirements are shown on Figure 52 for the fixed-engine orbiter and on Figure 53 for the swing-engine orbiter.



2 Motors Required

Thrust Start (S.L.) = 500 K Lb
Thrust Burnout (S.L.) = 380 K Lb
I_{sp} S.L. = 242 Sec
Burn Time = 14.6 sec
Chamber Press. (Avg) = 1400 psia
Total Impulse S.L. = 645 x 10^6 Lb - Sec
Inert Weight = 6 K Lb
Propellant Weight = 26,700 Lb.
Total Weight = 32,700 Lb.
Mass Fraction = 0.82

Figure 52 Abort SRM Characteristics



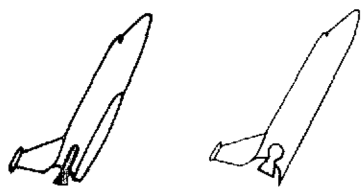
Thrust Start (S.L.)	= 900 K Lb
Thrust Burnout (S.L)	= 680 K Lb
I_{sp} S.L.	= 242 Sec
Burn Time	= 15 Sec
Chamber Press (Avg)	= 900 psia
Total Impulse	= 11.7 x 10^6 Lb-Sec
Inert Weight	= 10,721 Lb
Propellant Weight	= 48,838 Lb
Total Weight	= 59,559 Lb
Mass Fraction	= 0.82

Figure 53 Abort SRM Characteristics – Swing Engine Configuration

We concluded from our configuration studies that both approaches to providing pad abort capability were feasible, but that the swing engine configuration had a number of advantages relative to a conventional orbiter. It permits the use of a single rocket and provides a convenient mounting location for it and it eliminates the concern about propellant line disconnect clearance for abort separation of the orbiter from the tank. Furthermore, since it allows a more efficient HO tank design (with the LO_2 tank aft) it results in a lower GLOW configuration. Figure 54 summarizes the major weight increments of the swing engine orbiter configuration relative to the baseline for both the no-pad-abort and pad-abort cases. Note that although the swing engine orbiter is somewhat heavier than the baseline, the improvement in tank efficiency more than overbalances the orbiter weight penalty to the extent where, even with pad abort capability, the swing engine system is lighter than the no-pad-abort baseline.

WHAT IS THE IMPACT OF PROVIDING PAD ABORT CAPABILITY?

The weight penalties for implementing pad abort capability, shown on Figure 55, are on the order of 200-300K lb. in GLOW and 20-30K lb. in total inerts.



	No Pad Abort		Pad Abort	
	Series BRB Baseline	Swing Engine	Series BRB	Swing Engine
Δ Landed Weight, Lb	0	+54	+840	+2056
Δ HO Tank Dry Weight, Lb	0	-6698	+549	-6051
Δ GLOW, K Lb	0	-204	+237	-10

Figure 54 Pad Abort – Series BRB – Swing Engine

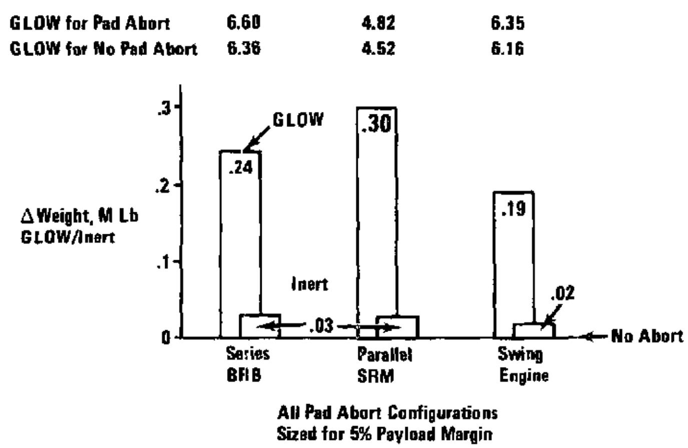


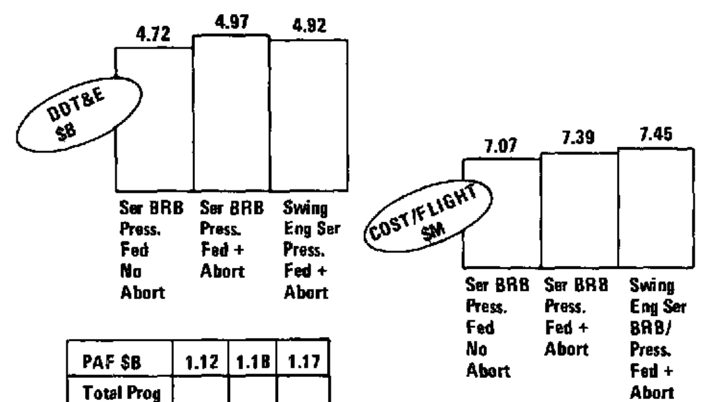
Figure 55 Δ Weight GLOW/Inert for Pad Abort

The cost impact is shown on Figure 56. Based on Apollo experience in designing and qualifying the launch escape system, we estimate a $250M development cost penalty for providing pad abort capability. The cost per flight increase is about $300K, the major portion of which is the cost of the abort rockets.

TO SUM UP

- The most critical pad failures for intact orbiter recovery were orbiter/booster explosion
- The use of SSME was not considered effective; SRM's were selected for baseline and swing engine orbiters

- The swing engine provides:
  - Convenient abort rocket location
  - Simple orbiter/tank interface (no propellant lines-safer)
  - Minimizes pogo potential
- For the baseline BRB, pad abort adds approximately:
  - 60-65K lb inert weight to staging
  - 250K lb GLOW
  - 800 lb orbiter dry weight
- Cost increases by approximately $250M DDT&E, $0.30M per flight
- Pad abort is feasible but will complicate design effort



PAF $B	1.12	1.18	1.17
Total Prog $B	9.35	9.77	9.71

Figure 56 Pad Abort Cost Comparisons

Our overall system evaluation and comparison was generally confined to those configurations which survived the pre-screening applied in each of the study areas discussed previously. For the case of series/BRB baseline, we did, however, consider both pressure-fed and pump-fed boosters, and for the representative 14x45 orbiter configuration we used the 120" rather than the 156" booster stack as having the lowest development cost of all options studied. The factors used for evaluation were the usual cost elements: development, per flight and total program; technical factors related to design complexity; inflight abort capability, severity of induced environment and control of the combined configuration; and the impact on the environment.

The results of our evaluation were summarized on Figure 6, which is reproduced here for the reader's convenience as Figure 57. We have checkmarked the configurations which we consider the best performers relative to each of the evaluation factors used.

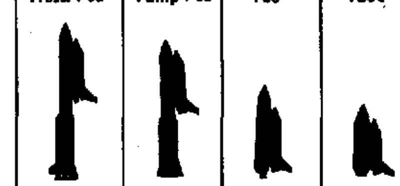
	Ser/BRB 15 x 60 Orbiter		Par/SRM 15 x 60 Orb.	Par,SRM 14 x 45 Orb.
	Press. Fed	Pump Fed	156"	120"
				
Lowest Total Program Cost, $B	9.35 ☐	8.66 ☑	10.46 ☐	10.39 ☐
Lowest DDT&E, $B	4.72 ☐	4.23 ☐	3.85 ☑	3.79 ☑
Lowest Cost/Flight, $M	7.07 ☑	6.62 ☑	13.1 ☐	13.1 ☐
Least Complex Design • Least Acoustic Impact	☑	☑	☐	☐
• Easiest Ascent Control	☑	☑	☐	☐
Least Environmental Impact	☑	☑	HCL ☐	☐

Figure 57 Configuration Comparison Summary

The lowest total program cost system turned out to be the series/pump-fed BRB configuration. This is the consequence of the lowest cost/flight combined with relatively low development cost of a system using that type of booster. If only DDT&E are considered, the parallel/SRM configurations are the best performers, with the small payload bay orbiter showing only relatively minor reduction in development cost, however, as compared to the standard orbiter/parallel/SRM system. Cost per flight favors the series systems, since the recoverability of the liquid propellant boosters significantly reduces the out-of-pocket costs for each launch. The pump-fed booster system exhibits a somewhat lower launch cost than the pressure-fed. This is attributed to the fact that the pump-fed booster, employing only four high thrust engines, has a smaller base crossectional area than the 7-engine pressure-fed booster, thus allowing use of a deployable shield for engine protection at water impact and a commensurate reduction in refurbishment cost.

In the technical areas affecting design complexity, the series systems are generally superior. The control problem is simpler, since the roll moments, which pose the most stringent control authority requirements, are lower. The acoustic and thermal induced environments are more benign, since the orbiter engines are not fired during combined ascent boost. The series system abort capability is somewhat better because the orbiter T/W at equivalent energy-levels is higher, but neither system has an abort gap.

The parallel solid systems do exhibit an adverse environmental impact characteristic in that they generate HCl as a combustion product, but the total amount of the pollutant is very small compared to that produced throughout the world by industrial operations.

Based on the above evaluation, our conclusions and recommendations can be summarized as follows:

SRM's

- All SRM's have lower DDT&E but higher cost per flight than recoverable liquid boosters
- All SRM applications make program potentially more sensitive to environmental issue
- On the basis of higher cost/flight, SRM's appear less attractive than liquids over the long haul. For the lowest costs during development, SRM's are preferred, but the shuttle program becomes more vulnerable on the environmental issue

LIQUIDS

- On balance, we prefer the series liquid boosters for shuttle development opting for the lowest cost per flight as vital to the future of the program
- Between liquids, the pump-fed booster has the right combination of cost/risk/performance

PAYLOAD

- Most cost reduction benefit is derived from payload weight reduction of 20K rather than inert weight reduction of orbiter
- Balance of orbiter is difficult; bay needs 50 ft length with 3 x 472 SSME's or lower thrust engines (3 x 380K) must be provided
- If we must minimize DDT&E, reduce payload requirement first, – but hold on to 60 ft bay

PAD ABORT

- Can be achieved, but as on previous programs will compound the design effort
- Will increase cost per flight by 300K
- Let's make sure we understand all implications before we proceed with requirements
- Swing engine is preferred arrangement for pad abort – minimizes cost to system