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FINAL REPORT

ALTERNATE CONCEPTS STUDY

EXTENSION

VOLUME II

CONCEPT ANALYSIS AND DEFINITION

Part 3

SRM Boosters

CONTRACT NAS8-26362

Prepared for George C. Marshall Space Flight Center by

Manned Space Programs, Space Systems Division

LOCKHEED MISSILES & SPACE COMPANY, INC.

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Final Report

ALTERNATE CONCEPTS STUDY EXTENSION

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LOCKHEED MISSILES & SPACE COMPANY
SUNNYVALE CALIFORNIA

FOREWORD

This is the final report of a four-month extension of the Phase A Study of Alternate Space Shuttle Concepts (NAS 8-26362) by the Lockheed Missiles & Space Company (LMSC) for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center (MSFC). This study extension, which began on 1 July 1971, was to study two-and-one-half stage, stage-and-one-half, and SRM interim booster systems for the purpose of establishing feasibility, performance, costs, and schedules for these system concepts.

The final report consists of three volumes (6 books) as follows:

- Volume I - Executive Summary
- Volume II - Concept Analysis and Definition
  - Part 1 - O4OA System
  - Part 2 - One-and-One-Half Stage System
  - Part 3 - SRM Booster
  - Part 4 - Avionics
- Volume III - Cost Analysis

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Section 1
INTRODUCTION

1.1 SOLID ROCKET BOOSTER TASK OBJECTIVES AND APPROACH

As defined in the contract statement of work and the study plan (Ref. 1-1) the broad objective of the Alternate Concepts Study was to consider configuration and programming options which would lower peak annual funding requirements.

As program redirections occurred, specific study objectives were somewhat modified, though in a broad sense they remained relatively unchanged in principle. Specific objectives applicable to the SRM booster task, as stated in the Study Plan were:

- Establish an optimum interim expendable solid rocket booster concept
- Determine technical risks involved
- Determine safety and mission success probabilities
- Determine an optimal flight test approach

Subsequent redirections impacted these objectives in that the booster philosophy was inclining toward elimination of interim configurations, and that expendable solids were considered unlikely candidates for final booster configurations. Accordingly, the level-of-effort on SRM boosters was severely reduced during the last half of the study period.

The retrenchment of interest in the solids, however, was offset by the following three factors which came to a focus in the last half of the study:

- Performance capabilities of SRM boosters improved and estimated costs decreased as detail design and costing tasks were completed.
- Performance and Cost uncertainties and program risks proved to be less than anticipated earlier for the SRM boosters.

- The two foregoing considerations for the prime booster candidates appeared to be become more questionable or uncertain as the detail studies progressed.

Accordingly, although operating at a comparatively limited level of study, the activities were partially reoriented to provide some indication of SRM booster capabilities in the context of a single-booster (no interim) program approach.

This volume of the final report was therefore assembled to emphasize the solid rocket booster potentials in the context of a single booster program. Data are shown as developed for the interim program, where such data are applicable (or can be converted to be) to the single booster concept. Time and budget available, however, permitted only cursory comparative examination with other competing candidates.

After completion of the initial sizing studies a baseline SRM-stage configuration, consisting of a single-stage SRM booster with four 156-in. dia. solid rocket motors (SRM), was selected. The subsequent activities are summarized in Fig. 1-1 and consisted of three activity groupings: (1) study of SRM performance characteristics and cost estimates, (2) preliminary design and cost estimates for the stage subsystems, and (3) analysis of the system potentials based on projections from the present studies.

SOLID ROCKET MOTORS (SRMs)

- PERFORMANCE DATA COMPARISONS
- COST DATA COMPARISONS
- SUBCONTRACTOR STUDY - DATA CRITIQUES
- DOCUMENTATION OF SUBCONTRACTOR STUDIES

SRM STAGES

- STAGE WEIGHTS AND COSTS
- SRM STAGE MANRATING REQUIREMENTS
- RECOVERY AND REFURBISHMENT POTENTIALS
- PROGRAM COSTS

SRM BOOSTER SYSTEM STUDIES

- SRM BOOSTER COST-EFFECTIVENESS POTENTIALS
- SYSTEM UNCERTAINTY AREAS
- FUTURE STUDY NEEDS FOR SRM BOOSTERS

Fig. 1-1 SRM-Stage Activities

Since the level of effort on SRM boosters was reduced after the midterm briefing (by NASA direction), the study activity was based on extension of the original work completed. Therefore, the data presented herein do not conform with the latest groundrules for the remainder of the study. Except for some work presented in regard to future potentials for an SRM booster which is employed as a final operational booster, the work represented presumes utilization of the SRM as an interim booster for 12 operational launches.

The baseline SRM booster selected was the Configuration 5B (shown in Fig. 1-2), which consists of a cluster of solid rocket motors (SRMs) serving as Stage 1, with a tandem-mounted hydrogen/oxygen tank to which is attached the orbiter vehicle. The single SRM booster stage was studied both in a four-motor configuration and in a three-motor configuration, each providing the same value of delta-V and each utilizing the same total amount of propellant (at 3.6 lbm). Most of the design analysis was performed on the four-motor configuration.

The three-motor configuration offers several advantages which came to light later in the study. This configuration provides a reduced recurring cost with a slightly increased nonrecurring cost. Also, it provides a better configuration orientation for the thrust termination ports.

1.2 SRM STUDY APPROACH

For the prime subsystem in an SRM booster, namely the solid rocket motors (SRM), it was decided that the objectives outlined in the previous section would be accomplished best by formally introducing the experience of SRM contractors. The bulk of the large solid motor technology has been collectively developed by four major propulsion companies. A working relationship was established and maintained throughout this study to ensure that the SRM data being used were current and valid, and represented the judgment of the most knowledgeable sources. The four companies were funded the latter part of August to generate and provide to this study specific design and cost parametric data that could be used to establish realistic SRM motor designs and the respective development and recurring costs. Figure 1-3 provides an outline of these subcontract efforts. The statement of work for these studies is contained in Ref. 1-2.

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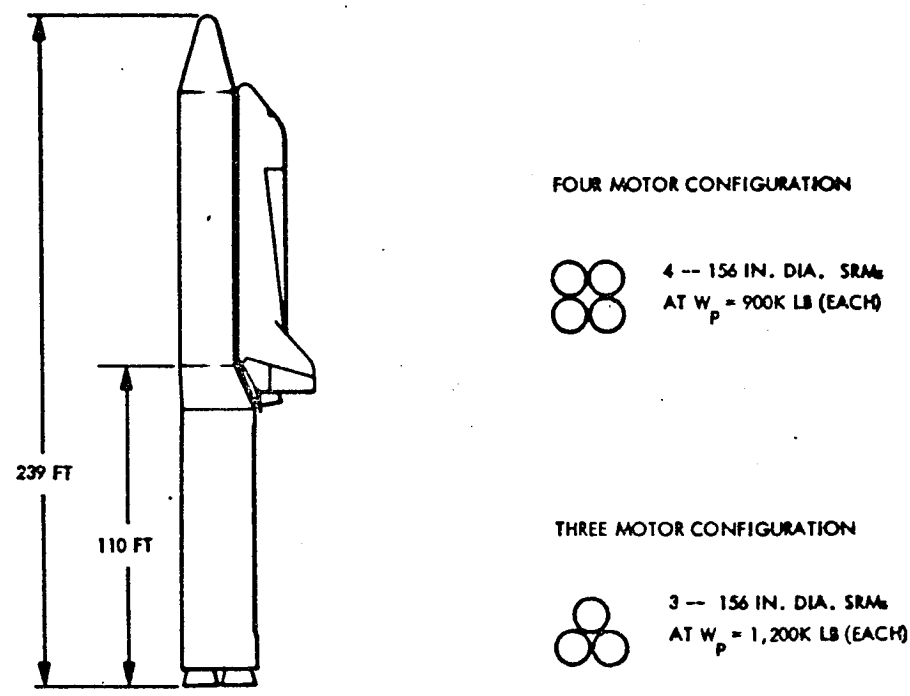

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Fig. 1-2 Configuration 5B SRM Booster

PURPOSE: TO OBTAIN COMPARATIVE PARAMETRIC DESIGN, PERFORMANCE, AND COST DATA FOR SRMs FOR POTENTIAL SPACE SHUTTLE APPLICATIONS

MOTOR SIZES	COMPANY	GO-AHEAD DATE	INITIAL INPUT	INTERIM REPORTS	FINAL REPORTS
156/260 IN.	AEROJET GENERAL (AGC)	8-24	9-14	10-22	11-15
120/156 IN.	LOCKHEED PROPULSION (LPC)	8-24	9-14	10-22	11-15
120/156 IN.	UNITED TECHNOLOGY (UTC)	8-31	9-21	10-22	11-15
120/156/260 IN.	THIOKOL CHEMICAL (TCC)	8-24	9-14	——	10-22

DESIGN DATA

- MASS FRACTIONS
- SPECIFIC IMPULSE
- MOTOR LENGTHS
- 3 MEOPs
- TWO BURN TIMES
- TYPICAL MOTORS
- SPECIAL DESIGN TRADEOFFS

COST DATA

- NONRECURRING: 8 & 12 MOTOR FIRINGS
- RECURRING: 20M & 40M LB/YR PROPELLANT PRODUCTION
- SPECIAL FACTOR INFLUENCES ON COST

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Fig. 1-3 SRM Subcontractor Studies

Preliminary submittals of data were received, analyzed, and integrated into the LMSC study effort. Final reports from the SRM contractors are being included as reference documents to this report. Engineering memorandums have been prepared which present and evaluate the contractor data submittals in some detail.

A primary goal of the four subcontract studies was to reduce uncertainty ranges on both performance and cost parameters and thereby increase the confidence in the maturity of SRM state-of-the-art. Accordingly, the studies were deliberately designed to partially overlap and provide cross-check verification of performance and cost consistency in the industry. By formulating the studies in terms of system-related tradeoff parameters, the SRM data are usable to configure and price virtually any large SRM needed for any space shuttle candidate system. The studies also provided input information to allow a rational formulation of SRM booster man-rating design criteria which at least qualitatively optimized safety and mission-success dictates as opposed to program costs. The results of the SRM design, performance, and cost studies are presented in sections 2 and 3.

1.3 STAGE SUBSYSTEMS APPROACH

Preliminary designs of the baseline vehicle stage subsystems were formulated, taking into consideration the primary system constraints, design conditions, and configuration arrangements. Potentially governing problem areas were analyzed with the purpose of generating data for use in trade studies to determine cost-effective design decisions. Concurrent with the stage configuration preliminary design and the SRM subcontractor studies updated man-rating design criteria were synthesized for SRM booster systems.

The stage subsystems were each preliminarily defined in the minimum detail necessary to determine crude estimates of weight. The weights were used, in conjunction with some historical data from Minuteman, Poseidon, and Titan SRM stage programs, to arrive at initial estimates of subsystem development and production costs. A second design analysis iteration was then accomplished to provide higher confidence weight

and cost estimates, which provide the basis for reasonably confident SRM stage design and program cost projections. The projections derived are conservative when compared with Titan III C/D SRM stage experience. Section 4 provides a summary of the stage subsystem studies.

1.4 SYSTEM EFFECTIVENESS ANALYSIS

The SRM and stage subsystem characteristics and costs were then analyzed from the point of view of optimizing total space shuttle program cost effectiveness. Also, the usefulness of SRM boosters in achieving the prime goal of lowering peak annual funding requirements was partially evaluated. This analysis necessitated the assembly of reasonably valid SRM booster program development plans which were consistent with compatible orbiter program concepts, and reflected proper treatment of the man-rating criteria being crystallized.

The system effectiveness analyses are summarized in Section 5. Also included therein is a review of system performance and cost uncertainties, so that SRM booster system technical and cost risks could be appraised.

REFERENCES

1-1 Lockheed Missiles & Space Compnay, Study Plan for Extension of Alternate Space Shuttle Concepts for Contract NAS 8-26362, LMSC-A990581D, 30 June 1971

1-2 Lockheed Missiles & Space Company, Engineering Memorandum LH-08-02-M1, Statement of Work for SRM Contractors, 3 September 1971

Section 2
SOLID ROCKET MOTOR CHARACTERISTICS

The bulk of the large solid motor technology has been developed by four major propulsion companies. A working relationship was established and maintained with these companies throughout this study to ensure that the SRM data being used were current and valid and represented the judgement of the most knowledgeable sources. The four companies were funded over the last half of this program to generate and provide to this study specific design and cost parametric data that could be used to establish realistic SRM motor designs and their respective development and recurring costs.

A primary goal of the four subcontract studies was to reduce uncertainty ranges on both performance and cost parameters and, thereby, increase the confidence in the maturity of SRM state-of-the-art.

2.1 MOTOR CHARACTERISTICS FOR THE INTERIM SRM BOOSTER CONFIGURATIONS

2.1.1 Source of Motor Design Data

At the beginning of the Interim Booster Study, requests for SRM data from in-house and contracted studies on large solid motors from 120 in. to 260 in. diameter were sent to the four propulsion companies who have been active in large solid rocket developments. These were: (1) Aerojet General Corporation which has performed the bulk of the exploratory development effort on the 260 in. diameter motors, including three test firings; (2) Lockheed Propulsion Company and (3) Thiokol Chemical Corporation, which have conducted the development efforts on the 156 in. diameter motors including 9 full-scale firings; and (4) the United Technology Corporation, which developed and is producing 120 in. diameter segmented-case motors for the TITAN III C/D. UTC has fired a total of 15 full-scale motors in the development and PFRT program for the operational 5-segment motors, and has

produced 82 motors to date with additional production being negotiated. In addition, they have fired 4 full-scale motors in a 7-segment motor development program.

The data obtained from these companies were used to establish initial values of motor mass fractions (propellant weight divided by total motor weight), specific impulses and motor lengths for use in the initial synthesis and analysis of candidate space shuttle vehicle configurations.

The second cycle proceeded from the results of the initial vehicle configuration studies in which five configurations were selected as candidates for further study. The preliminary SRM requirements for these configurations were then sent to the propulsion companies with requests for preliminary designs or evaluations of motors to fulfill the requirements (Ref. 2-1). For each configuration, the rocket motor requirements were identified as to number of motors, propellant loads, maximum thrusts, burn times, and thrust vs. time shape. The five configurations included three single-stage booster configurations using 120, 156 and 260 in. diameter motors, and two double-stage booster configurations using the 120 and 156 in. diameter motors.

Responses were received during the second month of this study from three of the companies with newly generated point design data for the various configurations. Table 2-1 presents a summary of these data along with the LMSC-specified requirements for each of the 7 motors representing the 5 booster configurations. As might have been anticipated, there was good agreement on delivered specific impulse. However, a comparison of the mass fraction values showed considerable disparity. Some of the differences were due to different design values used by the companies for MEOP, port-to-throat area ratios and maximum structural design stress as noted in Table 2-1. Company B's mass fraction values were adjusted to the same MEOP and design stress as used by the other two, but significant differences still remained. To examine this problem further, the inert weights of the major motor components were compared in Table 2-2. Comparison of these data show that companies C & D were in close agreement. Company B's weights, even after applying corrections to base the weights on the same MEOP and design stress as C & D, were generally lighter.

Table 2-1

SUMMARY OF PROPULSION COMPANY MOTOR DATA: 12 AUGUST 1971

REQUIREMENTS	I	II	III	IV	V	VI	VII
Motor Diameter	120	120	120	150	150	150	260
Stage/No. of Motors	1--9	1--5	2--2	1--4	1--2	2--1	1--1
Propellant Wt/Motor	560K	514K	514K	1,089K	1,158K	1,158K	3,802K
Maximum Thrust/Motor	1,035K (SL)	1,410K (SL)	1,588K (VAC)	2,020K (SL)	3,360K (SL)	3,630K (VAC)	7,070K (SL)
Burn Time, sec	149	93	105	143	90.5	102.5	147
Maximum P_c, psi	750	675	675	500	1000	1000	620
ϵ, 17.5 deg Half Angle	8.0	8.0	9.0	6.8	8.75	8.75	7.0
I_{SP}, Del Vac; All Companies	265-266.5	264.5-266	267.5-269	262-263.5	267.5-268.5	267.5-268.5	262-265.5
MEOP/Port-to-Throat Area Ratio							
Company D	-	-	-	603/1.3	1205/1.3	1205/1.3	749/1.3
Company B	780/1.2	707/1.2	702/1.2	522/1.2	1043/1.2	1038/1.2	670/1.2
Company C	900/2.0	816/1.44	816/1.44	600/1.3	1200/1.3	1200/1.3	-
MASS FRACTION							
Co. D (200K Design Stress)	-	-	-	0.922	0.844	0.881	0.900
Co. B (210K Design Stress)	0.929	0.926	0.928	0.941	0.910	0.912	0.936
Co. B Corrected to AGC/UTC MEOP and 200K Design Stress	0.918	0.916	0.917	0.933	0.896	0.898	0.929
Co. C (200K Design Stress)	0.880	0.893	0.890	0.923	0.870	0.869	-
OVERALL LENGTH, INCHES							
Company D	-	-	-	1335	1367	1372	1746
Company B	1036	1054	1038	1307	1329	1329	1689
Company C	1136	1074	1087	1316	1341	1341	-

Table 2-2

COMPARISON OF MAJOR COMPONENT INERT WEIGHTS

Motor Config.	Company	Case, Igniter, & Misc. Weight	Insulation Weight	Nozzle Seal, & Actuator Weight	Total Inert	Mass Fraction	MEOP
I (120 in.)	B	33,050	3,330	6,550	42,930	0.929	780
	C	50,129	15,919	10,269	76,317	0.880	900
	B*	40,000	3,800	6,050	49,850	0.918	900
II (120 in.)	B	30,240	2,650	8,230	41,120	0.926	707
	C	41,564	10,156	9,934	61,654	0.893	816
	B*	36,650	3,020	7,600	47,270	0.916	816
III (120 in.)	B	29,360	2,790	7,960	40,110	0.928	702
	C	41,564	10,607	11,553	63,724	0.890	816
	B*	36,000	3,180	7,350	40,530	0.917	816
IV (156 in.)	B	49,310	5,080	13,750	68,140	0.941	522
	C	57,182	16,887	16,759	90,828	0.923	600
	D	55,670	17,472	19,180	92,322	0.922	603
	B*	59,700	5,800	12,700	78,200	0.933	600
V (156 in.)	B	95,260	4,640	15,160	115,000	0.910	1,043
	C	129,200	20,874	22,665	172,739	0.870	1,200
	D	113,563	16,524	22,494	152,571	0.884	1,205
	B*	115,300	5,290	14,000	134,590	0.896	1,200
VI (156 in.)	B	← Similar to V →					
	C						
	D						
	B*						
VII (260 in.)	B	204,390	11,400	42,640	258,430	0.936	670
	D	298,026	69,013	55,476	422,515	0.900	748
	B*	240,000	13,000	39,400	292,400	0.929	670

NOTE: B* represents B data corrected to 200,000 psi design stress (from 210,000 psi) and to MEOP indicated:

$$\text{Case Wt*} = \text{Case Wt} \times \frac{2.1}{2.0} \times \frac{\text{MEOP*}}{\text{MEOP}}$$

$$\text{Insulation Wt*} = \text{Insulation Wt} \times \left(\frac{\text{MEOP*}}{\text{MEOP}}\right)^{0.8}$$

$$\text{Nozzle Wt*} = \text{Nozzle Wt} \times \left(\frac{\text{MEOP}}{\text{MEOP*}}\right)^{0.73} \times \frac{1 + 2.1/2.0}{2}$$

Closest agreement was in the inert weight of cases with corrected values ranging from agreement with C & D values to about 20 percent lighter. Significant disagreement occurred for nozzle, seal, and actuator weights with Company B's values running 20 to 35 percent lighter. A gross disagreement occurs in the case of insulation weights with Company B's values running a factor of 3 to 6 times lighter. Discussions with the propulsion companies regarding the insulation calculations evoked responses from Companies C & D that their values were conservative and from Company B that theirs was optimistic. Since the insulation weight represents something less than 2 percent of the total motor weight, further investigation was not pursued.

2.1.2 Normalizing and Averaging of Design Data

A study was made to correlate and parametrically relate the point design motor data received from the propulsion companies. Parametric curves were established for overall rocket motor length, case weights, insulation weights, and nozzle assembly weights as a function of major design parameters. The derivation of these curves is presented in Ref. 2-2.

Although the accuracy of the absolute values of the motors thus characterized might be questionable, the accuracy of the different size motors relative to each other was quite adequate for the purpose of studying different booster configurations representing different diameter motors, numbers of motors, one stage vs two stage, and booster stage related variables.

Figure 2-1 presents the composite data on delivered vacuum specific impulse supplied by the propulsion companies. Data from Companies A, B, and C represent a PBAN (polybutadiene/acrylic acid/acrylonitrile) type propellant. This propellant is well characterized and can be readily processed with an aluminum loading of 16 to 18 percent and a total solids content of 84 to 86 percent. Company D based their data on a propellant currently under development; namely, HTPB (hydroxyl-terminated polybutadiene). This propellant has several advantages; the more significant ones include a possible higher solids loading of 90 to 92 percent, giving an increase in density of about 4 percent without sacrificing processability, and a potential lower cost. (A discussion of the characteristics and technological status of HTPB is presented in Reference 2-2.) For this study, the well-developed PBAN propellant was selected. The dashed line in Fig. 2-1 represents the delivered vacuum specific impulse values that were assumed.

2.1.3 Description of Motors Selected for Final Interim SRM Booster Configuration

The Interim SRM Booster Configuration studies identified five candidate configurations. Configurations 5A and 5D were based on 120 in. diameter motors; Configurations 5, 5B (with 2 motor cluster versions), and 5C used the 156 in. diameter motors. Booster propulsion requirements were specified as to total propellant load, maximum total thrust, number of motors per stage, and thrust vs time shaping for a single-stage booster and for each stage of a two-stage booster. Figure 2-2 presents the required thrust vs time profiles which were determined to limit maximum dynamic pressure and maximum acceleration during booster operation.

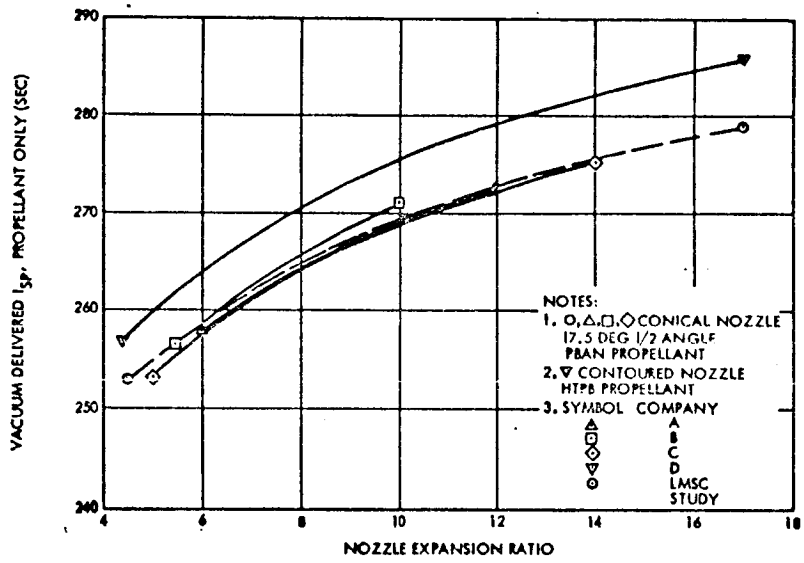

Fig. 2-1 SRM Specific Impulse Comparison

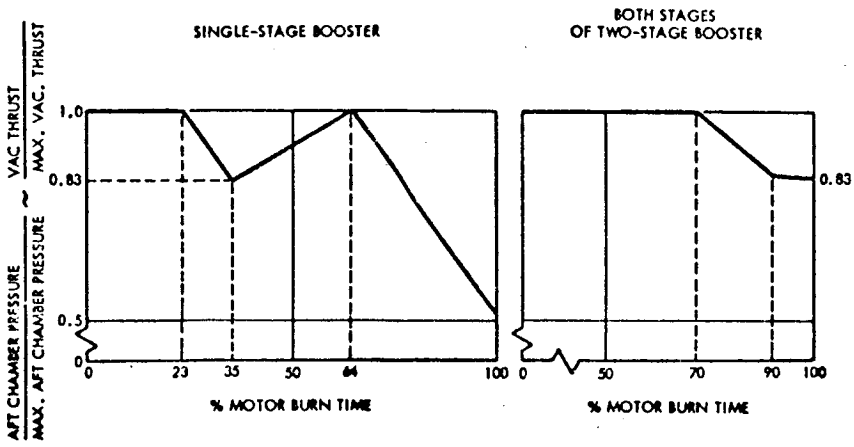

Fig. 2-2 Motor Thrust-Time Profile

In determining motor designs to fulfill these requirements, design criteria and constraints, based on information from the propulsion companies and from previous studies, were imposed as follows:

(1) A maximum aft chamber stagnation pressure (and MEOP) was selected to permit an expansion ratio of about 10 without the nozzle exit exceeding the motor case diameter.

(2) A minimum grain port-to-throat area for 156 in. diameter motors of 1.3 for MEOPs of 950 psi, and 1.5 for MEOPs of 1200 psi. For 120 in. diameter motors, a web thickness to motor case radius of 0.6, which corresponds to the current production tooling, was used.

(3) PBAN-type propellant with a 16 percent aluminum and an 85 percent solids loading and an average density of 0.064 lb/cu in. Delivered vacuum specific impulse was as depicted in Fig. 2-1. Minimum and maximum grain burn rates at 1000 psia were 0.30 and 0.70 in. per sec, respectively.

(4) An ultimate working stress of 200,000 psi for the case and nozzle structural material. This is obtainable with D6ac steel, the material used in the TITAN III C/D solid rocket motors, and is also readily obtainable with 18 percent nickel maraging steels while maintaining good impact properties.

(5) Asbestos/silica filled Buna-n rubber for case insulation and conventional carbon phenolic and silica phenolic tape wrap for nozzle insulation.

(6) A safety factor of 1.25 on the ultimate working stress for structural components, and a safety factor of 1.50 on insulation design.

Table 2-3 presents the propulsion requirements for the alternate Interim Boosters and the calculated ballistic performance characteristics for the respective motor designs. Table 2-4 presents the calculated dimensional and weight data for the motor designs using the correlation design curves described in Section 2.1.2.

Table 2-3

SRM PERFORMANCE CHARACTERISTICS OF INTERIM BOOSTER CONFIGURATIONS

Vehicle Config.	5A	5D		5B-1	5B-2	5C	5	
SRM Dia., In.	← 120 →			← 156 →				
Stage	1st	1st	2nd	1st	1st	1st	1st	2nd
Motors/Stage	9	5	2	4	3	4	2	1
Prop't Wt/Motor, 10^6 Lb	0.452	0.470	0.470	0.896	1.195	1.090	1.047	1.047
Max. Thrust/Motor, 10^6 Lb	0.875 (SL)	1.328 (SL)	1.485 (VAC)	1.745 (SL)	2.328 (SL)	2.140 (SL)	3.150 (SL)	3.425 (VAC)
Performance								
Type Propellant	← 16% AL PBAN, Density = 0.064 Lb/In.^{3} →							
Max. Aft Stag'n Pc, psi	750	750	750	750	750	750	1000	1000
Nozzle Expansion Ratio	10	9.4	9.4	10	9.2	9.8	9.3	9.3
Nozzle Expansion Half Angle, Deg.	15	15	15	17.5	17.5	17.5	17.5	17.5
Del. Spec. Impulse, SL, sec	239.7	240.3	240.3	239.7	240.5	239.8	247.5	247.5
Del. Spec. Impulse, VAC, sec	269.7	268.6	268.6	269.7	268.1	269.5	269.0	269.0
Burn Time, sec	149.5	103.0	103.0	140.5	141	139.7	85.2	85.2
Port/Throat Area Ratio	2.20	1.46	1.46	1.3	1.3	1.3	1.5	1.5
Web Fraction, %	60	60	60	67	62	64	59.6	59.6
Burn Rate at 1000 psi, In./sec	0.302	0.446	0.446	0.413	0.379	0.394	0.54	0.54
Nozzle Throat Dia., In.	31.0	38.0	38.0	44.4	51.5	49.2	51.3	51.3

Table 2-4

SRM DIMENSIONS AND WEIGHTS
OF INTERIM BOOSTER CONFIGURATIONS

Vehicle Config.	5A	5D		5B-1	5B-2	5C	5	
SRM Dia., In.	← 120			← 156				→
Stage	1st	1st	2nd	1st	1st	1st	1st	2nd
Motors/Stage	9	5	2	4	3	4	2	1
Prop't Wt/Motor, 10^6 Lb.	0.452	0.470	0.470	0.896	1.195	1.090	1.047	1.047
Max Thrust/Motor, 10^6 Lb.	0.875 (SL)	1.328 (SL)	1.485 (VAC)	1.745 (SL)	2.328 (SL)	2.140 (SL)	3.150 (SL)	3.425 (VAC)
Dimensions								
Case Length, In.	831	862	862	930	1250	1130	1100	1100
Case L/D; No Center Seg.	7.81;5	8.26;6	8.26;6	6.03;3	8.01;3	7.25;3	7.05;3	7.05;3
Total Motor Length, In.	937	992	992	1083	1415	1299	1266	1266
Nozzle Exit Dia. In.	98.2	120	120	141	156	154	156	156
Weights								
MEOP, psi	852	920	920	950	950	950	1200	1200
Case Material			D6ac					
Design Stress, psi			200,000					
Factor of Safety	← 1.25 on Structure; 1.50 on Insulation							→
Bare Motor Wt., 10^6 Lb*	0.50726	0.52694	0.52694	1.0062	1.3391	1.2210	1.1923	1.1923
Prop't Mass Fraction*	0.892	0.892	0.892	0.891	0.893	0.893	0.879	0.879
Primary Data Source	← UTC		→	← AGC, LPC, TTC, UTC				→

*Thrust Termination and TVC actuation system weights not included.

These designs include a flexible bearing on the nozzle joint to permit thrust vector control of ±8 deg. The TVC actuators and power supply are not included in the motor weights. The motor designs also include the capability for thrust termination. However, the weight of the external thrust termination stacks are not included.

Other requirements imposed by the interim SRM booster configurations on motor design were investigated by the propulsion companies and determined to be within the known state-of-the-art of motor technology. These were requirements for nozzle cant angles up to 20 deg, nozzle deflection angles up to ±15 deg, and thrust termination capability at any time during motor burn which would result in no or only slight negative thrust.

2.2 PARAMETRIC DESIGN DATA FROM SUBCONTRACTOR STUDIES

2.2.1 Description of Study Subcontracts to the Propulsion Companies

The third cycle of activity with the propulsion companies was formalized with subcontracts to each of the four participants (AGC, LPC, TCC, and UTC) with common work statements to prepare parametric design and cost data which would reflect the effects of design variables such as motor propellant weight, chamber MEOP, motor burn time, thrust vs time, and several cost-related variables. The statement of work is contained in Ref. 2-3. The design groundrules are summarized in Table 2-5. The data generated by the four contractors were constrained to a consistent set of design groundrules so that easy comparisons could be made. This effort was begun the last week of August with final reports due just prior to completion of this study phase (mid-November). Interim reports received during the course of these subcontractor studies resulted in preliminary presentations and comparisons of the data reported in Ref. 2-4, and the critique of the data also provided in Ref. 2-5. The scope of responses of the SRM contractors relative to the requested design parametrics is shown in Table 2-6.

Table 2-5

SPECIFIED DESIGN GROUNDRULES

Motor Diameter, In.	120, 156, 260
Motor Case L/D	5.0 to 10.0
Nozzle Exit Diameter, In.	Not to exceed motor dia.
TVC System	Flexible Seal Nozzle
Max TVC Deflection, deg	± 10
Max Nozzle Rate, deg/sec	15
MEOP, psi	800, 1000, 1200
Safety Factor	
Case Structure (Ultimate)	1.25
Insulation	1.50
Nozzle Ablator	1.50
Fwd & Aft Skirt Length	20% of motor dia.
Burn out Altitude, ft	
100 sec burn	80,000
135 sec burn	180,000
Guaranteed Std. Specific Impulse, sec (sea level, 0 Deg half angle)	249
Phase C/D Go-ahead	1 April 1972
Manned Flight Qualification Date	7/78

Table 2-6

SRM CONTRACTOR DESIGN DATA RESPONSE
(Based on Current Technology)

Contractor	Motor Diameter (in.)			Burn Time (sec)		MEOP (psi)			Regressivity	
	120	156	260	100	135	800	1000	1200	Mod	Min
A	Design Data Only	Yes	No	156 in. Only	Yes	Yes	Yes	Yes	Yes	156 in. Only
B	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	(Used their Standard Regressivity)	
C	Yes	Yes	No	Yes	Yes	No	Yes	No	Yes	Yes
D	No	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes

Table 2-7

MAXIMUM ALLOWABLE EXPANSION RATIOS

Motor Diameter (in.)	Contractor			
	A	B	C	D
120	*	10.0	No Limit	-
156	*	10.0	No Limit	17.0
260	--	10.0	--	9.0

*12 at 800 MEOP, 14 at 1000 MEOP, 16 at 1200 MEOP

2.2.2 Evaluation of Interim Submittals of Parametric Design Data

Data generated by the four contractors were in general responsive to the groundrules. The only divergence of importance was in the selection of nozzle expansion ratios. This deficiency is not critical, since it has second-order effect on mass fraction and the performance could be easily corrected by LMSC for those design points of interest. The expansion ratios used by the contractors either corresponded to a nozzle-exit diameter equal to the motor diameter, or to an arbitrary limitation at some expansion ratio. Table 2-7 presents the maximum allowable expansion ratios assumed.

A sample comparison of the data submitted for a motor corresponding to the propellant load requirements of vehicle configuration 5 (using a single-stage interim booster with motor MEOP = 1000 psi) is shown in Table 2-8. The vacuum I_{sp} comparison is based on an expansion ratio of 8.0. The values calculated by LMSC for the interim booster configuration definition prior to receipt of the SRM contractor data are shown in the last column for comparison purposes.

The differences between companies in data values for other motor design conditions follow:

- Contractor B consistently showed the highest mass fractions and Contractors C or D showed the lowest. The relatively large divergence in the mass fraction values of Contractor B was due in part to their use of a structural design stress of 220,000 psi compared to a nominal 200,000 psi used by the other contractors. The reasons for the remainder of the differences was not resolved. Detail design investigation at the next lower component level is required to determine the reasons for the divergence.
- Contractor D performance calculations were based on a contoured nozzle and the recently developed propellant, HTPB, which is more energetic than the PBAN-type propellants assumed by the others, which accounts for this contractor's higher values of specific impulse. Other contractors assumed a conical nozzle with a 17.5 deg divergence half angle.

Table 2-8

SRM PERFORMANCE CHARACTERISTICS COMPARISON

Motor Diameter (in.)	Propellant Weight (lb.)	Contractor				
		A	B	C	D	LMSC
		MOTOR MASS FRACTIONS				
120	0.5×10^6	0.896	0.916	0.880	--	0.892
156	1.0	0.904	0.918	0.903	0.902	0.893
260	3.5	--	0.918	--	0.913	0.910
		EXPANSION RATIO				
120	0.5×10^6	12.1	10.0	11.9	--	10.0
156	1.0	11.5	10.0	10.3	12.4	9.2
260	3.5	--	10.0	--	9.0	8.0
		Vacuum I_{sp} (sec) for $\epsilon = 8$				
Any	Any	265	266	264	270	265

The individual contractor inputs of mass fractions, expansion ratios, and specific impulses are summarized in Ref. 2-4. Figures 2-3 through 2-9 are plots of the data grouped according to motor sizes and design MEOPs.

Figures 2-3 through 2-5 compare the contractor data for an MEOP (maximum expected operating pressure in the chamber) of 1000 psia for the three motors (120, 156, 260 in. diameter). Differences in the mass fraction values between contractors are apparent; differences in expansion ratios and specific impulses do not account for the mass fraction differences. Figures 2-6 through 2-8 present similar comparisons, but for MEOPs of 800 and 1200 psi. The differences between contractor data for the MEOPs of 800 and 1200 psi are nearly identical to that for the MEOP of 1000 psi, so that little or no study is required of the 800 and 1200 MEOP plots.

Figure 2-9 shows the effect of MEOP on mass fraction for a typical motor for each of the three diameters. Excellent agreement between the contractors is observed. Inspection of the curves also shows about a 0.01 mass fraction change for a 200 psi change in MEOP. The vacuum specific impulse data that were submitted were essentially the same as obtained early in the study and which is presented in Figure 2-1.

2.2.3 Subcontractor Final Reports

Additional information in considerable detail on large solid rocket motor technology and design parametrics and procedures is contained in each of the final reports prepared by the four contractors (Refs. 2-6, 2-7, 2-8, and 2-9). Ballistic and physical properties of the PBAN and HTPB type propellants are described and discussed. Physical properties and fabrication considerations of structural steels are treated similarly. Criteria for grain design and case and nozzle insulation design are presented. Data and discussions are given on the design of TVC actuation and thrust termination systems. In addition, several point design motors are described with weight breakdowns and dimensional and material listings.

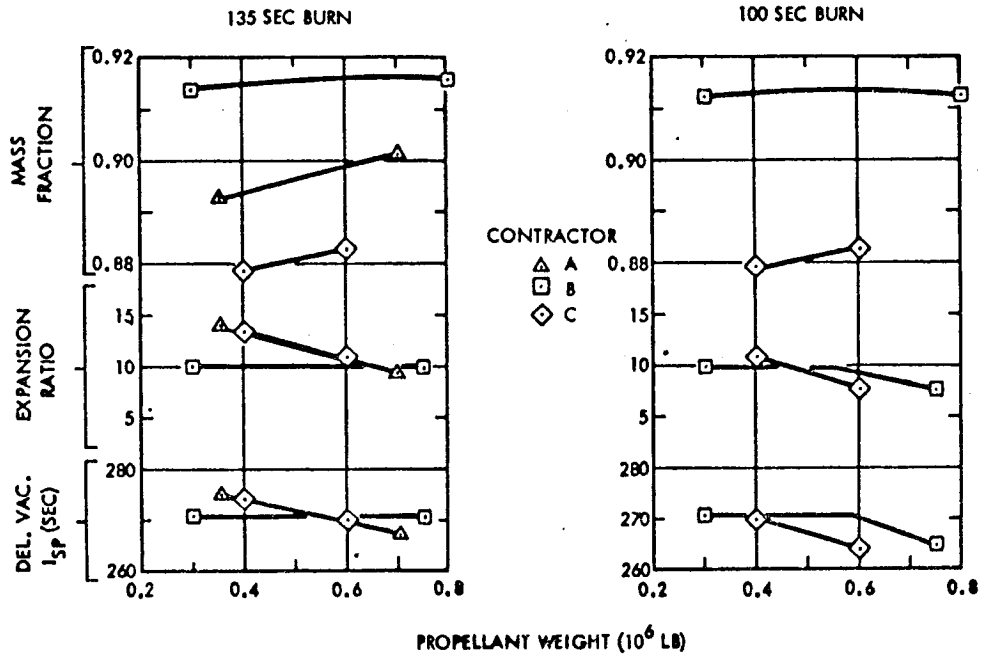

Fig. 2-3 Data Comparison For 120 In. Diameter Motors With 1000 MEOP

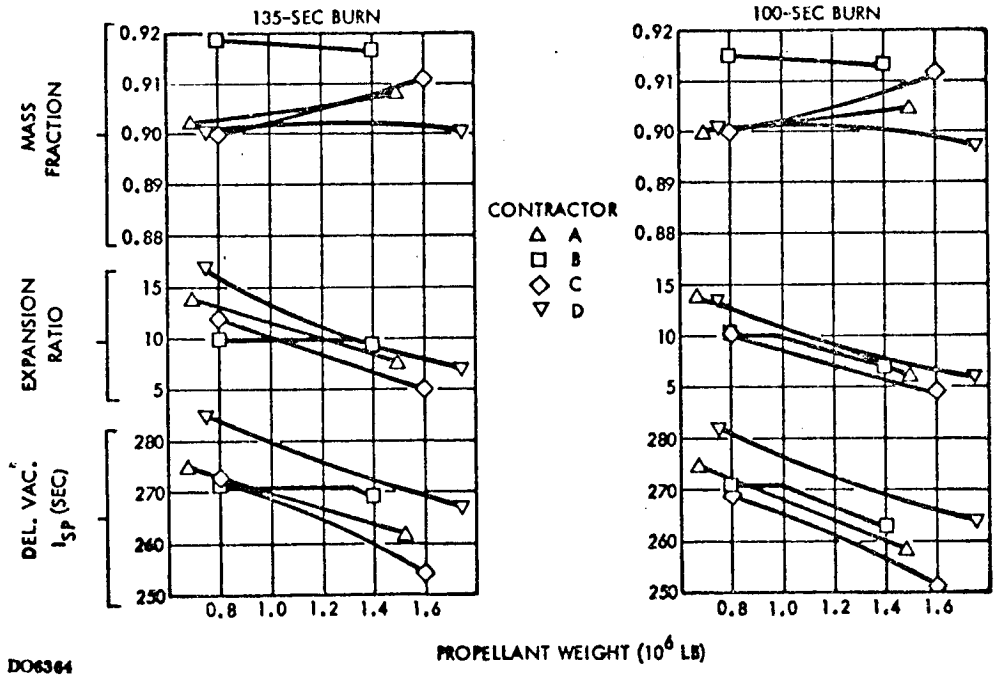

Fig. 2-4 Data Comparison For 156 In. Diameter Motors With 1000 MEOP

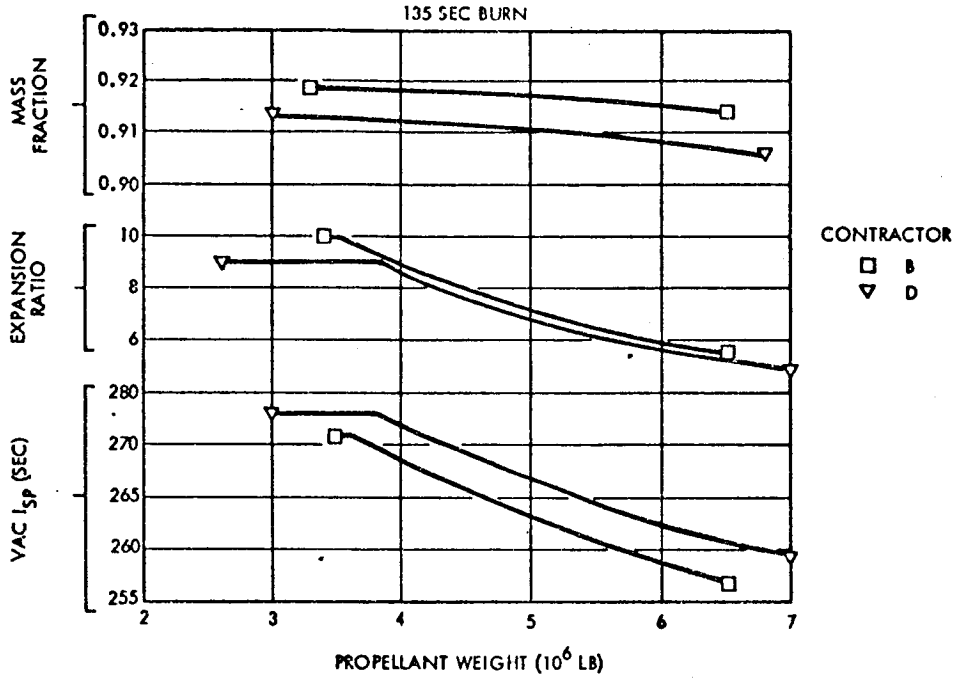

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Fig. 2-5 Data Comparison For 260 In. Diameter Motors With 1000 MEOP

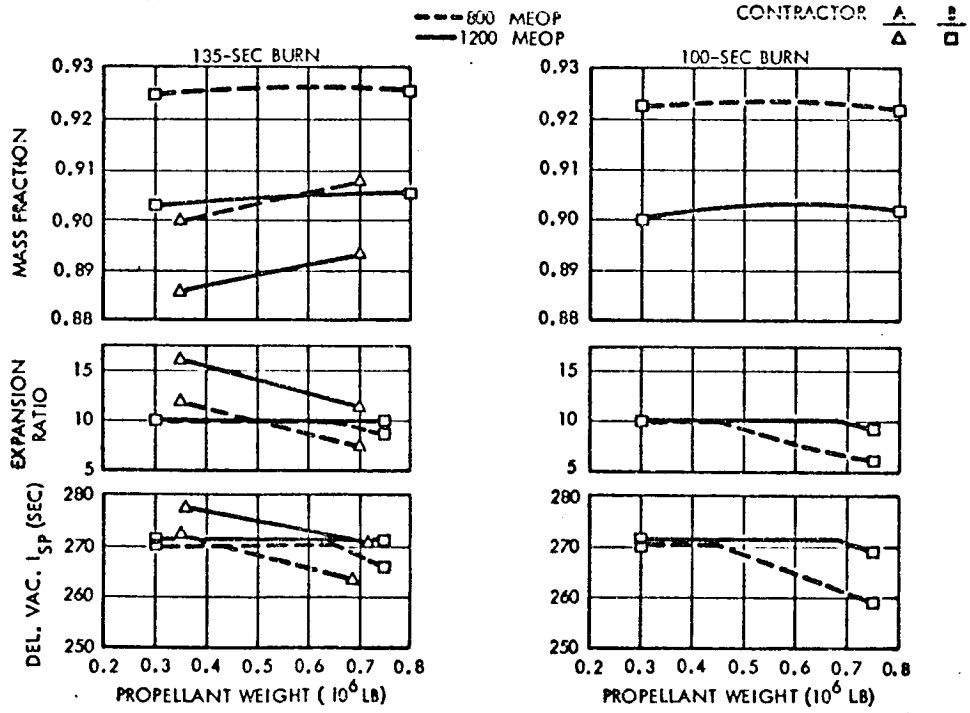

Fig. 2-6 Data Comparison For 120 In. Diameter Motors With 800 And 1200 MEOP

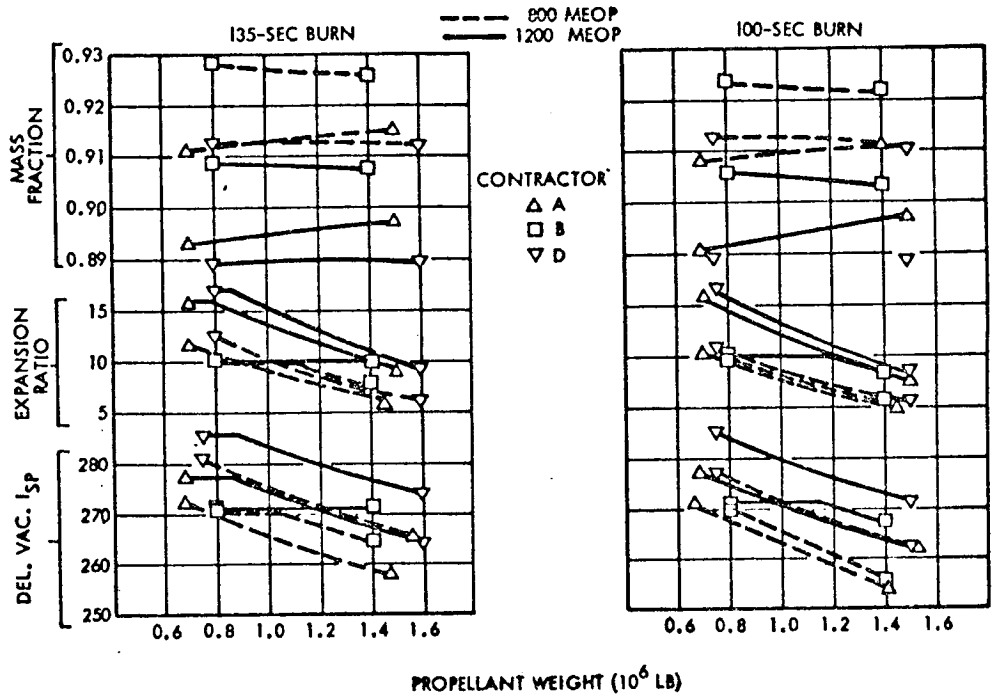

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Fig. 2-7 Data Comparison For 156 In. Diameter Motors With 800 And 1200 MEOP

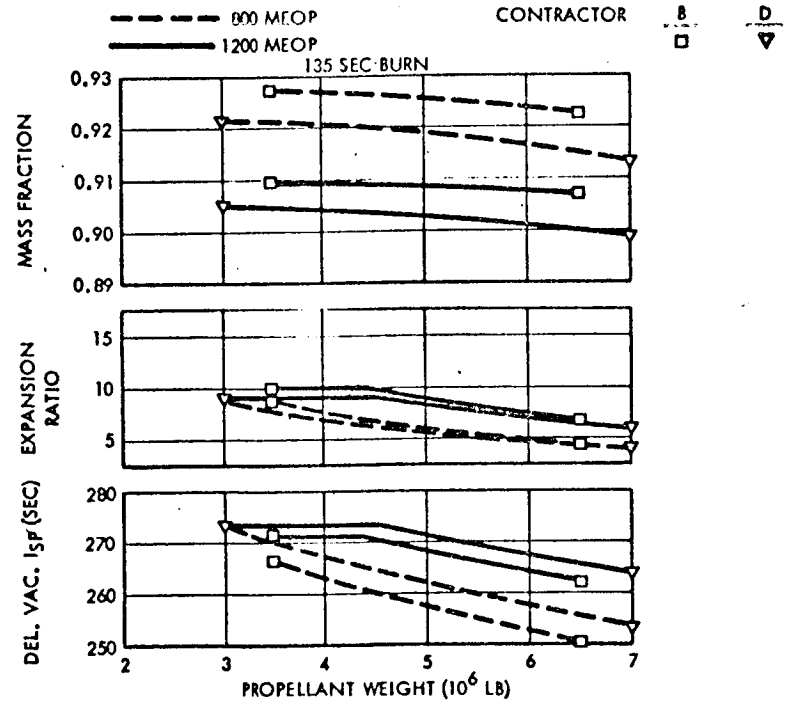

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Fig. 2-8 Data Comparison For 260 In. Diameter Motors With 800 And 1200 MEOP

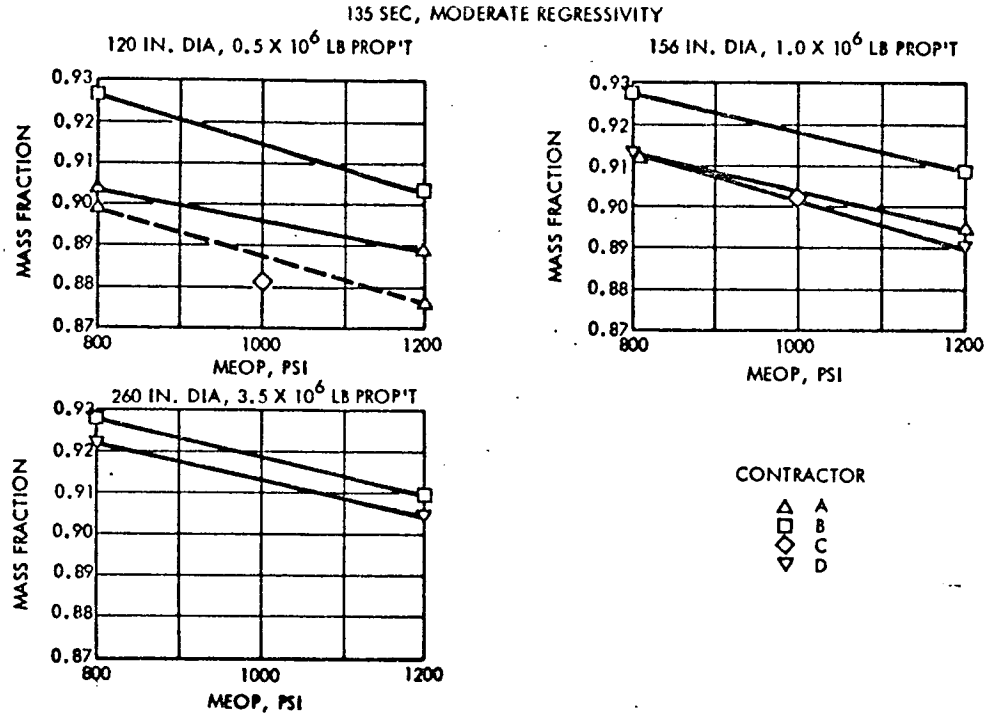

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Fig. 2-9 Effect of MEOP on Mass Fraction, 135 Sec, Moderate Regressivity

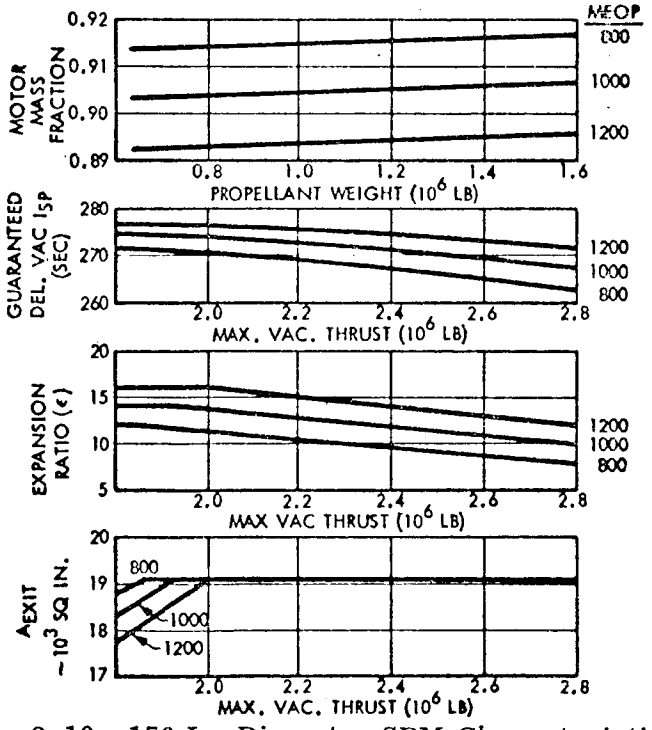

Fig. 2-10 156 In. Diameter SRM Characteristics

2.3 EFFECT OF MEOP AND EXPANSION RATIO ON BOOSTER PERFORMANCE

An analysis was conducted to determine the effect of design MEOP and nozzle expansion ratio on the design and performance of the motor for the interim booster. Using the data from the Interim Reports submitted by the propulsion companies on the subcontracted design and cost parametric studies, relationships were prepared showing the effect of MEOP on motor mass fraction and motor performance. These relationships are shown in Fig. 2-10 for the condition that the nozzle exit diameter equals the motor diameter. Computer sizing calculations were run to determine the effect on booster liftoff weight (BLOW) of motors designed for MEOPs of 600, 800, 1,000 and 1,200 psia. Studies were conducted on interim booster Configuration 5B with a single stage of three and four 156 in. diameter motors and on Configuration 5 with a first stage of two 156 in. diameter motors and a second stage of one motor. The results are shown in Fig. 2-11. The MEOP for the 3-motor single-stage booster optimizes at about 750 psi, and for the 4-motor single-stage at about 700 psi. A much higher optimum MEOP is indicated for the two-stage booster at about 1,050 psi. However, the principal result of the analysis is that the effect of MEOP is very slight on the total weight of the motors or BLOW. An MEOP that is 200 psi off optimum affects BLOW by only about 2-1/2 percent. These results agree with conclusions from the Saturn 1B Improvement Study (260 in. diameter SRM first stage) (Ref. 2-10) wherein the optimum MEOP was determined to be 770 psi.

The analysis of the effect of expansion ratio was made for a single-stage booster. The typical vehicle altitude-time trajectory for a single stage booster is shown in Fig. 2-12. For this trajectory, the average delivered specific impulse was calculated for different expansion ratios without regard to any maximum nozzle exit diameter limitation. The results are shown in Fig. 2-13 for three MEOPs. Again, these results agree with conclusions from the Saturn 1B Improvement Study by Douglas; for example, at an MEOP of 850 psi, the optimum expansion ratio is given as 12. The expansion ratios corresponding to exit diameter limitations equal to motor diameter are noted on these curves for 3- and 4-motor clusters. It can be observed that limiting the nozzle exit diameter (hence expansion ratio) to the motor diameter for the 3-motor cluster will result in an

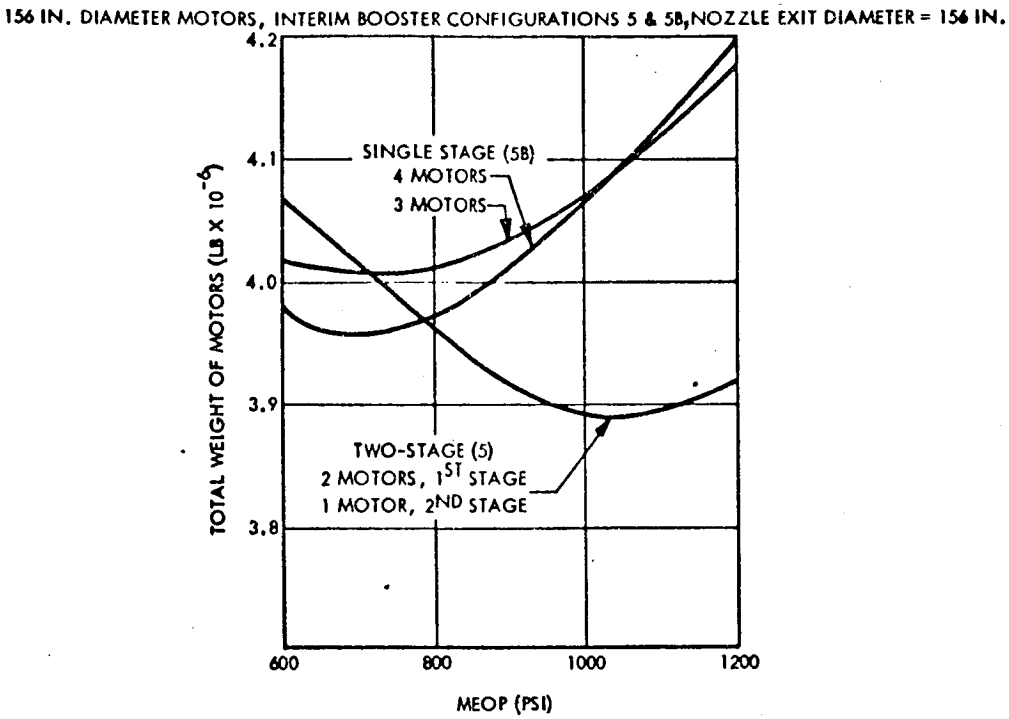

Fig. 2-11 Effect of MEOP on Motor Weights

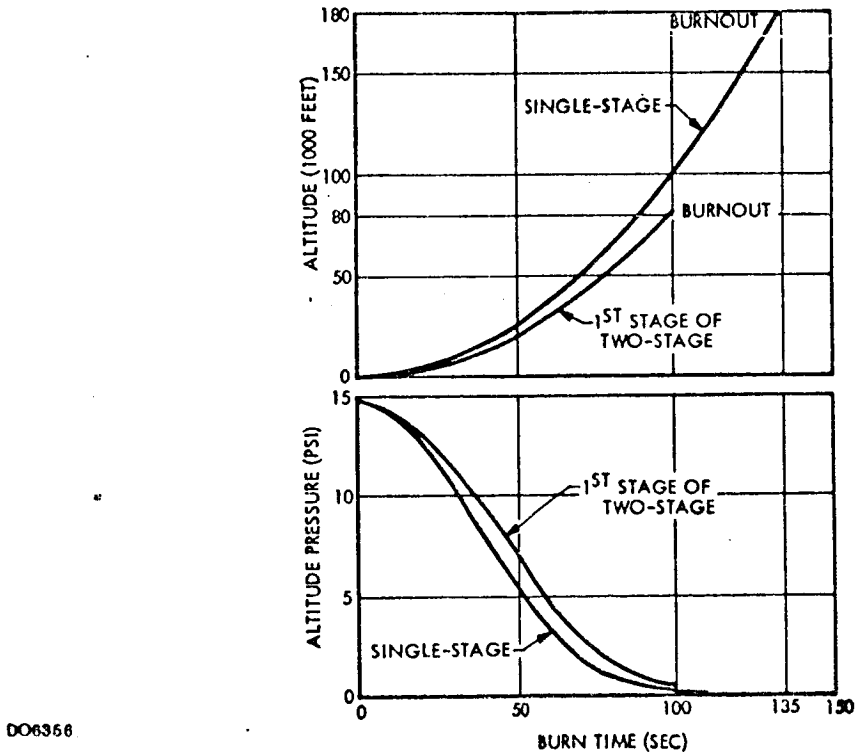

Fig. 2-12 Typical Altitude-Time History for Interim SRM Booster

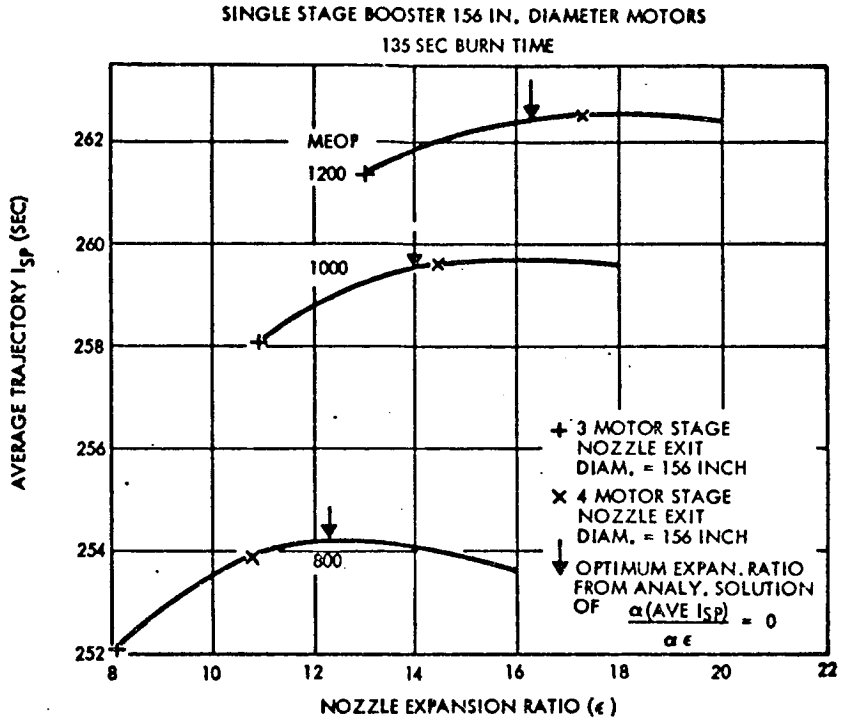

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Fig. 2-13 Effect of Nozzle Expansion Ratio

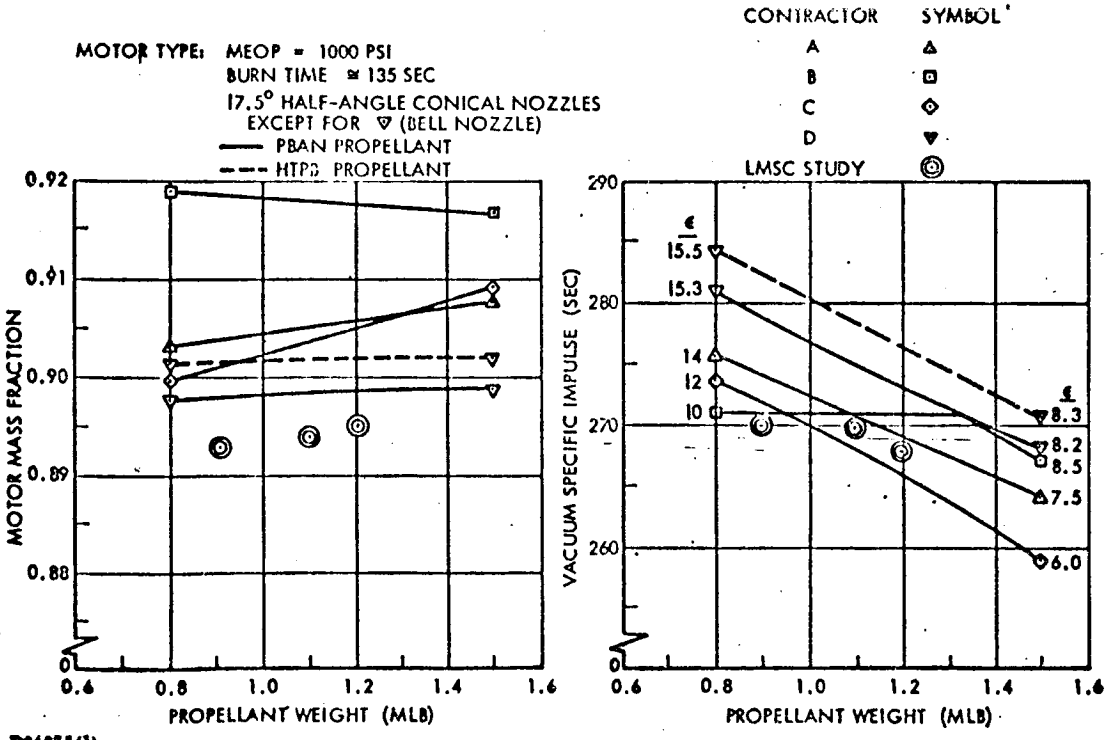

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Fig. 2-14 Baseline 156 In. SRM Performance Comparison

appreciable penalty to the average specific impulse of about 2 sec for the MEOP of 800, while the 4-motor cluster appears to be near optimum. For a typical single-stage SRM booster, one sec of specific impulse is equivalent to about 40,000 lb of booster liftoff weight. However, nozzle exit diameters larger than the motor diameter will include penalties in greater weight of the nozzle extension cone and the motor aft skirt and in added drag due to greater aft skirt flare. A more thorough and complete study is required to truly determine the best expansion ratio.

2.4 SRM DESIGN DATA CONCLUSIONS

2.4.1 SRM Subcontractor Studies

The SRM subcontractor studies have produced performance and design data on all three sizes of large solid rockets (120, 156, and 260 in. diameters) that are generally in remarkable agreement. Three of the subcontractors independently provided essentially identical characteristics, allowing for a rather narrow band of variation due to the absence of detailed specifications. One of the contractors provided appreciably more optimistic mass fractions due in large part to use of structural material with higher tensile strengths. The study results indicate that the proven and available state-of-the-art in SRM design technology can meet the propulsion requirements of the space shuttle booster, with very low risk levels in achieving performance and design goals.

2.4.2 Interim Booster SRM Characteristics

The results of the SRM subcontractor studies have shown that the motor characteristics selected for use in the interim booster configuration studies are conservative. This fact is displayed in Fig. 2-14, which is applicable to a single-stage booster consisting of three or four 156 in. diameter motors. The mass fraction values may be conservative by as much as 0.01, and the specific impulse by two or three sec if large expansion ratios can be used. In addition, the nominal MEOP of 950 to 1000 psi, which was selected for the baseline motors, may be about 200 psi above the optimum value which represents an additional conservatism in the LMSC interim booster study results.

For any future studies, these motor characteristics must be updated to represent motor design MEOP's and expansion ratios which are optimum from the standpoint of minimizing booster liftoff weight. Figure 2-10 is indicative of updated motor parametric characteristics that would be used for 156 in. diameter motors in a single-stage booster where nozzle exit diameter is limited to motor diameter. In addition, further investigation and consideration should be given to HTPB propellants which are currently being developed for on-going programs in the Department of Defense. This propellant shows further improvements to be obtained in motor specific impulse and mass fraction which would result in additional lowering of the vehicle liftoff weight and cost.

REFERENCES

2-1 LMSC Letter A1993344 dated 7 Aug 1971 and LMSC Wire Message A1993345 dated 7 Aug 1971 entitled, "Request for Preliminary Evaluation of Tentative Baseline SRM Configurations Used in NASA Space Shuttle Extension Study"

2-2 LMSC EM L4-08-02-M5, "SRM Design and Performance Data for Space Shuttle Interim Booster Alternatives", 8 Nov 1971

2-3 LMSC EM L4-08-02-M1, "Statement of Work for SRM Contractors", 3 Sep 1971

2-4 LMSC EM L4-08-02-M2, "Preliminary Evaluation of Design Data From SRM Contractors," 23 September 1971

2-5 LMSC EM L4-08-02-M4, "Critiques of SRM-Contractor Preliminary Data", 6 Oct 1971

2-6 Lockheed Propulsion Company Report 607-F, "Space Shuttle SRM Design and Cost Study", Nov 1971

2-7 Thiokol Chemical Corporation, Wasatch Division, No Report Number, "Cost and Performance Assessment of Large Solid Rocket Motors for Lockheed Missiles & Space Company", 22 Oct 1971

2-8 United Technology Corporation Report Number 2436-71-1, "Space Shuttle Booster Parametric Design and Cost Data," 15 Nov 1971

2-9 Aerojet Solid Propulsion Company Report Number 1873-02F, "Space Shuttle SRM Design Vs Cost Study", 12 Nov 1971

2-10 Douglas Missile and Space Systems Division Report Number SM-51896 Vol I, "Saturn 1B Improvement Study (Solid First Stage) Phase II, Final Summary Report", 30 March 1966

Section 3
SOLID ROCKET MOTOR COSTS

3.1 COMPARISON OF SUBCONTRACTOR COST DATA

3.1.1 Cost Study Groundrules

Cost data for solid rocket motors (SRM) were prepared as part of the subcontracted studies by Aerojet (AGC), Lockheed Propulsion (LPC), Thiokol (TCC), and United Technology (UTC), in response to the Statement of Work (SOW) contained in Ref. 1*. Interim evaluation of the data was reported in Ref. 2*. Subcontractor final reports were submitted on or before 15 Nov 1971. Design variables and cost groundrules, which were to be used in generating the data, are given in the SOW and are summarized in Table 2-5 of Section 2.2.

The preliminary responses indicated several inconsistencies between the contractors in the cost items included and the accounting methods for the various cost items. Additional instructions and clarifications were given to the contractors in order to have costs presented on a consistent basis in their final reports and in the LMSC ACS final report.

The four contractors were instructed to generate their data in accordance with a set of groundrules designed to facilitate the use of the data in space shuttle trade studies and to validate SRM costs. Parametric development and recurring cost plots were requested for a range of loaded propellant weights for specified values of motor design MEOPs, burn times, and thrust-time shapes. In addition, development costs were

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*References:

1. EM No. L4-08-02-M1, "Statement of Work for SRM Contractors," dated 3 Sep 1971.
2. EM L4-08-02-M3, "Preliminary Evaluation of Cost Data From SRM Contractors," dated 31 Oct 1971.

requested for 8-motor and 12-motor development programs, and recurring costs for a motor-delivery rate equivalent to 20 million and 40 million pounds of propellant per year for three years. Other applicable groundrules were specified as follows:

a. Costs to include all elements of the motor except for:
   - Thrust vector control (TVC)
   - Stage separation systems
   - Structural interstage or mounting components that are not integral with the SRMs
   - Second-stage application roll-control system

b. Costs to be based on these general guidelines:
   - Phase C/D Go-Ahead assumed for 1 April 1972
   - All cost data derived in 1970 dollars
   - Costs provided exclusive of any fee
   - Include all new NASA facility costs for brick and mortar, as well as operation and maintenance of new facilities (contractors to provide these costs as a separate line item, when detail costing is shown)
   - All Phase B costing groundrules utilized, except where modified herein
   - The unmanned test flight program, as required, to be completed and the vehicle qualified for manned flight by July 1978

c. For this study, motors for use in the unmanned flight qualification tests were assumed to be included in the recurring production costs. Development costs to include design, component developments and tests, tooling, fabrication, handling equipment, new facilities (for development and test), tests, support engineering, program management, and documentation.

d. Production costs to include program management and support engineering, tooling and facilities (beyond that included in development), fabrication, inspection, GSE, handling, transportation, storage, assembly, and launch support.

e. In estimating launch-base operational support in the recurring costs, each launch to assume a number of motors, which roughly corresponds to 4 million pounds of propellant.

3.1.2 Analysis of Interim Data

The contractors' responses relative to the requested matrix of parametric cost data are shown in Table 3-1. In total, an approximate 60 percent response was obtained, which is quite satisfactory in view of the short schedule and limited funding.

Table 3-1

PARAMETRIC COST DATA RESPONSE

CONTRACTOR	MOTOR DIAMETER (IN.)			BURN TIME (SEC)		MEOP (PSI)			FACILITIES COST	TRANSPORTATION EQUIP AND OPERATIONS	GROUND SUPPORT EQUIPMENT	LAUNCH FACILITY SUPPORT
	120	156	260	100	135	800	1000	1200				
A	NO	YES	NO	YES	YES	YES	YES	YES	YES	NO	•	•
B	YES	YES	YES	YES	YES	YES	YES	YES	•••	••	NO	NO
C	YES	YES	NO	YES	YES	NO	YES	NO	NO	NO	•	•
D	NO	YES	YES	NO	YES	YES	NO	NO	NO	••	••	NO

•INCLUDED IN COST DATA BUT NOT ITEMIZED
••DISCUSSED BUT NOT INCLUDED IN COST DATA
•••FACILITIES COST INCLUDED FOR 120 AND 156 INCH MOTORS BUT ONLY DISCUSSED FOR 260 INCH MOTOR

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As shown, cost data were submitted by: (1) two contractors for 120-in. diameter motors, (2) all contractors for 156-in. diameter motors, and (3) two contractors for 260-in. diameter motors. Also, examination of Table 3-1 shows the extent of the parametric data generated, which varied from nearly complete response by contractor B to a single set of motor-design characteristics for the 156-in. and 260-in. diameter motors by contractor D. In addition, there was (in the initial input) considerable inconsistency in the handling of cost items such as amortized facility, tooling, GSE, and transportation; facility and hardware; and support items in the area of motor transportation and launch facility assembly and checkout. In regard to the latter costs, additional instructions were issued to the contractors in order to standardize the treatment and inclusion of these costs in the final working data input due late in October.

A summary comparison of the submitted preliminary cost data is given in Fig. 3-1 for motor design requirements corresponding to a single SRM booster stage. Very good agreement exists for the recurring (production) program costs; variations in the development costs were within plus or minus $10 million of the norm. The individual contractor

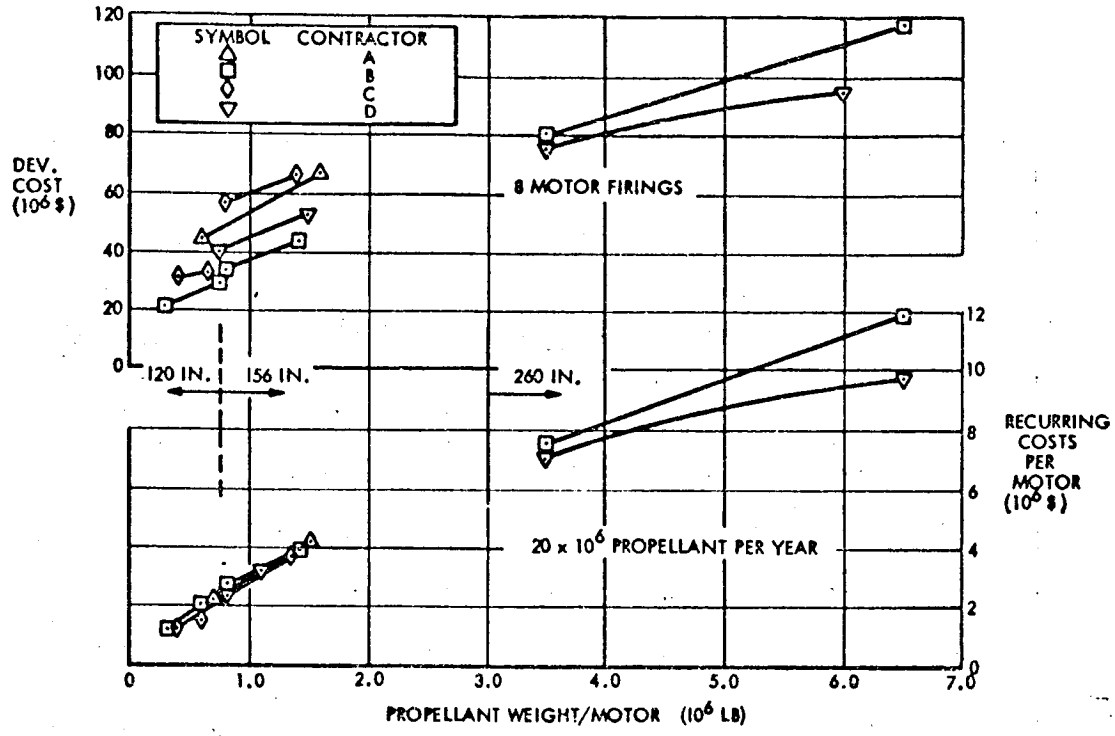


Fig. 3-1 Contractor SRM Cost Comparison Summary

inputs, presented in motor/size groupings, are summarized and discussed in Ref. 2. Comparisons of these preliminary data are presented in Figs. 3-2 through 3-5.

The comparison of the 120-in. diameter motor costs in Fig. 3-2 shows contractor C with the highest development costs but the lowest recurring costs. A probable reason for this is that contractor C included much of the facility, tooling, and GSE equipment costs in the development costs. Therefore, the differences should become much less when appropriate reallocations are made.

The comparison of the 156-in. diameter motor costs in Figs. 3-3 and 3-4 shows very good agreement on recurring costs. The larger spread in development costs will narrow when the contractor's costs are realigned. However, contractor B's costs appear to be unrealistically low and more detailed reevaluation of their data should be made in future studies. All contractors were urged to analyze and refine their cost data on the 156-in. diameter motors, since these appear to be of greatest interest in current space shuttle studies, and this is expected to eventually result in better

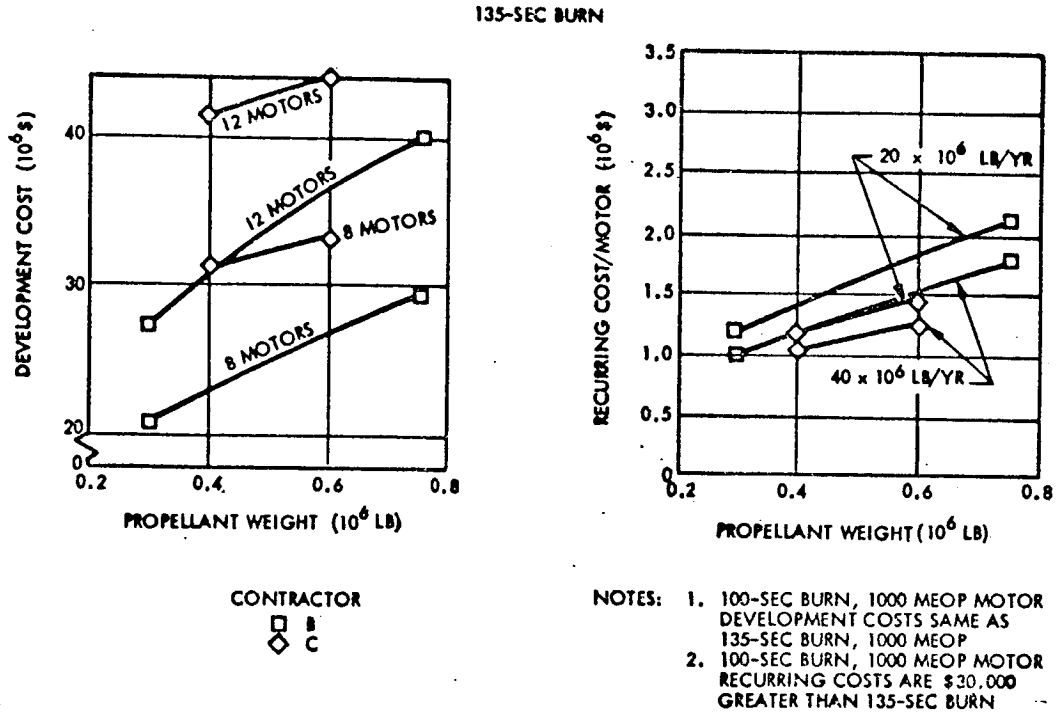


Fig. 3-2 Cost Comparison of 120-in. Diameter Motors at 1000 MEOP

agreement on development costs that currently exists. If the currently indicated trends continue, contractors A and/or C will be presenting the higher than nominal development costs, whereas contractor B will have lower than nominal costs.

The comparison of the 260-in. motor costs in Fig. 3-5 exhibits an ambiguous relationship between the data of contractors B and D. Although contractor B's costs are lower than contractor D's for the 156-in. diameter motors, the role is reversed for the 260-in. motors. This fact was brought to the attention of both companies for consideration when they reviewed their respective data for use in any future study.

Several general conclusions can be drawn relative to the affect on costs in relation to motor design parameters. Motor-burn time and thrust regressivity appear to have negligible effect. On the other hand, MEOP does have an appreciable effect. The additional cost of development and production of a 1200-psi MEOP design over an 800-psi MEOP design is approximately 7-to-9 percent for the 120-in. diameter motors, and approximately 4-to-6 percent for the 156-in. and 260-in. diameter motors.

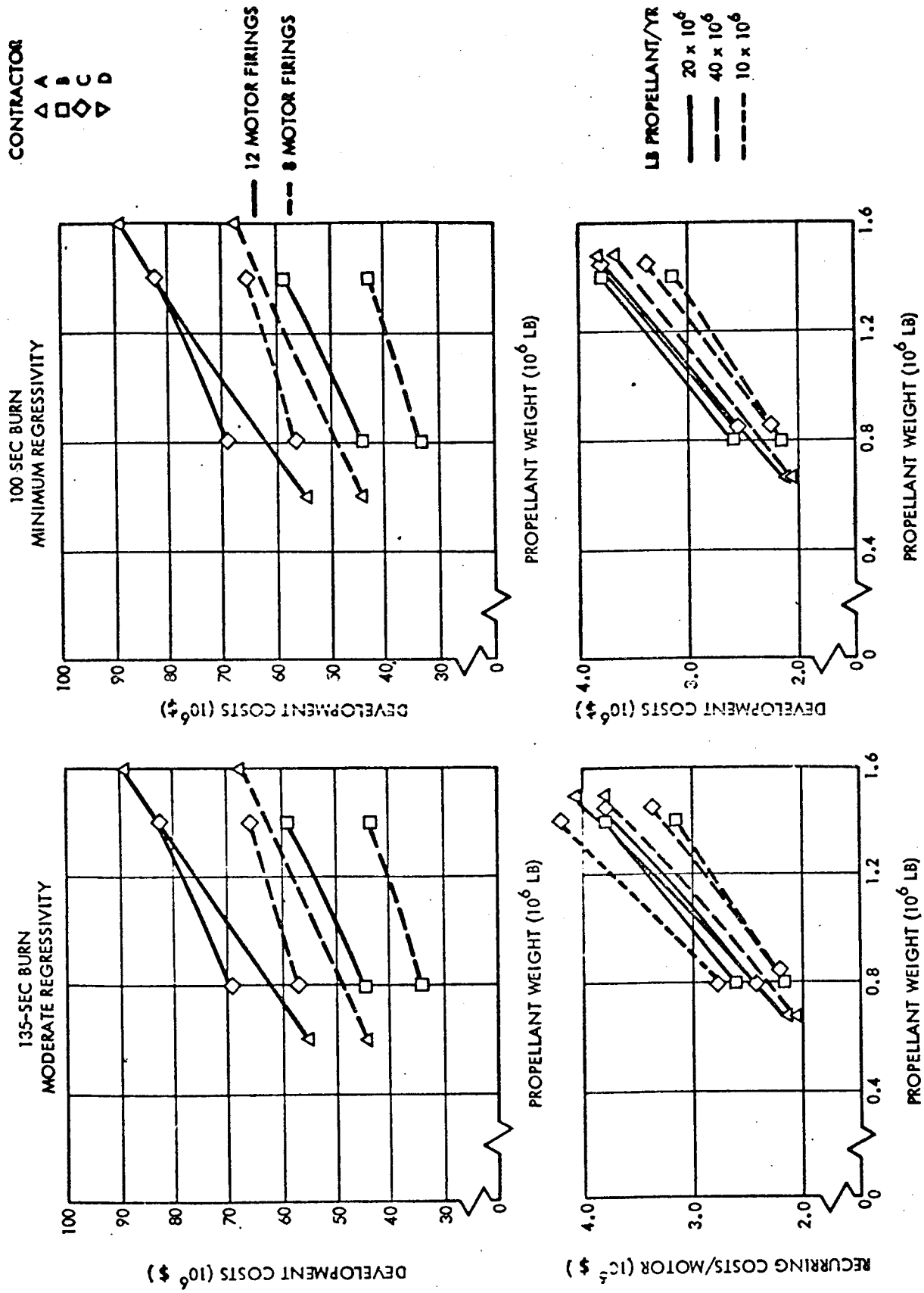


Fig. 3-3 Cost Comparison of 156-In. Diameter Motors for 1000 MEOP

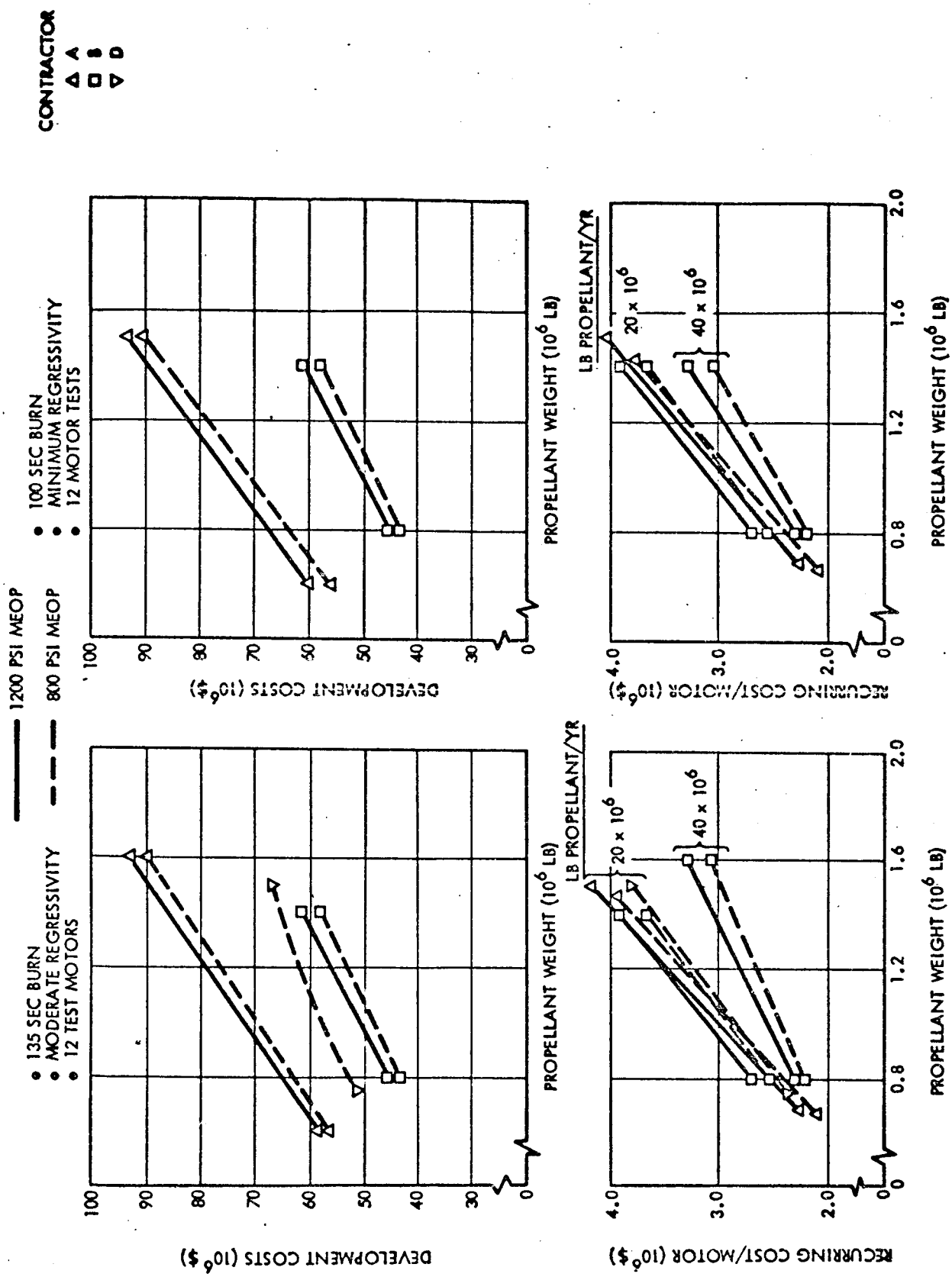


Fig. 3-4 Cost Comparison of 156-In. Diameter Motors for 800 and 1200 MEOP

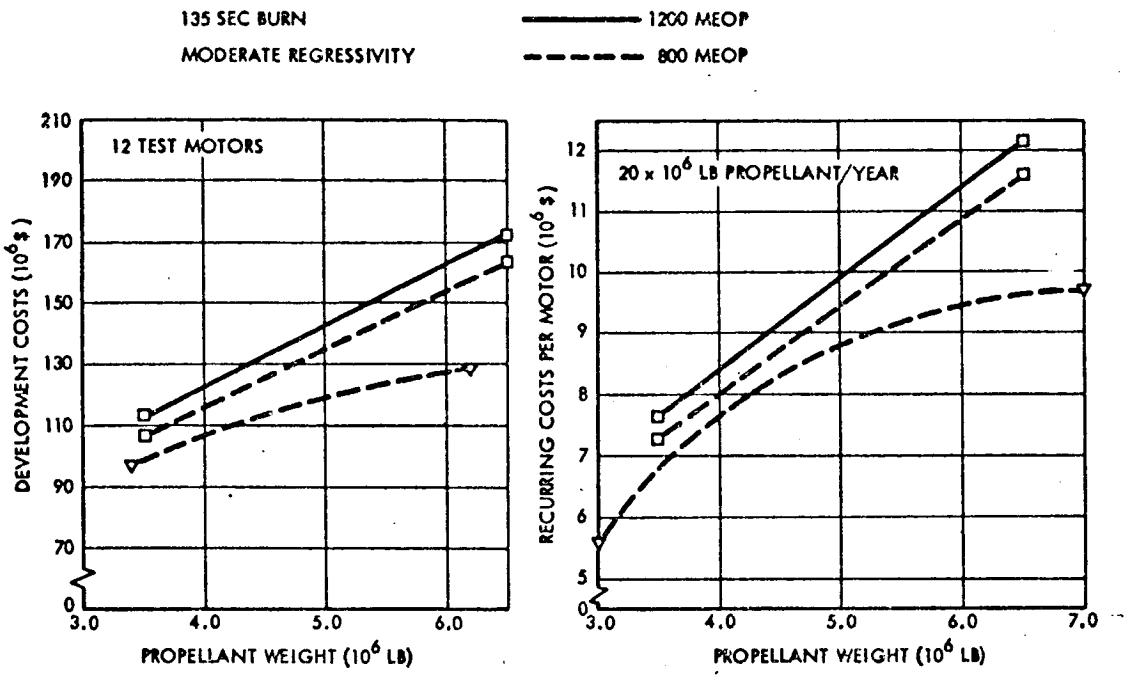


Fig. 3-5 Cost Comparisons of 260-In.-Diameter Motor for 800 and 1200 MEOP

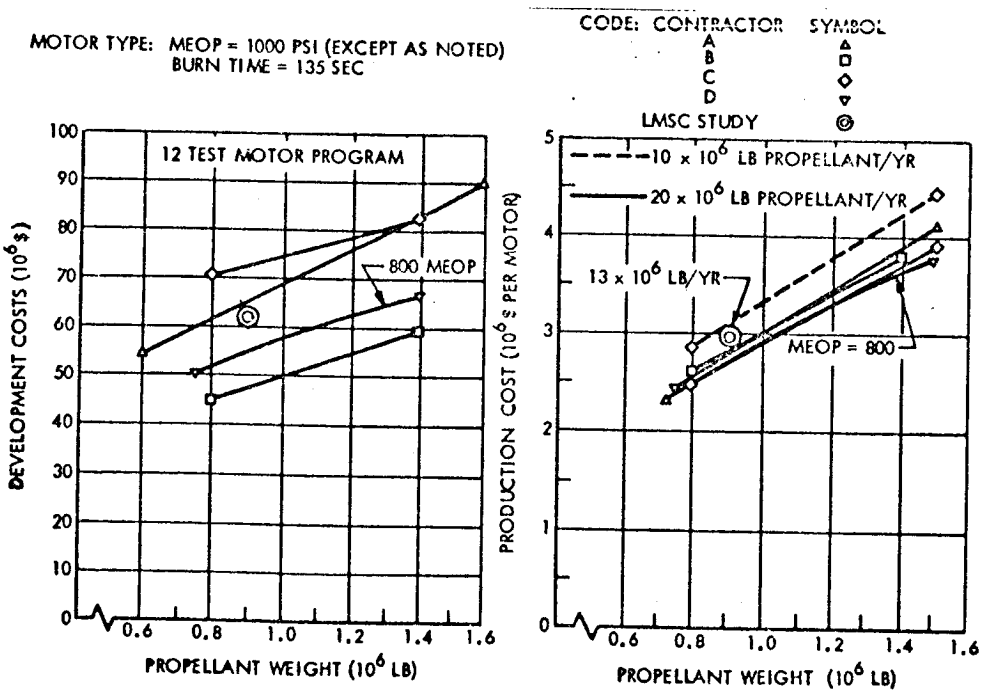
The increase in motor development costs associated with 12 motor firings instead of 8 varies from 20 to 30 percent. The percent reduction shown in recurring costs for yearly production rates equivalent to 40M lb of propellant compared to 20M lb of propellant varied widely: from 3.5 percent for contractor A, to 20 percent for contractor B. Again, these contractors were advised to reevaluate the effect of production rates for any future costing studies.

In addition to these production rates, the contractors were urged to project costs for a rate equivalent to 13M lb per year, which will provide a "learning curve" trend in the projected low launch-rate regime for the current groundrule interim booster program. In the event that larger launch rates are required, a significant further reduction of recurring costs can be expected, and a production rate point corresponding to 100 to 150M lb of propellant per year would be useful.

3.2 COST PROJECTIONS FOR INTERIM SRM BOOSTER MOTORS

Development and production costs were projected for a single-stage booster using a cluster of four 156-in.-diameter motors with a total propellant requirement of 3.6M lb or 0.9M lb per motor. Projected costs and their comparison with the contractor data are shown in Fig. 3-6. The development cost value represents a mean of the data shown by the four subcontractors and is within $10 million of the lowest and highest subcontractor values. This spread should be further reduced when the subcontractors reallocate or include costs for facilities, tooling, transportation equipment, and GSE to a common basis. Note that the costs are based upon a 12 motor development and PFRT test program which is probably more than required (ten test firings have been recommended by NASA/MSFC). Therefore, it is believed that the development cost shown for the motors is realistic or possibly conservative.

The projected production cost per motor is based upon a rate of about 14 motors per year. The value selected is based upon the average of the four contractor values, but



D06380 Fig. 3-6 Cost Comparison of 156-In. Baseline SRM

is corrected for a lower production rate (13M lb/yr) than represented in the data they submitted. Since the cost agreement among the contractors is excellent, a high level of confidence can be placed on the production cost projection. Prior to the receipt of the subcontractor data, LMSC used for the study baseline a value of $3.4 million per motor for the configuration 5B system in lieu of the presently designated value of $2.9 million.

3.3 COST DATA UNCERTAINTIES

Several factors must receive further consideration in any additional effort in determining cost-effective motor designs. One such item concerns the effect of design MEOP on motor procurement costs. The baseline motor costs used in the preparation of stage cost data for the configuration 5B were predicated on an MEOP of 1000 psi. Table 3-2 portrays the first order affects of reducing the design MEOP to 800 psi, utilizing the motor performance data from Fig. 2-10 of Section 2. The motor production cost would be reduced by about 4 percent. This is due primarily to the reduction in the motor inert

Table 3-2

SRM COST VS DESIGN TRADEOFFS

INCREASE OF MASS FRACTION

- +0.01 RESULTS IN SAVING ≈ 6 PERCENT IN STAGE COSTS (FROM STAGE WEIGHT REDUCTION)

INCREASE OF I_{SP}

- +1 SEC RESULTS IN SAVING ≈ 1 PERCENT IN STAGE COSTS (FROM STAGE WEIGHT REDUCTION)

REDUCTION OF MEOP OF 200 PSI

- INCREASES MASS FRACTION ≈ 0.01
- REDUCES I_{SP} ≈ 4 SEC
- REDUCES PER LB MOTOR AND STAGE COSTS ≈ 2 PERCENT

THEREFORE: REDUCING MEOP* FROM 1000 PSI (BASELINE) TO 800 PSI

REDUCES STAGE ONE COST = 6 PERCENT - 4 PERCENT + 2 PERCENT ≈ 3 PERCENT

*ON THE SINGLE-STAGE SRM BOOSTER

weight which has fabrication costs per pound that are an order-of-magnitude greater than the cost-per-pound of additional propellant required.

Another item that could impact the motor costs is the possible use of the HTPB propellant, currently undergoing a concerted development effort by the DoD, which should make it available on a timely basis for the Space Shuttle program. The advantage of this propellant's projected improvement in specific impulse of 1 sec and loading density of 4 percent would reflect an approximate 3.5 percent reduction in total booster stage weight. All propulsion companies agree that the cost per pound of this propellant would not be greater than the baseline PBAN propellant (and would probably be less); therefore, the cost saving on weight reduction alone would be about 3.5 percent.

A third item that might afford further cost reduction is the use of higher strength steels for the case and nozzle structure. A steel affording a 15 percent greater design tensile strength than used for the baseline motors (i.e., 230,000 psi vs 200,000 psi) would allow a mass fraction improvement of about 0.01 which would reflect a total stage weight reduction of about 5 percent.

A detailed analysis is required to determine whether the higher material and fabrication costs of the higher strength steels would partially or totally offset the cost reduction due to lower total stage weight.

Most of the cost uncertainties appear to provide more opportunity for cost reduction rather than cost increases.

3.4 CONCLUSIONS

Solid rocket motor development and production costs have been established with the assistance of the four large SRM propulsion companies. These companies were in excellent agreement on production cost data so that high confidence can be placed on the projected production costs of motors for the SRM-Booster stage. The divergences in the estimated development costs of about plus or minus $10 million from the value selected for this study reduces to a small percentage of the projected overall procurement costs.

Other cost uncertainties relating to the selection of a final design of solid rocket motors have been shown to contribute to conservatism in the projected SRM costs. Thus, further resolution of design and related cost uncertainties is expected to produce lower rather than higher dollar cost projections. Therefore, it is believed that the costs generated and projected in this study for the development and procurement of motors for an SRM-Booster stage represent a reliable estimate of what would be achieved in a real program.

Section 4

BOOSTER STAGE CONFIGURATIONS

4.1 CONFIGURATION SELECTION

The configuration selection activities on solid rocket motor (SRM) boosters were differently oriented in each of three separate phases:

- Initial Selection. This was based on the presumed requirement for an interim expendable booster, which would be replaced after twelve operational missions had been flown.
- Interim Booster Design/Analysis. The initial booster selected was studied in greater detail to achieve definition of performance and cost uncertainties associated with a typical SRM-stage so that the level-of-confidence to be expected for any SRM-booster could be established. In this phase, the booster configuration was allowed to change only in details which did not impact significantly the remainder of the system (orbiter/tank/ground facilities).
- SRM Final-Configuration Booster. With the change in groundrule to the effect that there would be no interim booster, a preliminary projection was made to conceive an SRM booster that might be cost-effective for use in the total 445-mission program.

During the course of the refinement phase, some conclusions which suggested modifications to the initial selection were evident. These were not allowed to change the refinement work underway, since the second phase objective could be accomplished using any reasonably representative expendable booster. However, the revised selection indications were impacted into the work undertaken in the last phase. Because of program redirection toward fully reusable boosters, however, the level-of-effort on SRM-boosters was severely curtailed during the last two phases. The following subsections review the configuration selection considerations in each of the three phases.

4.1.1 Initial Selection

In the initial selection activity, the SRM-booster configuration choice was influenced by each of the following factors:

(a) Interim vs Final Configuration optimization
(b) Smaller vs larger SRMs
(c) Parallel vs tandem orbiter/booster arrangement
(d) Single-stage vs two-stage SRM booster

Item (a) dictated strongly that the final flyback booster produce a delta-V (relative) that was of the order of 6,000 fps. The Item (b) tradeoff studies showed that for the interim-booster concept, the additional RDT&E costs to develop a 260-in-diameter SRM (above the costs for 120s and 156s) was a severe penalty. On the other hand, the relatively large number of 120s required to provide the desired delta-V provided a significant decrease in success probability, so that the 156-in-diameter booster appeared more desirable. Item (c) had relatively little impact on the character of the SRM booster, but vehicle control and stage separation considerations dictated in favor of the tandem arrangement. With regard to Item (d), the delta-V required was such that one SRM stage efficiently accomplishes the mission. Furthermore, two-stages increase development costs and decrease production costs, which (with the small number of missions) is not considered to be significant in the face of some reliability reduction for two stages.

The impact on success probability is shown in Fig. 4-1. For the one-stage system, the larger number of motors with the 120-in. booster (Configuration 5A) makes the failure probability three times as great as with the 156-in. booster (Configuration 5B Alternate), which was later selected as best for a single SRM stage. The implications of vastly reduced probability of failure shown in the same figure for two-stage vehicles refers to capability to abort with the second stage in the event of a malfunction in the first stage. Thus, from a mission-success standpoint, Configuration 5 has a slightly greater probability of failure than Configuration 5B Alternate (both with three SRMs) because of the greater number of stage subsystems required with the two-stage booster configuration.

SYSTEM	SOLID ROCKET MOTOR		MOTOR TYPE	NUMBER OF STAGES	PROBABILITY OF FAILURE (FAILURES PER MILLION)
	AVAILABLE PER SYSTEM	NEEDED FOR LIFTOFF			
5	3	2	156	2	40
5A	9	9	120	1	27×10^3
5B	4	4	152	1	12×10^3
5B (ALT)	3	3	156	1	9×10^3
5B (ALT) *	4	3	156	1	110
5D	7	5	120	2	93

ASSUMED PROBABILITY OF SUCCESS FOR SOLID ROCKET MOTORS IS 0.9977

* ASSUME THRUST/WEIGHT RATIO OF 1.15 WHERE SRM IS SIZED FOR ENGINE-OUT CONDITION AT LIFTOFF.

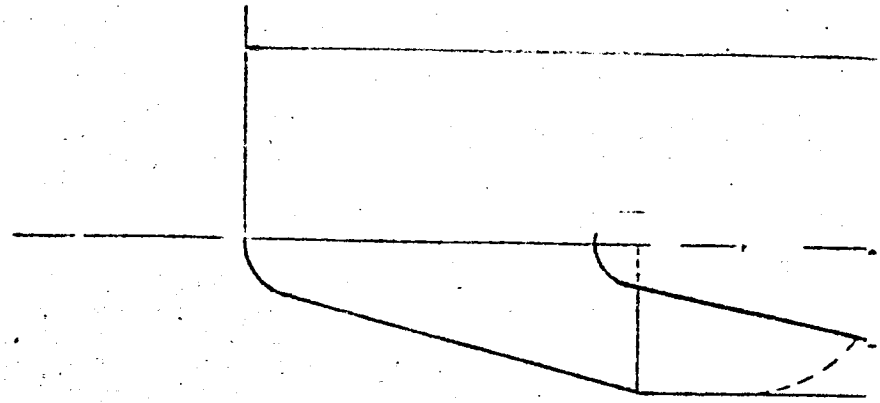
JAD-2
D04549(1)

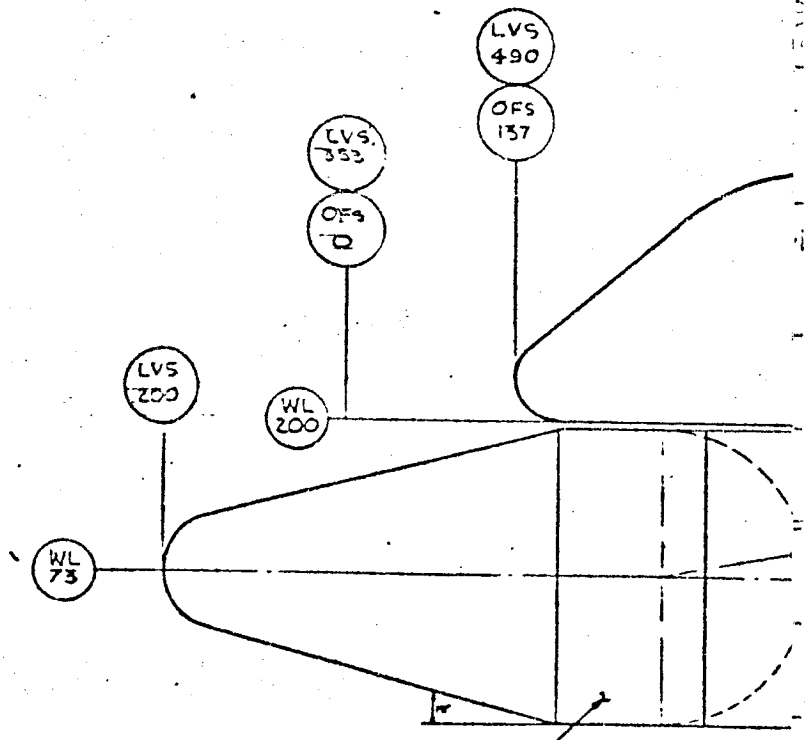
Fig 4-1 Success Probability of Solid Rocket Motor Booster System

Most of the above aspects of the interim booster configuration selection are summarized in Fig. 4-2. Details of selection activities, which involve the SRM boosters, are contained in Refs. 4-1 through 4-5. Configuration 5B was selected as the baseline interim booster for refinement study, and is illustrated in Fig. 4-3. The figure shows the four-SRM configuration on which most detail study effort was focused.

4.1.2 Interim Booster Design/Analysis

The second iteration of SRM booster design and cost data was accomplished on the booster of Configuration 5B, as described in the previous section. This work was accomplished over a seven-week period and produced the results and conclusions summarized in Sections 4.2 and 4.3. In this phase, the prime changes that proved desirable on the SRM booster are as follows:





LO_2 : 615,426 LBS
8,940 FT³

O/T WEIGHING :
84,900 LBS

- The three-SRM configuration is more attractive than the four, because with one fewer SRMs the system reliability is higher, SRM and stage costs are lower, and the stage mass fraction is better (higher).
- The TVC actuation system is more cost-effective if designed as a heavier, cold-gas blowdown system rather than as a gas-generator turbine-pump hydraulic system.
- System weight, cost, and design feasibility are improved if the SRM thrust termination systems employ two forward-head ports in lieu of the single port shown originally.

4.1.3 SRM Final Configuration Booster

With the groundrule of no interim booster, and ocean-recovery of ballistic boosters as an accepted reuse mode for the space shuttle booster, a preliminary projection was made of a cost-effective mode for SRM boosters. This projected configuration was characterized by the following:

- Use of two stages of SRMs, with first-stage consisting of two 156-in. diameter SRMs with 1.2M lb propellant load each, and the second stage consisting of one similar-sized SRM. These were the same size as the three SRM Configuration 5B (Alternate) motor grouping.
- With a delta-V (relative) of approximately 9500 fps, this allowed the orbiter to use the J-2S engines and a considerable size reduction of the LOX/hydrogen external tank.
- A recovery system concept for the SRM/stage hardware was outlined, aimed at achieving a 25 percent cost reduction in recurring operations costs for the SRM booster.

Because of limited time and budget available, little design/analysis work was done on this configuration. Parametric sizing and costing provided a basis for a rough-order-of-magnitude costing of a space shuttle program employing this system concept. These results are presented in Section 5.

4.2 STAGE DESIGN/ANALYSIS STUDIES

The SRM-stage design/analysis studies were oriented to cover three areas: (1) study special design or problem areas; (2) define preliminary subsystem design; and (3) provide a basis for preliminary estimates of stage subsystem costs. The primary activity on stage subsystem design and weight determination is summarized in a 47-page Engineering Memorandum, Ref. 4-5, and deals with the five Configuration 5 candidate booster systems. The results of this study is summarized in Section 4.3. The stage cost estimates assembled in conjunction with these design/analyses are summarized in Ref. 4-6.

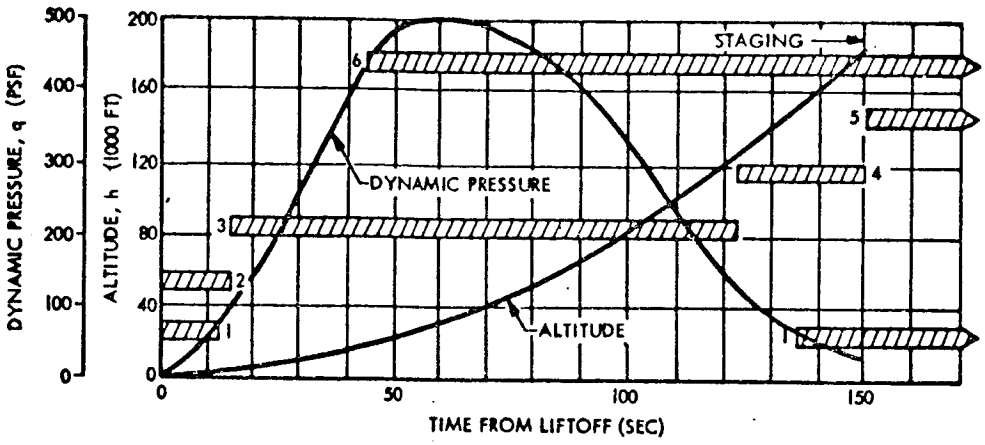
A series of special studies were undertaken to provide a quantitative basis for design selection and weight estimation. Each of these is reported in Engineering Memorandums. The specific studies, and the nature of their results and conclusions, are described below.

4.2.1 Boost Phase Abort Considerations (Ref. 4-7)

This study provides preliminary definition of potential abort situations for the Configuration 5B boost phase. Failure modes on the orbiter and the booster system were categorized into six subsets. The abort procedures during the boost-ascent phase were characterized as summarized in Fig. 4-4.

4.2.2 Nozzle Actuation System (Ref. 4-8)

Sizing three different TVC actuation systems was accomplished, with determination of weight, development costs, and recurring costs. An initial selection was made of the lightest system, which is a hydrazine-fueled turbohydraulic system. Subsequent evaluation indicated that a cold-gas blowdown system is more cost-effective. The two concepts lowest in cost are illustrated in Fig. 4-5; the weight and cost comparisons are summarized in Fig. 4-6.



ABORT ZONES (INDICATED BY CROSS-HATCHED BARS)

1. DROP EXTERNAL TANK - ROTATIONAL SEPARATION - DYN. PRESS < 60 PSF
2. ORBITER FALLBACK - CREW RAPID EGRESS (USE EJECTION SEATS) ALT. < 2000 FT
3. THRUST TERMINATE - SEPARATE - ORBITER FLYBACK TO LAUNCH SITE (BASED ON 8 MINUTES OF JET ENGINE FUEL AVAILABLE)
4. ORBITER LAND AT ALTERNATE FIELD OR DITCH IN OCEAN (FUNCTION OF LAUNCH AZIMUTH)
5. ABORT TO ORBIT OR COMPLETE MISSION
6. ORBITER MAIN ENGINE OPERATION - MINIMUM OPERATING ALTITUDE = 15,000 FT.

REF: EM NO. L4-01-03-MI-3

D04833

Fig. 4-4 Abort Modes During Interim Booster Ascent

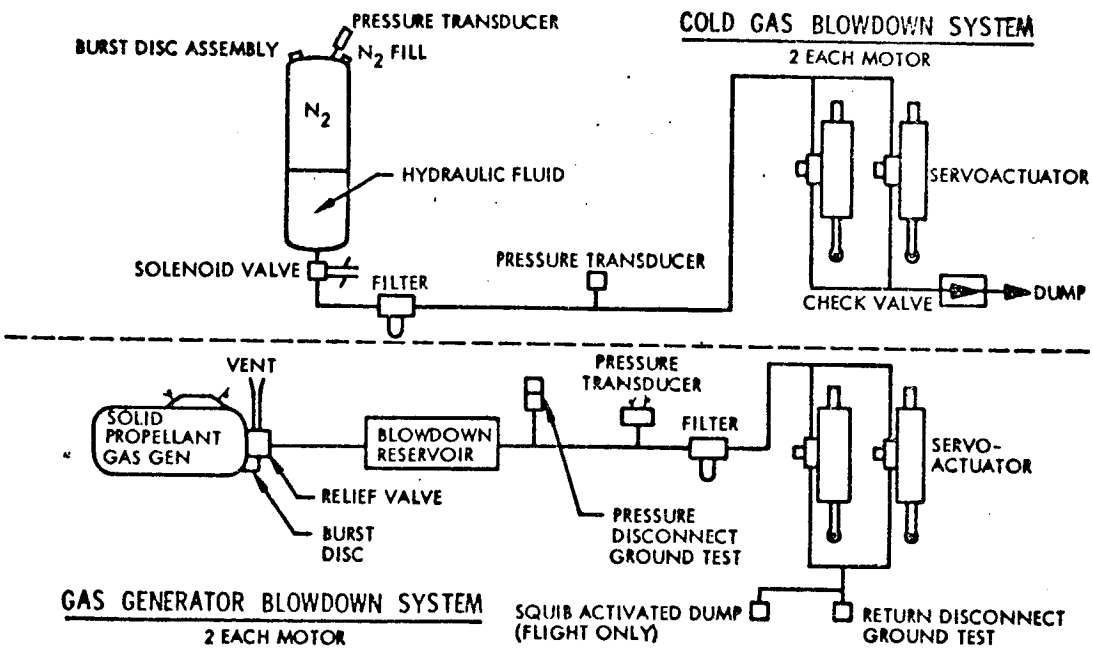


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Fig. 4-5 Blowdown System Schematics

SYSTEM	WEIGHT*	TOTAL PROGRAM COST X 10^{-6}
N_2 BLOWDOWN	2900	15
WARM GAS BLOWDOWN	1600	22
TURBOPUMP & HYDRAZINE FUEL	900	35

*20 DEG/SEC, 30 PERCENT DUTY CYCLE

D04807

Fig. 4-6 Alternate Nozzle Actuation Concepts Comparison

4.2.3 Thrust Termination Transient Loads (Ref. 4-9)

Because of the abrupt nature of load application during abort-condition thrust termination, a dynamic loads analysis was completed for interstage structural design of Configuration 5B. These were analyzed for several different structure-to-motor case stiffness ratios. Dynamic-response internal loads exceeded the maximum applied loads by no more than 10 percent in any of the typical conditions studied.

4.2.4 Thrust Termination Port Plume Characteristics (Ref. 4-10)

The thrust termination subsystem is best located on the forward domes of the SRMs, thereby resulting in proximity of the exhaust plume to LOX/hydrogen tanks and/or the orbiter. A parametric pressure/temperature/velocity profile was developed to allow determination of thrust-termination port locations which would not result in

excessive temperatures and pressures being imposed on tank or orbiter surfaces. This was used in conjunction with a three-dimensional model of booster orbiter arrangement in the configuration preliminary design phase.

4.2.5 Interstage Structural Analysis (Ref. 4-11)

A detailed structural analysis and weight calculation study was undertaken for Configuration 5B to obtain a realistic appraisal of weights and cost of interstage structures. Three major structural assemblies were analyzed for several different design conditions: the base support cluster, the forward SRM cluster, and the interstage structure. Included in the weight estimate is a +25 percent nonoptimum factor to account for design details, material gage variations, etc., which are not yet definitized.

4.2.6 SRM-Stage Manrating Requirements

The prime objective of the space shuttle program is "the development of an economical general purpose transportation system --- for transporting of crew, passengers and cargo to low earth orbit and their safe return to earth". (Quoted from Section 2.1 of the Space Shuttle Program SOW, preliminary draft dated 26 April 1971.) In view of the long period of intended routine operational use, with the likelihood of carrying a variety of passengers, such as VIP, scientific, NASA and/or military personnel, a reappraisal of manrating requirements for the booster was accomplished.

Previous efforts to establish requirements for manrating solid-propellant motors were associated with the Air Force MOL program. Manrating the SRM-stage involves essentially three aspects: (1) determination and use of adequate factors of safety for the static components of the bare motor; (2) use of redundant components for dynamic systems, such as the TVC power train, motor-igniter initiators, thrust-termination initiators and cutting ordinance, and separation ordinance and motors; and (3) an appropriate program of qualification testing of the SRMs and the assembled SRM-stage. The selected criteria are summarized in Fig. 4-7.

SPECIAL DESIGN FEATURES

- THRUST TERMINATION ON SRMs
- LAUNCH-HOLD TO T-1 SEC
- SRM PRESSURE ANOMALY SENSORS
- RIGOROUS NDT ACCEPTANCE PROCEDURES

REDUNDANT SUBSYSTEMS

- TVC ACTUATION
- SRM IGNITION INITIATORS AND COMMAND
- THRUST TERMINATION INITIATORS
- STAGING INITIATORS AND COMMAND

SRM DEVEL/QUAL TESTS

- 3 DEVEL. (FOR NEW 120 IN. SRM)
- 5 PFRT (FOR NEW 120 IN. SRM)
- 156 IN. MAY REQUIRE 9 TO 12 FIRINGS
- 260 IN. MAY REQUIRE 10 TO 15 FIRINGS

SAFETY FACTORS

- 1.15 PROOF TEST (ON MEOP)
- 1.4 ULT. (ON MEOP)
- 2.0 (ON THICKNESS) NOZZLE ABLATOR
- 2.0 (ON THICKNESS) CASE INSULATION
- 1.4 ULT. ON INTERSTAGE STRUCTURES

STAGE QUALIFICATION

- FULL-SCALE STATIC/DYNAMIC TEST
- ASSEMBLY/CHECKOUT OPS DEMONSTRATION
- ONE UNMANNED FLIGHT TEST (WITHOUT ORBITER)*
- SUBSYSTEM QUAL TESTS

*ASSUMED REQUIRED FOR ANY BOOSTER PRIOR TO FLIGHT WITH EXPENSIVE ORBITER

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Fig. 4-7 SRM-Stage Manrating Requirements

Special incorporated design features would enable rigorous NDT inspection of static components, and the sensing of impending or commencing motor failure. Thrust-termination is required to shut off SRMs in the event that abort procedure is dictated by events occurring anywhere in the Space Shuttle system. All test motors are instrumented to monitor stress and thermal history of critical components to ensure that the design margins-of-safety are being achieved. Proposed safety factors, which match currently used manrating requirements for the liquid-propellant stages, are in excess of the factors used in prior SRM applications that have already demonstrated an extremely high reliability status.

4.3 STAGE SUBSYSTEMS DESIGN

The SRM booster design/analysis activity was accomplished in three phases:

(1) A preliminary quantification of mass-fraction of the SRMs was accomplished for each configuration, and with it an allowance was estimated for stage

subsystem weight at two percent of the total SRM weight. For initial cost evaluations, a percentage of SRM costs was assumed for other stage subsystems.

(2) The second phase included a subsystem-by-subsystem preliminary conceptual design and weight estimation (Ref. 4-5) for the five Configuration 5 systems, with costs estimated for Configuration 5B by use of weight-dependent CERs for each subsystem (Ref. 4-6).

(3) In the last phase the weights and costs were reiterated for the Configuration 5B by injecting a number of corrections and improvements that came to light during the last month of the study.

For the five interim vehicle systems using tandem orbiter booster arrangements (per the second phase activity), the resulting stage inert weight breakdowns are presented in Fig. 4-8, and the contribution of the individual subsystems to the total stage inert weight is presented in Fig. 4-9. Included in these figures are the stage and composite booster propellant mass fractions (λ') in comparison with the propellant mass fractions for the SRMs proper comprising the stages. The weight of the portion of the orbiter-booster interstage remaining with the booster at separation is not included in the values listed. If this weight were included for the baseline 5B configuration, the stage $\Delta\lambda'$ in comparison with the λ' of the SRM proper would increase from 0.0201 to 0.0232.

The influence of the primary configuration design factors (number and size of SRMs, number of stages) on the reduction in $\Delta\lambda'$ between stage and SRM, based on these data, is as follows.

It appears that the stage-weight efficiency is directly reduced in proportion to the number of SRMs comprising the stage and that, consequently, use of 156-in. diameter rocket motors reduces the stage weight considerably in comparison with stages consisting of 120-in. diameter SRMs. In addition, and understandably, the single-stage configurations $\Delta\lambda'$ is also slightly decreased in comparison with two-stage configurations, viz, Configurations 5B and 5C vs 5 and Configuration 5A vs 5D. However, this decrease is not sufficient to offset the performance gain obtained by

Configuration	5		5A	5B	5D		5C
Stage	1st	2nd	1st	1st	1st	2nd	1st
SRM – Dia, in.	156		120	156	120		156
SRM – Number	2	1	9	4	5	2	3
Propellant Load, Klbm							
Stage	2,094	1,047	4,068	3,584	2,350	940	3,270
Booster	3,141		4,068	3,584	3,290		3,270
Inert Weight, Klbm							
Subsystems:							
Structures	27.40	13.70	74.41	50.00	36.18	14.47	39.28
TVC Actuation	3.45	1.72	6.12	4.08	4.80	1.92	5.07
Thermal Protection	2.80	1.60	8.73	6.65	5.96	2.97	5.20
Separation	8.05	1.98	6.75	5.80	9.30	1.60	5.00
Thrust Termination	8.55	5.94	6.57	16.00	6.31	5.06	11.02
Electric, etc.	1.20	.90	3.30	1.80	2.10	1.20	1.50
Subtotal	51.45	25.89	105.88	84.33	64.65	27.22	67.07
10% Contingency	5.15	2.58	10.59	8.43	6.47	2.72	6.71
SRM	290.60	145.30	497.70	440.80	284.50	113.80	393.00
TOTAL							
Stage	347.20	173.72	614.17	533.56	355.62	143.74	466.78
Booster	520.92		614.17	533.56	499.36		466.78
Liftoff Weight							
Stage	2,441.20	1,220.72	4,682.17	4,117.56	2,705.62	1,083.74	3,736.78
Booster	3,651.92		4,682.17	4,117.56	3,789.36		3,736.78
SRM – MEOP	1,200	1,200	852	1,000	920	920	1,000
Propellant Mass Fraction λ'							
• Stage	8,578	8,577	8,688	8,704	8,686	8,674	8,751
SRM	8,781	8,781	8,910	8,905	8,920	8,920	8,927
$-\Delta\lambda'$	.0203	.0204	.0222	.0201	.0234	.0246	.0176
• Booster	.8577		.8688	.8704*	.8682		.8751
SRM	.8781		.8910	.8905	.8920		.8927
$-\Delta\lambda'$	.0204		.0222	.0201*	.0238		.0176

*If the weight of the portion of the Booster-Orbiter Interstage separated with the Booster is included in the Booster inert weight, the Booster λ' reduces to .8673 and $\Delta\lambda'$ increases to .0232.

Fig. 4-8 Alternative SRM Booster Stage Inert Weight Breakdown

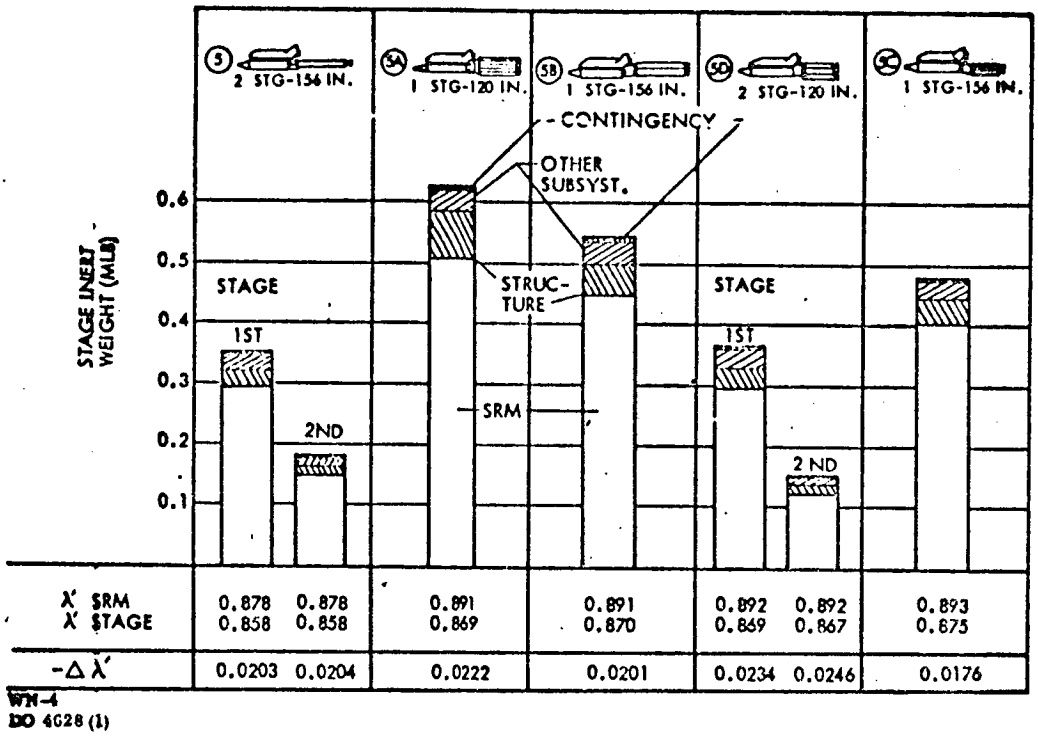

Fig. 4-9 SRM Booster Stage Inert Weight

the additional staging. In comparing the individual $\Delta\lambda'$ values obtained for the different candidate vehicle configurations with the Phase 1 assumed constant increment of 0.0200, it appears that the dispersion of -12 percent to +19 percent is significant enough to require that a relatively detailed weight analysis be included in final trade studies between competing staging and vehicle arrangement concepts.

Considering the limited depth of analysis on which these results are based, it is obvious that they should not be used without reservations in drawing final conclusions for the comparative evaluation of candidate vehicle configurations. However, as a byproduct of this study, conclusions can be drawn on the relative importance of areas where effort should be increased to provide the background for a valid, final comparative evaluation of remaining candidate interim SRM-booster configurations. These conclusions are summarized in the following subsections.

4.3.1 Stage Design and System Integration

The candidate configurations shown were the result of a preliminary design effort performed before important characteristics and interface requirements of the associated stage subsystems had been established. Consequently, many of the designs result in conditions which, to a varying degree, are unfavorable or even incompatible with the development of feasible solutions for the design and integration of pertinent subsystems. This is particularly true with regard to the thrust termination (TT) subsystem, and to some extent probably also with regard to the separation system. A second stage-design iteration would be required to eliminate obvious deficiencies.

4.3.2 Structures

As shown in Fig. 4-8, after the SRM inert weight, the structure provides by far the largest stage inert weight increment. However, only a limited, general approach was taken in estimating the discrete stage inert weights. In addition, the interface effects of many subsystems, especially for thrust termination and separation, are not taken into account by the applied methodology. The performance of a detailed, integrated design-load/stress analysis which would include interface effects of all major subsystems is a requirement for all final candidate configurations in order to permit a comparison on an equal basis of definition.

4.3.3 Thrust Vector Control Actuation Subsystem

The depth of analysis from which the weight of this subsystem was determined for the different configuration alternatives exceeds, with the exception of the SRM subsystems, the effort spent on the other subsystems. In addition, the relative contribution of this subsystem to the overall stage inert weight is relatively small and does not change significantly between different configurations. It appears that for the purpose of the evaluation of candidate configurations on the basis of performance alone, no particular additional effort would be required for this subsystem. However, with regard to other evaluation criteria, alternative TVC actuation system design concepts should be considered.

Considering development and recurring costs, use of the simple pressure blowdown system for the generation of hydraulic power should be reexamined. This concept was originally rejected in favor of the more complex APU-driven, high-pressure pump system on the basis of a weight reduction of about 50 percent. However, considering the small effect of the TVC actuation system on overall stage inert weight, use of the considerably less expensive blowdown system may prove to be the better solution. (This was done in the Phase 3 iteration.)

4.3.4 Thermal Protection Subsystem

The weight increments for this subsystem were obtained by a very preliminary and approximate scaling relationship. A more valid determination would require that the detail design characteristics of the individual stage base areas, as well as of appropriate heat shield structures, be defined under the above-discussed stage design effort.

4.3.5 Separation Subsystem

This subsystem contributes, with some vehicle alternatives, rather significantly to the total stage inert weight. The preliminary analysis used to define its weight requires considerable extension to provide adequate confidence in the results. Basic separation modes for the different vehicle alternatives must be defined, including the determination of methods for maintaining control authority during separation, determination of possible interference by impulses generated by plume impingement of the various vehicle propulsion systems involved, and consideration of both normal and malfunction operational modes. Only after the basic requirements are established by such analyses can trade studies be performed to determine whether separation rockets, as assumed in this study, should be considered for all alternatives, or whether use of mechanisms, or of a combined mechanism-rocket system would be advantageous under particular conditions. It can be expected that the individual separation subsystem weights, determined from better defined design requirements, may considerably differ from those presented at this time.

4.3.6 Thrust Termination Subsystem

As determined by this study, the contribution of this subsystem to the stage inert weight is relatively large and also highly dependent upon the particular vehicle configuration.

Among all subsystems, with exception of the SRM proper, the mutual interaction between subsystem and overall vehicle system characteristics and interface requirements is probably the most pronounced for the TT subsystems. While incorporated only for use in an emergency operation mode, accommodation of the resulting blast impingement on the orbiter and of potential shock loads on the entire vehicle structure may easily become a determining factor in the overall design of a final, fully-feasible composite vehicle configuration.

In this respect, most of the alternative vehicle configurations investigated in this study have not been sufficiently developed to ascertain functional feasibility with regard to both the subsystem and the integrated vehicle. Equally, the derived subsystem weight increments must be considered to represent at best an approximation that reflects only crudely the trends between the different types of vehicle configuration, since these configurations were not iterated to establish an equal level of feasibility for the assumed TT subsystem design.

The basic methodologies for determining both the detail design requirements for the TT subsystem, as well as the blast and load effects on the vehicle system, are available. However, it will require analytical, design, and integration efforts to arrive at a satisfactory solution. Considering the far-reaching effects on overall vehicle configuration and weight, the majority of all subsystems and technologies comprising the composite vehicle will probably be involved to some degree in such analysis, the results of which could easily become a decisive factor in the selection of the preferred configuration.

4.3.7 Electric, Electronic, and Instrumentation Subsystem

While only of minor influence on the stage inert weight, the pronounced effect of these subsystems on system reliability, safety, and cost will require a better definition of design requirements and selected design concepts for the individual configurations to contribute to the overall comparative evaluation of the vehicle alternatives.

4.3.8 SRM Subsystem

For comparative evaluation of the alternative vehicle configurations, the definition of performance, design, weight, and cost characteristics of the different SRM units used in the alternatives, obtained on the basis of contractor-furnished information, is considered adequate. Additional efforts, however, are expected to result in some increase in performance, reduction in estimated costs, and possibly operational cost savings through recovery and refurbishment of the expended motors.

4.3.9 Final Configuration Weights and Costs

In the third phase of stage design study improvements and corrections were incorporated in the weight estimates and in the cost projections for Configuration 5B. In Fig. 4-10, SRM-stage subsystem weights, development costs, and recurring costs per stage are shown, wherein the weights are more accurate than shown in Fig. 4-8. Recurring costs are based on an assumed procurement of twelve operational stages, plus one unmanned flight-test stage. The recurring cost (and dollars per pound) are average values for the total procurement. It should be remembered that first-article costs are considerably greater, and by the same token, last-article costs are significantly lower. The considerable impact of increased production (beyond the 13 sets) is quantified in a later discussion in Section 5.

The SRM data are derived, as discussed in Section 3, from four subcontract studies by SRM contractors. Values shown are conservatively high, and are attainable through fixed-price contract procurements with negligible risk of future cost growth.

The other subsystem weights and costs are based on a preliminary design of the baseline-vehicle SRM stage and prior experience in development and fabrication of similar subsystems. Considerable care was exercised in assuring that subsystem

SUBSYSTEMS		WEIGHT (KLB)	DEVEL COSTS ($M)	RECURRING COSTS PER STAGE	
				$M	DOLLARS/LB
SRMs	PROPELLANT	3,584.0			
	INERTS	440.8			
	TOTAL	4,024.8	62	11.6	2.88
STAGE STRUCTURES		61.9	6	3.7	60
TVC ACTUATION		11.0	3	0.8	75
THERMAL PROTECTION		6.6	1	1.0	150
SEPARATION		5.8	4	0.9	150
THRUST TERMINATION		3.4	2	0.3	100
ELECTRIC/ELECTRONIC		1.7	4	0.7	400
TOTAL STAGE		4,115.2	82	19.0	4.62

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Fig. 4-10 SRM Stage Weights and Costs

DATA CONSISTENCY: BETTER THAN EXPECTED

- MASS FRACTIONS – 3 VERY CLOSE, 1 AT 0.015 HIGHER
- SPECIFIC IMPULSE (VACUUM) – ALL WITHIN ±3 SEC
- DEVELOPMENT COSTS – 3 VERY CLOSE, 1 AT 30 PERCENT LOWER
- PRODUCTION COSTS – ALL WITHIN ±5 PERCENT

INTERFACE PROBLEM AREAS: REQUIRES CLARIFICATION IN SRM SPECS

- DEFINITION OF DELIVERED I_{SP}
- WEIGHT AND COST FOR THRUST TERMINATION AND TVC ACTUATION
- SAFETY FACTOR CRITERIA
- TOOLING AND FACILITY COSTS BETWEEN DEVELOPMENT AND PRODUCTION

NUMERICAL VALUE RESULTS

LMSC INITIAL VALUES			RECOMMENDED SRM VALUES
0.893	SRM MASS FRACTION	0.900	
270	$I_{SP_{VAC}}$ (SEC)	273*	
$62M	DEVELOPMENT COST	$62M	
$3.4M	PRODUCTION COST/SRM	$2.9M	

*MAY GO UP TO 277 IF HTPB IS USED

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Fig. 4-11 Recap of SRM Studies

elements on the SRMs were neither forgotten nor included twice (in the SRM, and in the other stage subsystems). In the Phase 2 study of SRM stage costs, such overlaps in fact, did occur, which accounts for the changes in weights (and costs) that are represented in Fig. 4-10, as compared with the earlier, preliminary data. Two significant changes are: (1) TVC actuation system, which now reflects a low-cost, but much heavier, cold-gas blowdown power system; and (2) thrust termination, which now removes previous overlap with weight and costs included in the SRM values. Other small reductions in weights (and costs), accrue mainly from the use of a three-motor cluster in lieu of the four-motor cluster represented in the initial baseline.

Both the weights and the costs shown in Fig. 4-10 are believed to be conservatively high. The structural weights include a nonoptimum factor of 1.25, and a contingency factor of 1.10, both of which have been proven to be realistic in other Lockheed programs. However, due to the basic simplicity of the SRM-stage system, and conservatism in design loads, the "other subsystem" weights and costs are likely to be proven in the order of 10 percent too high in more detailed design studies.

4.4 SRM STAGE DESIGN RESULTS AND CONCLUSIONS

The studies of the SRM booster potentials, although severely curtailed during the last half of the ACS program, have provided clear delineation of primary design considerations, performance and cost projections, and the extent of program risks or uncertainties. The SRM subcontractor studies have produced a remarkable consistency of both performance and cost characteristics (see Fig. 4-11) in all three sizes of large solid rockets (120-in., 156-in., and 260-in. diameters). Three of the subcontractors independently provided essentially identical characteristics, allowing for a rather narrow band of variation, with really "noise-level" differences. One of the subcontractors provided appreciably more optimistic characteristics, both in mass-fraction (performance) and in development costs. This contractor, however, has extensive experience in developing and producing a wide variety of experimental and operational solid rockets, and has a proven history of "ability to deliver," both in developmental and production programs.

Using data represented by the average of the three similar sets of characteristics (excluding the apparently optimistic set), the SRM performance characteristics used by Lockheed in the Space Shuttle System studies must be upgraded. With the data now available, the selection of a most cost-effective MEOP still remains to be optimized. When these several changes are incorporated, a saving of approximately 10 percent is expected in the SRM booster stage program costs. Also, concurrence exists among the SRM contractors that the HTPB (Hydroxyl Terminated Polybutediene) propellants are now "state-of-the-art" and planned for on-going missile programs. Because of their higher density, improved specific impulse, and small change in costs, additional reductions in SRM stage cost of approximately 5 percent can be expected. Other major cost reductions accrue with increased production rates (i.e., by increasing from 12 to 445 operational missions), and potentially by introducing ocean recovery and refurbishment of expended solid stages. The effects of these considerations is discussed in Section 5.

The study results indicate that the SRM booster technology required for the space shuttle is proven state-of-the-art, with very low risk levels in achieving both performance and cost goals. Lockheed is convinced that this technology (for the 120-in. and 156-in. motor sizes) can be realistically procured by standard competitive fixed-price contract procedures.

REFERENCES (Section 4)

4-1 EM L4-01-01-03-M1, "Initial System Safety and Abort Capability," dated 7/28/71

4-2 EM L4-01-01-08-M1, "Systems Evaluation Plans and ACS Follow-On Concepts Coarse Screening," dated 8/2/71

4-3 EM L4-01-01-M2-3, "Effect of Solid Rocket Motor Thrust Shape on Ascent Performance," dated 9/3/71

4-4 EM L4-06-M1-2, "Ground Operations Analysis of Selected ACS Space Shuttle Vehicle Configurations," dated 9/7/71

4-5 EM L4-08-01-M1, "Subsystem Design and Weight Characteristics of Interim SRM Booster Stages," dated 9/15/71

4-6 EM L4-08-01-M2, "Single Stage SRM Stage Costs, dated 9/15/71

4-7 EM L4-01-01-03-M1-3A, "Space Shuttle Boost Phase Abort Considerations – SRM Configuration 5B," dated 9/15/71

4-8 EM L4-08-04-M1, "Nozzle Actuation System for Space Shuttle 156-In. Expendable Booster," dated 9/15/71

4-9 EM L4-01-01-03-M1-4, "Thrust Termination Transient Load Study – Configuration 5B Interim," dated 9/20/71

4-10 EM L4-08-01-M3, "Thrust Termination Port Plume Characteristics," dated 9/30/71

4-11 EM L4-01-01-03-M2-5, "Configuration 5B Orbiter Droptank, Final Interstage and SRM Booster Cluster Structural Analysis," dated 9/30/71

Section 5
PROGRAM ALTERNATIVES SUMMARY

In addressing the prime objectives of the Alternate Concepts Study in relation to SRM-boosters (see Section 1.1), the following questions must be answered:

a. What is the optimal approach to flight-qualifying an SRM-booster system?

b. What is the optimum interim-expendable SRM-booster configuration, and what are the costs for that type of interim-booster program?

c. What SRM-booster concept, and associated space-shuttle total-program plan, produces lowest peak annual funding (especially in the initial years of the program) and at the same time requires a low total-program budget?

d. What are the comparative safety and mission-success probabilities, and how do technical and cost uncertainties for the SRM-boosters compare with other booster candidates?

These questions are addressed in the following sections, followed by a summary and conclusions discussion.

5.1 FLIGHT QUALIFICATION PROGRAM

The basic criteria for man-rating an SRM-booster are summarized in Section 4 of this report. Key items in these criteria are the requirements for: (1) a full-scale static/dynamic test program, (2) a complete assembly/checkout operations demonstration, and (3) one unmanned flight test (of booster without an orbiter vehicle). These major efforts have been specifically identified as being over and above the conventional PFRT requirements for the SRMs, and the typical subsystem qualification testing performed on each of the subsystems utilized on manned flight vehicles.

Prior to the unmanned-flight test, the major portions of the ground test program of a completely assembled SRM-stage, with its interstage to the tandem tank, should have been completed. This program is activated concurrently with the SRM-stage subsystem development efforts costed in Fig. 4-10. It continues to a later date so that subsystem testing, which requires use of large elements of the SRM-stage, can be accomplished after the component and breadboard tests have been essentially completed. This program will include: (1) structural static and dynamic tests, (2) vibration tests, (3) separation tests, and (4) integrated-stage subsystem tests. Costs of this program are summarized in Table 5-1.

Table 5-1
SRM-STAGE GROUND TEST PROGRAM

ITEM	COST
FOUR INERT-PROPELLANT SRMs (USING MOTOR PARTS FROM DEVELOPMENT MOTORS)	$ 6M
STAGE STRUCTURES	4M
TEST FIXTURES AND INSTRUMENTATION	3M
TEST ENGINEERING AND OPERATIONS	2M
GROUND TEST PROGRAM TOTAL	$ 15M

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A flight test program of a fully equipped, operational-configuration SRM-stage is considered advisable as part of the development/qualification of the interim-expendable booster, prior to the operational manned flights. Such a program can be approached in either of two ways:

- A launch with some useful unmanned payload, such as one of the S-IVB stages from the existing inventory with an appropriate orbital mission spacecraft
- A launch with a mass-properties simulated tank-and-orbiter combination, with at least partial simulation of upper-stage aerodynamics and separation characteristics.

The preliminary assembly of costs for such a demonstration is summarized in Table 5-2, and is based on the latter approach.

Table 5-2

SRM-STAGE FLIGHT DEMONSTRATION PROGRAM

ITEM	COST
FLIGHT-OPERATIONAL SRM STAGE	$ 19M
SIMULATED UPPER STAGES	9M
INSTRUMENTATION AND TEST ENGINEERING	3M
TEST OPERATIONS AND LAUNCH FACILITY REFURBISHMENT	2M
FLIGHT TEST PROGRAM TOTAL	$ 33M*

*DOES NOT INCLUDE A NONRECURRING LAUNCH FACILITY MODIFICATION.

The approach using the S-IVB would presumably have space-program value separate from the space-shuttle program with budgets allocated from that area. While that approach would not demonstrate all the space-shuttle trajectory events contemplated for the latter approach, it is considered to be sufficiently applicable to meet qualification-test primary objectives. Study of SRM-boosted S-IVB stages is currently being carried out by the McDonnell Douglas Co. for NASA/OART under Contract NAS 2-6446. Cooperative data exchanges are occurring between that work and this ACS program. Accordingly, LMSC requested data on mission, performance, and cost consideration for this flight-test option. The Appendix provides the data provided by McDAC, which includes many attractive mission/configuration options for the unmanned flight test required for space-shuttle booster qualification.

The unmanned flight test, with or without simulated space-shuttle upper stages, will include: (1) demonstration of SRM and stage-assembly operations; (2) preflight SRM-stage checkout; (3) launch facility preflight preparations; (4) SRM-stage launch operations; (5) ascent-trajectory maneuvering, with pitch, yaw, and roll control characteristics evaluation; (6) thrust-termination and upper-stage separation; and (7) postflight launch facility refurbishment.

5.2 INTERIM EXPENDABLE BOOSTER PROGRAM

The SRM-booster design/analysis activity discussed in Section 4 concluded that, for a space-shuttle development program which first uses an interim-expendable booster, and then transitions to a fully reusable booster, a single SRM-stage consisting of three 156-in.-diameter SRMs is the optimum expendable booster. The study results provide development program preliminary definition (including costs), and recurring costs for twelve manned-flight launches. Using the data provided in Fig. 4-10, and Tables 5-1 and 5-2, a total program cost for the recommended interim SRM-booster is summarized in Table 5-3.

Table 5-3
INTERIM BOOSTER PROGRAM COSTS

NONRECURRING ITEMS	COST ($M)
SRM DDT&E*	$ 62
STAGE SUBSYSTEMS DDT&E	20
STAGE INTEGRATION AND SYSTEM SUPPORT	13
STAGE GROUND-TEST PROGRAM*	15
STAGE FLIGHT-TEST PROGRAM	33
NONRECURRING COST TOTAL	$ 143
RECURRING ITEMS	
SRMs (11.6M x 12)	$ 139
STAGE SUBSYSTEMS (7.4M x 12)	89
RECURRING COST TOTAL	$ 228
TOTAL PROGRAM COST**	$ 371

NOTES: * INCLUDES FACILITIES, TOOLING, AND FIXTURES
** BASED ON 12 INTERIM SYSTEM LAUNCHES AND ONE UNMANNED BOOSTER FLIGHT TEST

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Groundrules utilized in assembling costs were:

- SRM contractor costs were evaluated, and a cost selected close to the upper extreme of the scatterband.
- Only 1970 dollars were used.

- Minimum technology advancement was utilized, except possibly for the movable nozzle TVC system.
- No guidance avionics was on the SRM stage; no orbiter costs; no spares for SRMs or stage structures, but spares were included in TVC and smaller subsystems.
- One major ground-test program was conducted for stage qualification.
- One unmanned-flight test of an operational SRM-stage was conducted.
- Stage Integration and System Support was estimated as 15 percent of SRM plus stage-development cost.
- Twelve SRMs were required for 156-in.-diameter motor development and PFRT.
- KSC launch-facility and assembly-modification costs were not included. (See Fig. 6 in Ref. 4-4.)
- Spent motors were reused for ground-test SRM-stage qualification-test program.

For the groundrules specified by NASA and the SRM-stage configuration and development program philosophy represented in this study, this total-program estimate is considered accurate within ±10 percent.

5.3 PROJECTED OPTIMUM SRM BOOSTER

With the revised program groundrules, which require one booster development and allow ocean-recovery of the booster, an entirely new view can be taken of SRM-booster options. The following relaxations of critical-program problem areas can be envisioned, if SRM-boosters are utilized and allowed to deliver a "most cost-effective" delta-V:

- Three 156-in.-diameter SRMs, of the same size as is used in the single-stage three-SRM interim booster, can deliver 50 percent more delta-V to the orbiter/tank upper-stage (at essentially the same cost) if used in a two-stage combination (two SRMs in the first stage, one in the second stage).

- A two-stage SRM-booster eliminates the "deadband" in abort capability of all other booster systems (in the period of 15 or 20 sec immediately after launch) by providing an escape-rocket in the second-stage SRM.
- The size of droptank required with the orbiter is drastically reduced. Also, the engine-requirement on the orbiter is satisfied with J-2S engines, and in smaller numbers than for any other booster.
- Development-program costs for the booster are low, and peak annual funding in the early years of the program are lowest of all program concepts studied.
- The usefulness of the SRM-booster in other program applications, such as with the SIVB to deliver heavy and/or large volume payloads (see Appendix A), is an attractive offshoot.
- The MOL launch facility at WTR is readily adaptable as a permanent second launch base, and if used for the initial unmanned-test flight, an early-year fiscal-budget savings of some $30 or $40 million results.
- The two-stage SRM-booster is a highly cost-effective two-stage combination, which, if used with the second stage of the Titan III concept for USAF candidate-missions "requiring" the TIIIL, would eliminate any need for Titan IIIL development.
- Development program risks for evolving the booster are believed to be more state-of-the-art (hence less risk) than for any other booster candidate. At the same time, by reducing size of propellant and thrust requirements for the orbiter, complexity and risk in the orbiter development are also reduced.
- If the orbiter payload is temporarily reduced and propulsion system installation is delayed, the orbiter development can be accelerated by launching it as a glider on top of the two-stage SRM/SIVB combination. This parallels studies performed under Contract NAS 2-6446 (see discussion in previous Section 5.1.)

The primary objection to such an SRM-booster is its presumed higher total-program cost; however, this objection has not been adequately evaluated. Evaluation is needed for: (1) the cost reductions that result from use over the entire 445 mission model, and (2) the possibility of recovering and refurbishing major portions of the SRM-stages.

The first of these factors was preliminarily analyzed, to determine a rough-order-of-magnitude of booster costs if used over the entire mission model. In the analysis, the SRM-booster was presumed to be configured as a two-stage system, using two 156-in.-diameter SRMs as first-stage (total w_p = 2.4 M lb) and one identically sized SRM as second-stage (total w_p = 1.2 M lb). As far as SRM-booster costs are concerned, this two-stage system development and production program is considered to be of essentially the same cost as the three- or four-motor single-stage booster discussed for the interim booster system. However, it provides the added advantages shown in Table 5-4, with the full-time abort capability (by thrust termination of the first stage and lightoff of the second stage) as a major capability not provided by most, if not all, other booster concepts.

The recurring launch costs shown previously were predicated on only 12 operational missions. As shown in Table 5-4, for the full mission model of 445 launches, the SRM-booster costs reduce from $19M to $12.5M average cost. The learning-curve cost reductions represented are considered to be conservative, based on Titan IIID SRM-stage cost-improvement experience. The potential additional saving by recovery and refurbishment is estimated to net between 20 and 40 percent less than the $12.5 million.

This second factor, namely the potential savings by recovery and refurbishment, was also preliminarily studied. Refurbishment and multiple reuse of rocket motor components have been standard practice in SRM test programs and have been successfully accomplished in the 120-, 156-, and 260-in.-diameter motor development programs. For the shuttle program, the potentials are summarized in Table 5-5. All structural components normally are thermally protected so that no degradation occurs during the motor operation and, therefore, they are "good-as-new" for reuse. The effect of short-time immersion in seawater should be comparable to the long-term saltwater environment to which Fleet Ballistic Missile motors and high-strength steel hulls of deep submergence vehicles have been exposed without deleterious effects. The remaining question to refurbishing and reuse is the possibility of structural damage upon impact with the water.

Table 5-4

SRM-BOOSTER COST-EFFECTIVENESS POTENTIALS

PRIME FACTORS

DEVELOPMENT COSTS LOW:	• $373M (± $25M)
THREE 156-IN. SRMs IN TWO STAGES:	• REDUCES ΔV REQUIRED FROM ORBITER • ABORT CAPABILITY FROM LIFTOFF • MOL FACILITY AT WTR USABLE AS SECOND LAUNCH BASE
INCREASE IN PRODUCTION RATE:	• REDUCES BOOSTER RECURRING COSTS FROM $19M TO ≈ $12M
POTENTIAL RECOVERY/ REFURBISHMENT SAVINGS:	• 20 PERCENT TO 40 PERCENT REDUCED RECURRING COST

PRODUCTION RATE COST IMPACT

				REMARKS
NO. OF LAUNCHES	12	200	445	
LAUNCH RATE (ANNUAL)	4	20	45	
SRM COST PER BOOSTER ($M)	11.6	9.0	8.3	93.5 PERCENT LEARNING
OTHER BOOSTER COSTS ($M)	7.4	4.8	4.2	89 PERCENT LEARNING
AVERAGE TOTAL BOOSTER COST ($M)	19.0	13.8	12.5	

Table 5-5

RECOVERY AND REFURBISHMENT POTENTIALS

REUSABLE COMPONENTS

SRM
- MOTOR CASE
- NOZZLE FLEX JOINT
- THRUST TERMINATION SYSTEM
- NOZZLE STEEL SHELL

STAGE
- 50 PERCENT (APPROXIMATELY) OF STRUCTURE
- MOST OF ELECTRICAL SYSTEM
- TVC ACTUATION SYSTEM
- 50 PERCENT (APPROXIMATELY) OF RECOVERY SYSTEM

REPLACED/REFURBISHED ELEMENTS

SRM
- PROPELLANT AND IGNITER
- CASE INSULATOR
- NOZZLE ABLATOR

STAGE
- INTERSTAGE (TO BOOSTER)
- STAGING JOINTS AND ROCKETS
- PART OF RECOVERY SYSTEM

ESTIMATED COST SAVINGS — FROM 20 TO 40 PERCENT REDUCTION OF RECURRING SRM BOOSTER COSTS

A study by the National Engineering Science Corporation on booster recovery at sea* concludes that recovery can be successfully accomplished without structural damage if minor design requirements are incorporated in the motor, and the motor entry into the water is maintained near vertical by the descent recovery equipment. Additional study is required to evaluate the cost of descent recovery systems, retrieval and transport, and refurbishing. One concept for recovery of major segments of an SRM-booster is illustrated in Fig. 5-1. In the case of the configuration 5B SRM-stage consisting of three SRMs, the illustration represents the sequence for each of the three primary motors, to which are attached reusable elements of interstage structure and subsystems. Such a recoverable stage would incorporate separation joint designs that would leave most of the interstage elements attached to the SRMs, with a minimum of quick-replacement components at the separation joints.

Electrical wiring and control system components would be potted to assure survival in the recovery-phase environment. The cold-gas storage TVC actuation system is of a basically rugged design and can readily be configured to require simply a gas recharge as part of the next preflight assembly and checkout operation. SRM-case insulation and nozzle ablator will require refurbishment, prior to reloading the motor with the solid propellant grain. The flex-joint assembly, between the nozzle and the motor case, is expected only to require a replacement of the joint thermal-protection membrane. Recovery system components will require some refurbishment, some replacement, and repackaging. The one component that requires peculiarly tailored design and development testing is the impact bag and its pressure-relief valving.

Using the projected two-stage SRM-booster "optimum" configuration outlined above as a basis, a space shuttle total-program cost is presented in Table 5-6. This table shows the estimated costs for the full 445-mission operational model, with the presumption that a 25 percent reduction of SRM-booster recurring costs can be achieved by use of a recovery system and refurbishment. The resulting space shuttle total-program cost is $8.8 billion.

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* "Recovery of Boosters at Sea," National Engineering Science Corporation, Pasadena, California, Report No. S-260, April 1967, under Contract NAS 7-394.

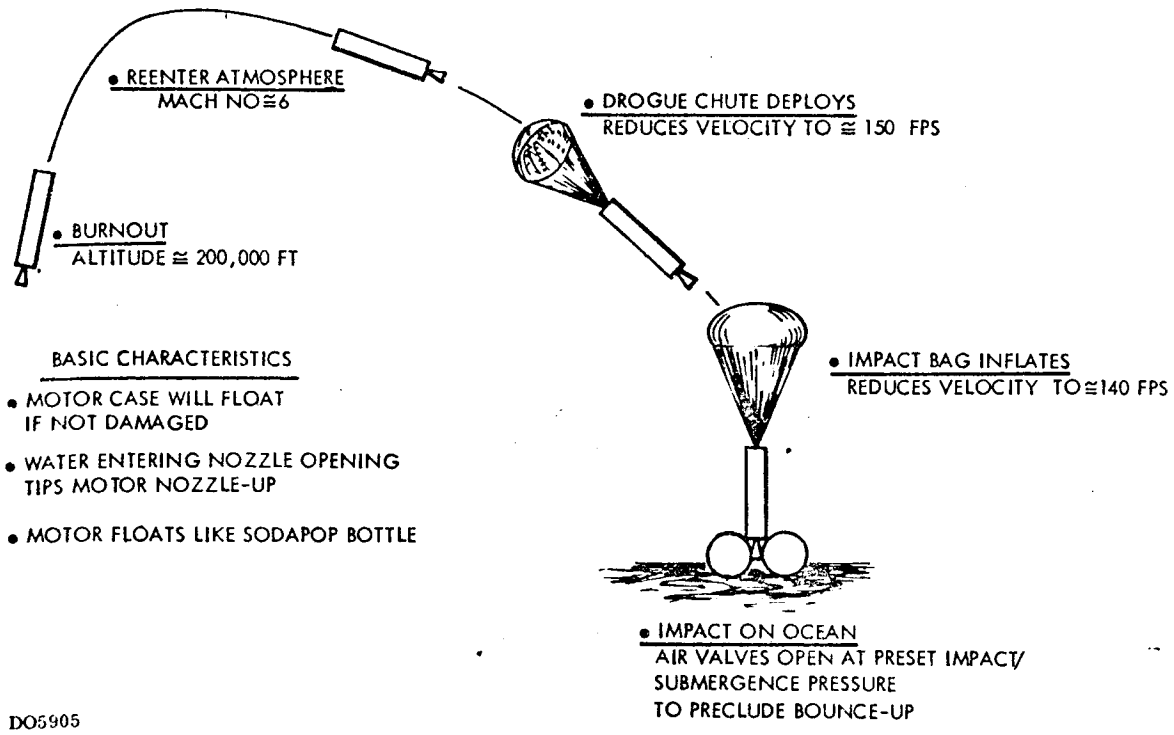

Fig. 5-1 SRM Booster Recovery Concept

Table 5-6

POTENTIAL RECOVERABLE SRM-BOOSTER PROGRAM COSTS

CONFIGURATION AND PROGRAM ASSUMPTIONS:

- ORBITER WITH
  - TWO 18K LB (DRY) EXTERIOR TANKS ($W_{P_{tot}}$ = 565K LB)
  - FOUR J-2 ENGINES
- TWO-STAGE SRM-BOOSTER (2+1 156 IN. SRMs AT 1.2M LB W_p)
- SRM-BOOSTERS USED FOR ENTIRE 445 LAUNCHES
- RECOVERY/REFURBISHMENT REDUCES BOOSTER RECURRING COSTS 25 PERCENT

DDT&E COSTS	
ORBITER	1770
EXTERIOR TANKS	165
MAIN PROPULSION	20
SRM BOOSTER	145
OPERATIONS	127
FLIGHT TEST	98
SUBTOTAL	$ 2325 M
MANAGEMENT AND INTEGRATION (AT 10%)	233
TOTAL (DDT&E)	$ 2558 M

PEAKS: $ 585M/1976
$ 803M/1987

RECURRING COSTS		
RECURRING PRODUCTION		326
RECURRING OPERATIONS		5385
EXTERIOR TANKS	469	
SRM BOOSTER	3900	
OPERATIONS	1016	
SUBTOTAL		$ 5711 M
MANAGEMENT AND INTEGRATION (AT 10%)		571
TOTAL (RECURRING)		$ 6282 M

TOTAL PROGRAM = $ 8840M

DO5941(2)

Because of the low development cost for this booster, peak annual funding requirement is very low ($0.59B) in the early years of the program. After this low early-peak funding, there is a reduction of annual funding required for several years, until full operational deployment occurs. The requirements for subsequent years then buildup as the annual launch rates increase, so that, in the last years of operational utilization, the program peak annual funding attains the peak level of $0.80B. This is a favorable funding characteristic, since technology advancements by that time may likely warrant major modifications and/or improvements to the system of a presently unpredictable nature. Also, when investment-cost discounting economics are introduced, this low initial-cost characteristic becomes a predominating factor in favor of a system such as this projected SRM-booster.

5.4 PROGRAM RISK EVALUATION

The studies of SRM-booster design, cost, and development program plans indicate that all aspects of such a booster program are essentially "proven state-of-the-art." This is reflected by the willingness of all key contractors to work on such a project on a "fixed-price" contract basis. It is also reflected in very narrow scatterbands on performance characteristics and program costs. Accordingly, the development program technical and cost uncertainties/risks appear to be clearly lower than on any other booster concept considered to date. Among the SRM-booster candidates, obviously the RATO concept, which does not even require thrust vector control, provides the simplest, lowest-risk program.

With regard to mission success and safety, again the SRM-boosters appear quite attractive. By virtue of an excellent reliability record and very simple, low piece-part-count subsystems, the SRM-boosters can make claim to providing greatest assurance of mission success and safety.

Over and above these considerations, however, is the fact that with the two-stage SRM-boosters, an abort-escape rocket is provided (in the second-stage SRM) at no cost to the system. This feature obviates the need to consider crew-jettison-seat

systems (with no provision made for passengers), and eliminates concerns about the no-abort capability "deadband" of 15 or 20 sec immediately after launch, which plagues practically every other booster system concept.

5.5 CONCLUSIONS AND RECOMMENDATIONS

Primary conclusions of the SRM-booster study are shown in Table 5-7. One implicit conclusion, not specifically shown and which no longer is considered by most authorities as suspect, is the certainty of performance and costs indicated in the study. These values go with a projected reliability of 0.997 for each solid rocket motor. Contractor willingness to accept fixed-price contract commitments on the SRM (in 120-in. and 156-in. diameters) development and production attest to these conclusions.

Because of limited budget allocation to the study of SRM-boosters, extensive-iteration activities could not be undertaken. Accordingly, the final projection of SRM-booster potentials were accomplished without the benefit of detailed design-optimization effort.

Table 5-7

SRM BOOSTER STUDY CONCLUSIONS

- SINGLE-STAGE SRM-BOOSTER, WITH THREE 156-IN.-DIAMETER SRMs PROVIDES BEST "INTERIM" BOOSTER
- TWO-STAGE SRM-BOOSTER (2 + 1 156-IN.-DIAMETER SRMs) APPEARS TO BE MOST COST-EFFECTIVE CANDIDATE, WITH EXTERNAL TANK ORBITER AND HIGHER STAGING VELOCITY
- TWO-STAGE SRM-BOOSTER PROBABLY HAS BEST BOOST-ABORT CAPABILITY OF ANY SPACE SHUTTLE BOOSTER CANDIDATES
- WITH RECOVERY AND REFURBISHMENT, SRM-BOOSTERS MAY BE MORE COST-EFFECTIVE THAN RS-1C AND PRESSURE-FED LIQUID
- SRM-BOOSTERS PROVIDE LOWEST-RISK BOOSTER CONCEPTS, WITH LOWEST EARLY-YEAR PEAK-FUNDING REQUIREMENT

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Cost-effectiveness improvement trends, however, point to the fact that, if a SRM-booster is required, a two-stage SRM-booster is more cost-effective. This two-stage system, however, presumes that the delta-V provided by the booster is allowed to increase (over that provided by a single stage) and orbiter engine-thrust and external tank capacity are reduced accordingly. The mission-abort capability of such a two-stage booster is uniquely attractive.

If the SRM-booster is permitted to employ recovery and refurbishment, two conclusions are evident: (1) the SRM-booster is likely to be competitive with flyback LOX/RP and ballistic recoverable pressure-fed boosters; and (2) the development and cost uncertainties of the solid booster appear to be clearly the least among the three systems. If recovery of the pressure-fed and solid boosters is rejected as being economically impractical, preliminary analysis indicates the likelihood that the solid boosters are more cost-effective.

In view of the limited depth of SRM-booster design optimization activity in the present study, a listing is provided of activities that should be undertaken if a more extensive follow-on effort is authorized. Heretofore, the SRM-boosters have not been considered in the light of utilization over the entire space shuttle mission model, nor has there been any in-depth evaluation of recovery/refurbishment potentials. Preliminary analysis of these aspects indicates that one (or certainly both) of these would place the SRM-booster in a cost regime competitive with the other booster candidates. Projected primary tasks are shown in Table 5-8.

Table 5-8

SRM-BOOSTER CANDIDATE OPTIMIZATION TASKS

- DETERMINE SRM DESIGN DETAILS FOR RECOVERABLE VS EXPENDABLE SRM-BOOSTERS; DEFINE ASSOCIATED STAGE SUBSYSTEM CHARACTERISTICS
- DEFINE SRM-STAGE RECOVERY SYSTEM AND REFURBISHMENT PROGRAM
- PRELIMINARY DESIGN OF TWO-STAGE SRM, ORBITER WITH SMALL EXTERIOR TANKS AND DEFINE COST-EFFECTIVE SRM DESIGN DETAILS
- COMPARE COST-EFFECTIVENESS ASPECTS OF SRM RATO, SINGLE-STAGE AND TWO-STAGE BOOSTERS
- DEFINE PLANS AND COSTS FOR FULL-PROGRAM (445 LAUNCHES) PRODUCTION OF RECOVERABLE VS EXPENDABLE SRMs
- DETERMINE COMPARATIVE FULL-PROGRAM (445 LAUNCHES) PLANS AND COSTS FOR RECOVERABLE VS EXPENDABLE SRM STAGES

D05681(1)

APPENDIX A

MCDONNELL DOUGLAS

SRM BOOSTER DEMONSTRATION FLIGHT UTILIZING THE S-IVB STAGE

NOVEMBER 1971

MDC G2627

PREPARED BY:

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PREFACE

This report was prepared in response to Lockheed Missiles and Space Company letter LMSC-A993396, dated 1 October 1971, which requested certain configuration and cost data on utilizing an S-IVB stage in a flight demonstration program of an interim expendable Space Shuttle system booster comprised of 156-inch solid rocket motors.

The time period for this response was very short; hence the data contained herein must be considered preliminary, and the cost data ROM at best.

Requests for further information should be directed to the following:

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Section 1
INTRODUCTION AND SUMMARY

Current Space Shuttle System studies are addressing not only the fully reusable Booster-Orbiter system, but options based on an interim, expendable booster used with a reusable orbiter. One of the candidate interim boosters being proposed is comprised of 156-inch solid rocket motors (SRM) clustered to form a stage.

A flight demonstration program is considered advisable as part of the manrating development of the large SRM booster. Such a flight demonstration program could be accomplished with simulated upper stages, or with the use of a reliable upper stage such as the Saturn S-IVB along with some useful payload. The purpose of this report is to provide data pertinent to the integration of an S-IVB stage with a flight demonstration SRM booster.

These data were requested by the Lockheed Missile and Space Company, who are addressing the definition of the SRM interim expendable booster. Basic data requested, and included herein, cover the subjects of S-IVB configuration options, performance, weight and trajectory parameters, and various program/cost data.

The results of these analyses indicate a wide range of payload capability to low earth orbit. Payloads of 20-30,000 pounds are possible with three parallel-firing SRM boosters flying with shuttle booster burnout constraints. By staging the SRMs, and permitting relaxed booster end conditions (in terms of higher altitude and staging velocity), payloads in the neighborhood of 100,000 pounds can be achieved. It is estimated that an existing Saturn IB S-IVB and IU (GFE) can be prepared for and flown on such a test for approximately $11.8 million, whereas production and launch of a single new stage could run as much as $42.3 million. Acquisition of a stage from a new buy of twelve low cost S-IVB stages would involve a cost of approximately $22.5 million for the flight test.

Section 2
GROUND RULES AND ASSUMPTIONS

2.1 ASCENT CONSTRAINTS

A. Circular orbit, 100-nmi, due East launch from ETR.

B. Constrained performance based on first stage cutoff velocity of 6000 fps, and maximum acceleration of 3 g's (8000 and 9000 fps for particular cases).

C. Unconstrained performance based on maximum payload capability.

D. Staging altitude for single-stage (SRM) application approximately 180,000 ft; for two-stage application (SRM first stage) approximately 100,000 ft.

2.2 CONFIGURATION

A. New interstage, payload shroud and outer SRM nose shrouds required.

B. Saturn 1B/S-IVB preferred configuration; new hardware would be low-cost S-IVB including astrionics function.

C. New Skylab-type shroud would be lightweight configuration.

D. New interstage would be part of first stage.

E. S-IVB mods assumed to be primarily electrical.

2.3 COSTING

A. New stage buy (low cost S-IVB) based on quantity of twelve.

B. 1971 dollars, no escalation, no fee.

C. Integration and assembly costs (VAB) to be determined. Booster costs not included.

D. Launch operations and data reduction costs for the composite vehicle test operation to be determined (no payload integration and/or operations costs to be included).

Section 3
CONFIGURATION DESCRIPTION AND MISSION APPLICATIONS

3.1 VEHICLE CONFIGURATION

The candidate vehicle configuration is shown on Figure 3-1. Basically, it is comprised of a payload and shroud, Instrument Unit and/or S-IVB stage, and an SRM first stage.

The first stage consists of three 156-inch diameter solid rocket motors mounted in line, necessary clustering and staging hardware, SRM forward and aft skirts, nose fairings for the outer SRMs, and an interstage to mate the center SRM to the S-IVB stage. Each segmented SRM consists of a forward and an aft closure, and four or five equal center segments. Motor case material is steel, and the propellant is a PBAN formulation. A forward skirt provides the attach structure for either a nose fairing or an interstage; the aft skirts, in addition to providing protection for the nozzle and aft mounted equipment, provide the support for the entire stage/vehicle. The nozzle is slightly submerged in the aft closure and employs a Lockseal gimbal capable of gimbal angles of up to 10 degrees. Typically, the gimbal actuators would be similar to those of the Saturn S-IC stage (F-1 engines) and might be powered by modified Nike/Zeus hydraulic power units. These power units contain a small solid-propellant gas generator, an axial flow turbine, a hydraulic pump, accumulator and reservoir. The interstage would be a truncated conical section of either conventional skin-stringer-ring frame construction, or if quantities warrant, an integrally machined construction.

The S-IVB stage would be either a standard Saturn 1B configuration, or a low cost version. Both are single-burn stages powered by a single J-2 engine, developing 205,000 pounds of thrust (vacuum) at a nominal oxidizer/fuel (LOX/LH_2) mixture ratio of 5:1. The stage is 260-inches in diameter, approximately 58.5 ft. in length, and contains a nominal propellant load of 230,000

BASELINE VEHICLE CONFIGURATION

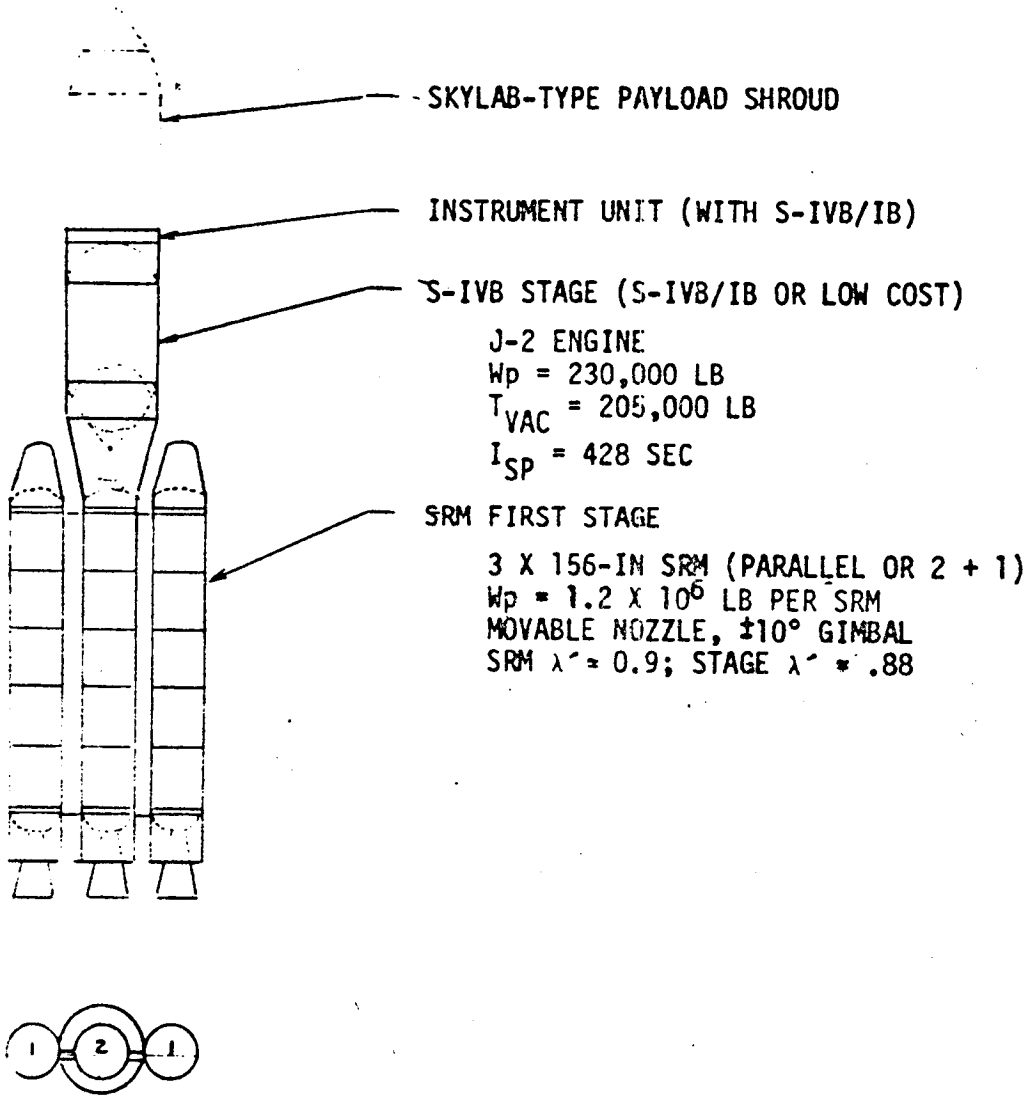

Figure 3-1. BASELINE VEHICLE CONFIGURATION

pounds. Unnecessary subsystems are removed from the stage to affect a low cost version. If a standard Saturn 1B/S-IVB were employed, a Saturn IU would also be included to provide necessary electrical and guidance and navigation functions. In the low cost version, these functions are incorporated into the S-IVB stage forward skirt.

A payload and payload fairing completes the vehicle stack. A typical fairing might be a lightweight version of the jettisonable Skylab shroud, which is 260-inches in diameter and has a double angle nose-cone.

3.2 AVAILABLE SATURN HARDWARE

A total of 27 S-IVB stages and Instrument Units have been built for Saturn-Apollo flights, fifteen of these in Saturn V flight configuration (500 series) and twelve in Saturn 1B flight configuration (200 series). Of these, all but stages 514, 515, 209, 210, 211 and 212 are definitely committed to either the Apollo or Skylab programs. However, stage 515 is designated as Skylab backup hardware, as is stage 209. Thus, stages 514, 210, 211 and 212 may be considered excess hardware. Further, some long lead items have been ordered for S-IVB stages 213 and 214, but would only comprise approximately 4 percent (by cost) of a new stage. No such buy has taken place for IU's 213 and 214.

In the event of a need for ten or twelve flight stages for an SRM-booster vehicle program, production of low-cost configuration S-IVB stages would be proposed, which would incorporate the astrionics-guidance and navigation function (no IU's would be built).

3.3 ALTERNATE MISSION APPLICATIONS

Since the proposed interim Shuttle booster (3 - 156 in. SRM) is well suited for use with the S-IVB upper stage, consideration has been given to carrying an S-IVB with a useful payload while flight qualifying the Space Shuttle SRM booster. Such a mission should:

A. Demonstrate SRM stage integrity, controlability and performance

B. Be unmanned for maximum personnel safety

C. Use a relatively low cost payload

D. Support manned flight program objectives.

Requirements A and B are mandatory while C and D are desirable. Two candidate missions which fulfill these requirements to varying degrees are discussed below:

3.3.1 Scale Model Orbiter

In this mission, suggested as a possible candidate by LMSC, a scaled down, unmanned model of the Space Shuttle Orbiter would be delivered to earth orbit. The scale model would demonstrate orbiter flight characteristics throughout boost, orbital coast, reentry and landing. Thermal protection, attitude control and landing system concepts could be verified prior to commitment on the full sized orbiter. If SRM/S-IVB capability is insufficient for orbit achievement with the model, a suborbital mission would still provide data for much of the orbiter flight regime. If separation and control problems should be identified, or should flight loads analysis determine that loads caused by the model exceed S-IVB capability, the model could be boosted through the atmosphere within a 260 in. diameter shroud of the Skylab configuration type. This would restrict the model to approximately 1/4 scale. Flight testing of this nature was performed on the Asset Program during the early 1960's when the Thor booster was used with lifting body models. At that time the models were not shrouded.

3.3.2 Smithsonian Earth Physics Satellite (Cannonball)

One of the principal recommendations of the Williamstown Study on Solid-Earth and Ocean Physics (1969) was that NASA develop techniques for ranging satellites to an accuracy of $\pm$2cm. Range measurements of this accuracy will be necessary to accomplish many objectives related to the Earth Physics Program including the determination of plate-tectonic motions (continental drift), earth rotation variations, wobble of the earth, and earth tidal motion. To make accurate measurements relating to the above objectives, a global system must be established in such a way that each position on the globe can be related to all others, and that complete sets of measurements can be obtained in less than a day. Also,

it is required that a continuity of such measurements be maintained over long periods of time.

The Smithsonian Astrophysical Observatory has proposed the use of a high density, spherical, passive laser-retro reflector satellite placed in a 3720 KM (2008.6 n.mi.) circular orbit at a 55-degree inclination. Such a satellite can provide the means for achieving all the required objectives with accuracies better than 10 cm. This satellite would provide the high mass/area ratio for precise, stable, attitude independent measurements. Because it would be visible in all parts of the world and will have a long life in orbit, the satellite would serve as a fundamental global standard and would constitute an important first step in the NASA Earth-Physics Program.

MDAC has recently completed a study for NASA on the feasibility of using an S-IVB to place such a payload in the prescribed orbit. Modifications to existing S-IVB stages required for the mission (principally for restart) cost between $.6M and $1.2M. The payload itself is basically a passive 8000 pound sphere of depleted uranium, approximately 30 inches in diameter with retro reflectors imbedded in its surface. Compared to typical satellites, it is of low cost. While this payload does not support manned flight program objectives directly, it is a mission of value to the scientific community and exhibits a relatively low investment risk suitable for the developmental nature of the primary mission.

3.3.3 CSM Earth Orbit Mission

The SRM/S-IVB launch vehicle might be beneficial in terms of a CSM sortie mission for earth resources experiments. During the final days of the Skylab mission, while astronauts were still aboard, an unmanned CSM could be placed in the vicinity of the Skylab. The astronauts could, using their own CSM, rendezvous with and transfer to the new CSM. Since the SRM/S-IVB has a payload capability in excess of 100K lbs (compared to the 37K lbs of the Saturn 1B), the unmanned CSM could be delivered fully fueled. Further, the excess payload capability could be used for increased experiment hardware. Thus both the earth coverage and number of experiments performable with the CSM would be significantly increased. The acceptability of mission risk using a first flight booster would have to be assessed. However, the risk to personnel would be avoided by keeping the flight unmanned (utilizing astronauts already in orbit).

Section 4
VEHICLE PERFORMANCE

4.1 SRM CHARACTERISTICS

The SRMs proposed for this application are the 156-inch diameter, segmented motors. A typical motor might consist of two end closures and five equal center segments, all of steel motor case material, and contain a PBAN formulation solid propellant. Typically, submerged, movable nozzles with a maximum TVC deflection of 10 degrees would be employed on these motors. Performance characteristics of the candidate SRMs are presented in the following paragraphs.

4.1.1 Single Stage Application

The characteristics of the SRM used for a single stage application are shown on Figure 4-1. In this configuration, the three motors are arranged in a row, and are all ignited simultaneously. Each motor contains 1.2 million pounds of propellant and burns for approximately 135 seconds. Maximum uniform thrust is maintained until 50 percent burn time, and then reduces linearly to 50 percent T_{max} at end of burn. Separation weight of the SRMs is 490,900 lbs. Other parameters are shown on the figure.

4.1.2 Two Stage Application

The characteristics of the SRMs used for a two-stage application are shown on Figure 4-2. In this application, the three motors are arranged in a row, but the center motor (CSRM) does not fire until the two outer motors (OSRM) have fired and are staged. Motor burn time is 100 seconds. Maximum uniform thrust is maintained until 70 percent burn time and then reduces linearly to 75 percent T_{max} at end of burn. Separation weight of the OSRM's is 327,300 lbs; CSRM separation weight is 163,600 lbs. As is noted, the thrust-time history for the outer and center motors are slightly different.

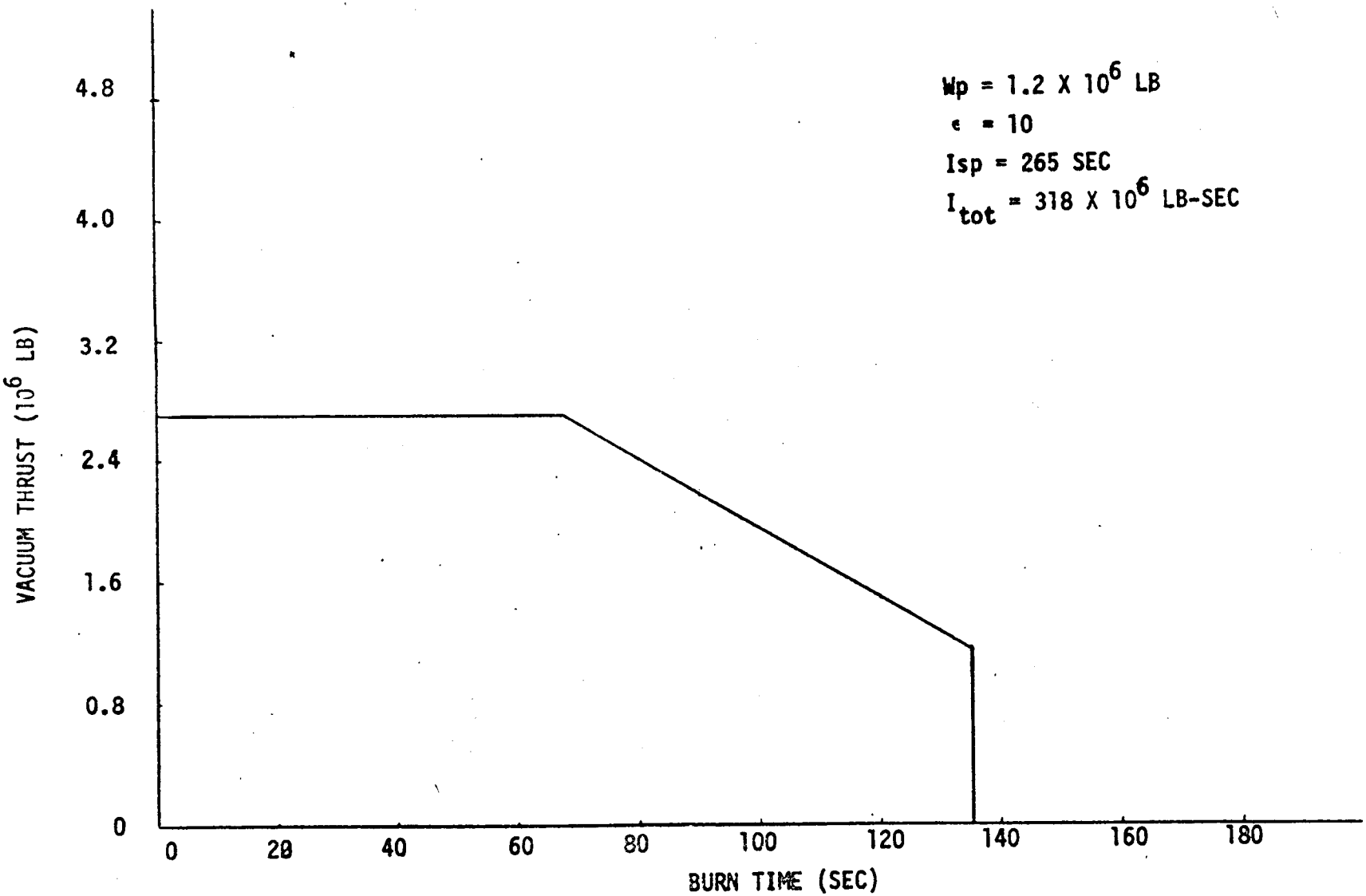

Figure 4-1 SRM THRUST HISTORY - SINGLE STAGE APPLICATION

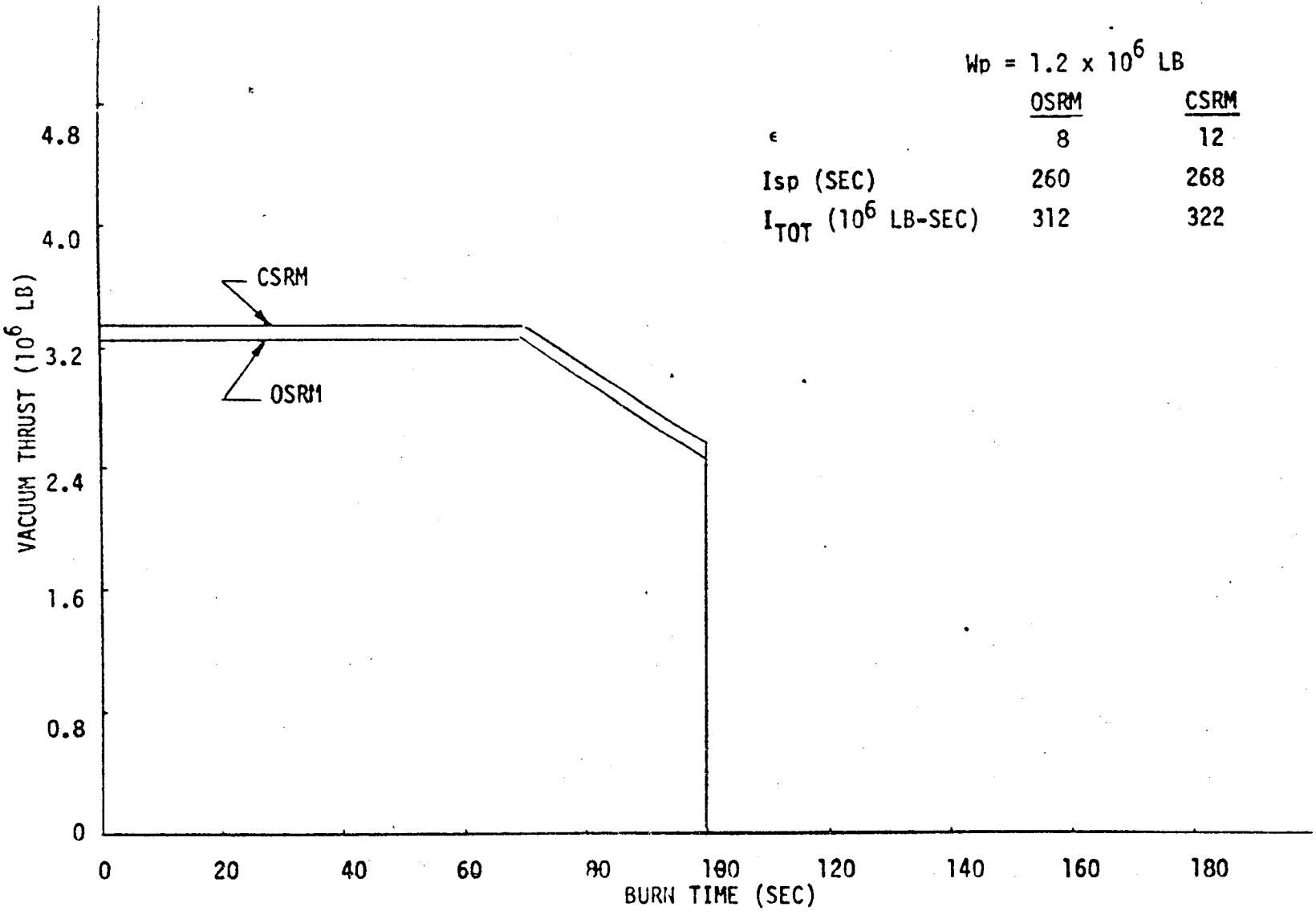

Figure 4-2 SRM THRUST HISTORY - TWO STAGE APPLICATION

4.2 S-IVB CHARACTERISTICS

The thrust profile for the S-IVB stage involves a two step approach rather than the single, nominal mixture ratio/thrust level. Initially, a mixture ratio of 5.5:1 is used, providing a vacuum thrust of 230,000 pounds. At approximately 228 seconds of burn, the mixture ratio is shifted to a value of 4.5:1, resulting in a vacuum thrust of 190,000 pounds. The stage is flown at this latter value to burnout, about 476 seconds. The specific impulse values for these two steps are 425 seconds and 430 seconds, respectively.

The S-IVB useful propellant load was assumed to be 228,700 pounds (123,900 pounds at 5.5:1 and 104,800 pounds at 4.5:1). Other pertinent weight data are as follows:

Dry Stage	22,300 lbs.
Residuals	2,480
FPR	2,000
IU (if applicable)	4,280
Weight at B.O.	31,060 lbs.

4.3 VEHICLE PERFORMANCE SUMMARY

Performance of the 156-inch SRM/S-IVB vehicle was determined for a number of options. As was outlined in the section on SRM characteristics, both a one stage and a two stage boost application were considered. In the one stage application, all three SRM's were fired together, while in the two stage application, two (outer) OSRM's were fired first, followed by the (center) CSRM, a "2 + 1" mode.

In addition, two basic flight profiles were considered, a constrained profile and an unconstrained profile. In the constrained profile, the booster flight profile was assumed to be as it would be in a Space Shuttle Booster/Orbiter configuration, thus providing a somewhat exact simulation of those flight conditions. In this case booster burnout altitude and velocity were specified; the trajectory, therefore, was only simulated from the time of booster burnout

to orbit insertion (through S-IVB flight). Under these specified conditions, then, booster liftoff weight (BLOW) is not pertinent.

The other flight profile considered was an unconstrained mode in which the payload was optimized. The only restrictions considered in this case were to attempt to hold Saturn V trajectory limits to minimize impact on the S-IVB, i.e., max dynamic pressure (q max) of 740 psf and max axial acceleration (a_x) of 4.7 g's. Consequently, the trajectory was simulated from liftoff to orbit insertion and shaped for optimum performance, both with and without the dynamic pressure constraint. For these conditions, the BLOW was held constant at 4.1 million pounds.

The results of the various performance runs, both one and two stage boosters and constrained and unconstrained trajectories, are presented in Table 4-1.

Discussion of the data presented in this table follows.

The first case (1) presents constrained performance to a 100 n.mi circular orbit. Since trajectory simulation started at specified booster burnout conditions, BLOW and/or number of boost stages is not significant. As is noted, burnout altitude and velocity were specified at 180,000 and 6,000 fps, respectively. Along with this, maximum dynamic pressure was 650 psf and maximum accleration was 3 g's. Burnout flight path angle (γ, degrees from horizontal) was not specified; based on previous SRM booster trajectory runs, a value of 13.9 degrees was used. Also, in order to meet the specified booster burnout conditions, a gross weight of about 1.1 million pounds was necessary above the booster. Thus it would be necessary to carry some dead weight (ballast) with the vehicle that could be dropped at S-IVB stage separation. Values for this ballast, which could be contained in the booster to S-IVB interstage, are noted in the table. As is shown in the table for case 1A, payload to orbit was about 21,000 lbs. Since the flight path angle of 13.9 degrees was more applicable to a 50 x 100 n.mi. orbit, a flight path angle of 30 degrees was also investigated (case 1B), and the resulting performance was increased to about 26,000 lbs.

Case 2 in the table presents the results of an investigation of burnout velocity effects. The orbit used was 50 x 100 n.mi., and velocities of 6,000, 8,000 and

Table 4-1

(2 + 1) 156-IN. SRM/S-IVB LAUNCH VEHICLE PERFORMANCE

Case	Orbit (NM)	Booster Burnout			q_{max} (PSF)	a_{max} (g)	Ballast (10^3 Lb)	Payload (10^3 Lb)	Remarks
		Alt (10^3 Ft)	Vel (10^3 FPS)	γ (Deg)					
Constrained									Blow N/A
1A	100 x 100	180	6	13.9	650	3	819	21	
1B	100 x 100	180	6	30			814	26	Booster Burnout
2A	50 x 100	180	6	13.9			817	23	Cond. Specified
2B	50 x 100	200	8	13.9			800	39	
2C	50 x 100	200	9	13.8			791	49	
Unconstrained									Blow = 4.1M
3A	100 x 100	230	12.3	16	1535	5.0	N/R	90	1 Stg, Max Pld
3B		480	9.8	84	1040	5.0		45	1 Stg, Loft Traj
4A		260	13.8	11	940	4.7		119	2 Stg, Max Pld
4B		400	11	-	740	4.7		93	2 Stg, Loft Traj

9,000 fps were considered. Resulting payloads were 23,000, 39,000 and 49,000 lbs, respectively. Note that required ballast is still significant.

Cases 3 and 4 present the results of the unconstrained performance runs, i.e., full trajectory simulation from liftoff with unspecified booster burnout conditions. All of these cases were run for a 100 n.mi. circular orbit, and a specified BLOW of 4.1 million pounds. Case 3 was for the one stage applications, and case 4 for the two stage application.

In the first of these cases, 3A, maximum performance of about 90,000 lbs was obtained, but the resulting max dynamic pressure was prohibitive. In an effort to reduce this q, a lofted trajectory was considered (case 3B). As a result, payload degraded to about 45,000 lbs but the q was still excessive. With the given thrust-time history employed (Figure 4-1), the only other means of reducing this dynamic pressure would be to load ballast. Of course shaping the SRM thrust-time curve would probably be the most effective method. Time did not permit exploration of this option.

Similar results are presented for the two stage (2 + 1) configuration by cases 4A and 4B. Case 4A is that of maximum performance, which turns out to be about 119,000 lbs. Max dynamic pressure is high, though. With a lofted trajectory, however, (case 4B) max dynamic pressure is reduced to a Saturn V level, 740 psf, and performance is still respectable, about 93,000 lbs. Clearly, then, a staggering of the SRM burning is the more advantageous flight mode.

Section 5
COST SUMMARY

A number of options exist for derivation of an S-IVB stage for use in the SRM booster. As previously noted, a number of stages exist as "excess hardware;" however, these could become assigned, in which case another stage would have to be assembled from some long lead items, existing excess components and considerable new construction. In the event a series of SRM-boosted vehicles were needed, S-IVB stage production would be reopened to produce a low cost version stage.

Various cost data for these three basic options are presented in Table 5-1. For option number 1, wherein an existing S-IVB and IU are reassigned to this mission, the total cost is $11.8 million (assuming the S-IVB and IU as GFE). For production of a new stage utilizing the small amount of available spare parts (option number 2), the cost would be approximately $42.3 million. Considering option 3, wherein a series of twelve low cost S-IVB stages are produced, the unit cost is estimated to be $22.5 million. Further discussion of these three categories is offered in the following paragraphs.

5.1 OPTION 1 - EXISTING S-IVB

In this option, an existing Saturn 1B/S-IVB stage and Instrument Unit are obtained as GFE from existing Saturn inventory. A number of minor changes would be required for the SRM-boosted vehicle application. S-IVB changes are expected to be primarily electrical, and are estimated at $0.3 million. This assumes that there would be no requalification of critical components required. This requalification due to the increased acoustic and vibration environment from the SRMs, if required, would cost as much as $2.5 million.

A number of minor changes would have to be accomplished by IBM in the IU, such as minor logic changes, FCC changes, and malfunction logic changes. These are estimated to cost $0.2 million.

Table 5-1

(2 + 1) 156-IN. SRM/S-IVB LAUNCH VEHICLE COSTS

(MILLIONS OF DOLLARS)

Cost Element / Option	1) Existing S-IVB and IU	2) New Stage With Spare Parts	3) New Buy-Low Cost S-IVB
S-IVB Stage/J-2 Engine	0.3	31.0	14.6
Astrionics	0.2	Incl Above	Incl bove
Payload Fairing	5.0	5.0	5.0
Integration and Assembly with SRM Booster	3.0	3.0	1.0
Launch Operations and Data Reduction	2.1	2.1	0.7
Mission Support	1.2	1.2	1.2
Total	11.8	42.3	22.5
Production Quantity	1	1	12
Launch Rate (Per Year)	1	1	4

NOTES: 1) Assumes no requalification of existing hardware

2) Nonrecurring startup and development costs included

The nose fairing assumed for all three options is similar to a Skylab shroud, but need only be about half-weight. It would include a standard double-angle nose cone and adapter for the S-IVB stage. The cost is estimated to be $5 million.

The costs for integration and assembly with the SRM first stage (assumed supplied fully assembled and with an interstage) is estimated at $3.0 million, and $2.1 million is estimated for launch operations and data reduction. Both of these costs are based on this being a one-shot affair (production quantity of one, launch rate of one/year), and thus involve support of a KSC crew for a full year.

Mission support costs, including those of the J-2 engine contractor, are constant for all three options at $1.2 million.

5.2 OPTION 2 - NEW STAGE WITH SPARE PARTS

Option number two is based on building a new S-IVB/1B stage utilizing those long lead time items and spare parts which are available (estimated to be no more than about $2 million worth). The cost for such a stage (production quantity of one) is estimated to be $31 million, including nonrecurring startup and development costs. For this option, no IU would be built, but rather, necessary Saturn type astrionics for the simple earth-orbit mission would be included within the S-IVB stage-forward skirt. Other costs for this option are the same as those for the previous one, since again a production quantity of one and a launch rate of one/year are involved.

5.3 OPTION 3 - NEW BUY - LOW COST S-IVB

The third option considered is that involving a new buy of twelve low-cost S-IVB stages, with one taken out for the SRM booster test vehicle. This configuration is greatly simplified, with various subsystems deleted or modified to reflect the simple earth orbital mission with brief stage lifetime. A simplified astrionics system is included in the stage. The stage unit cost for this configuration is estimated to be $14.6 million including nonrecurring start-up and development costs. Dual-burn capability would involve about $0.5 million additional per stage.

In this case, the production quantity is twelve, and a launch rate of four per year is assumed. Thus integration and assembly costs are estimated at $1.0 million per stage, and launch operations and data reduction costs at $0.7 million per stage. Mission support costs remain at $1.2 million.