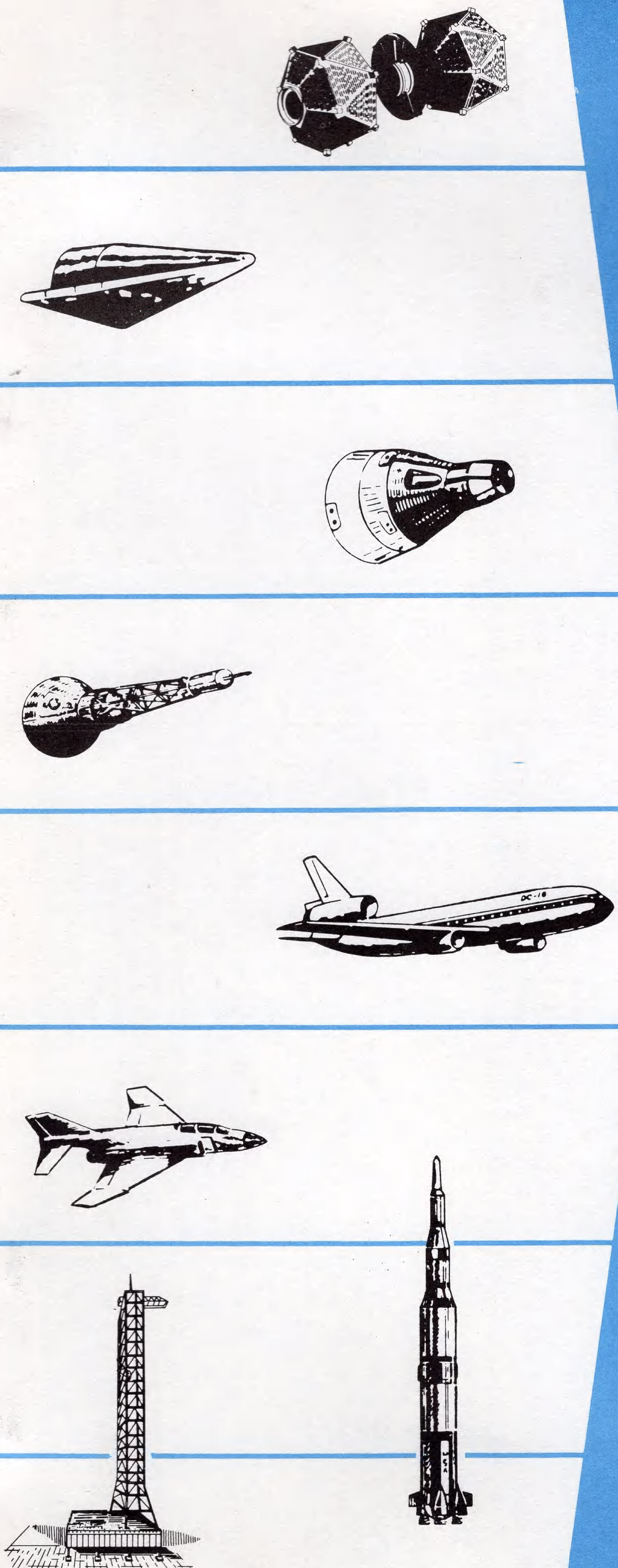

MDC E0558

Phase B System Study Extension

FINAL REPORT

Part I

EXECUTIVE SUMMARY

HIGH VALUE

SPACE SHUTTLE

LOW COST

MCDONNELL DOUGLAS CORPORATION

TRW SYSTEMS GROUP

PAN AM

COPY NO. 372

SPACE SHUTTLE PHASE B
SYSTEM STUDY EXTENSION
FINAL REPORT

15 MARCH 1972 **MDC E0558**

Part I

EXECUTIVE SUMMARY

Submitted Under NASA
Modification 24
Supplemental Agreement
to Contract NAS 8–26016
CDRL Item MA–04

MCDONNELL DOUGLAS CORPORATION
TRW
PAN AM

FOREWORD

The Space Shuttle Phase B Extension Study was conducted under Modification 24, Supplemental Agreement to Contract NAS 8-26016 by a team headed by the McDonnell Douglas Corporation. The period of performance for the study was 1 November 1971 through 15 March 1972. Members of the study team included the McDonnell Douglas Corporation, the Martin Marietta Corporation, TRW Incorporated, Pan American World Airways Incorporated, and Sperry and Norden as subcontractors.

This document constitutes Part I of the final documentation for the study which consists of the following:

CDRL Item MA-04 Final Report
- Part I Executive Summary
- Part II Technical Report
  - Volume I System and Orbiter
  - Volume II System and Booster
  - Volume III MMC Activity
- Part III Development Requirements
- Part IV Cost and Scheduling Data

CRDL Item SE-03
- Part V Design Data Book
  - Volume I Program and System Baseline
  - Volume II Drawings
  - Volume III Orbiter Aero Design Data Book
  - Volume IV Booster Aero Design Data Book (CDRL Item SE-01)

CDRL Item SE-04
- Part VI Final Mass Properties Report

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1 INTRODUCTION

The most promising candidate Shuttle configurations to evolve from the Phase B extension study are:

- Series Burn Pressure-Fed Booster with a large Orbiter payload (15 ft dia by 60 ft)
- Series Burn Pump-Fed (F-1) Booster with a large Orbiter payload
- Parallel Burn Twin SRM Booster with a large Orbiter payload
- Parallel Burn Twin SRM Booster with a smaller Orbiter payload (14 ft dia by 45 ft)

The study of candidate systems emphasized both minimum total costs and minimum annual funding. Expendable Boosters were first studied as interim systems with the intent to expand the system midway through the program with a fully reusable (flyback) Booster. Because of its costly development the fully reusable Booster was eventually dropped and emphasis was placed upon a water recoverable (non-flyback) Booster.

A parallel burn twin 156-inch SRM Booster with a large cargo bay Orbiter is currently the recommended system. This system is illustrated in Figure 1 and, with a thrust-to-weight ratio of 1.405, is sized to deliver 65,000 lb into an Easterly orbit and return with 40,000 lb. The recommended system and its primary characteristics and costs are shown in Figure 2.

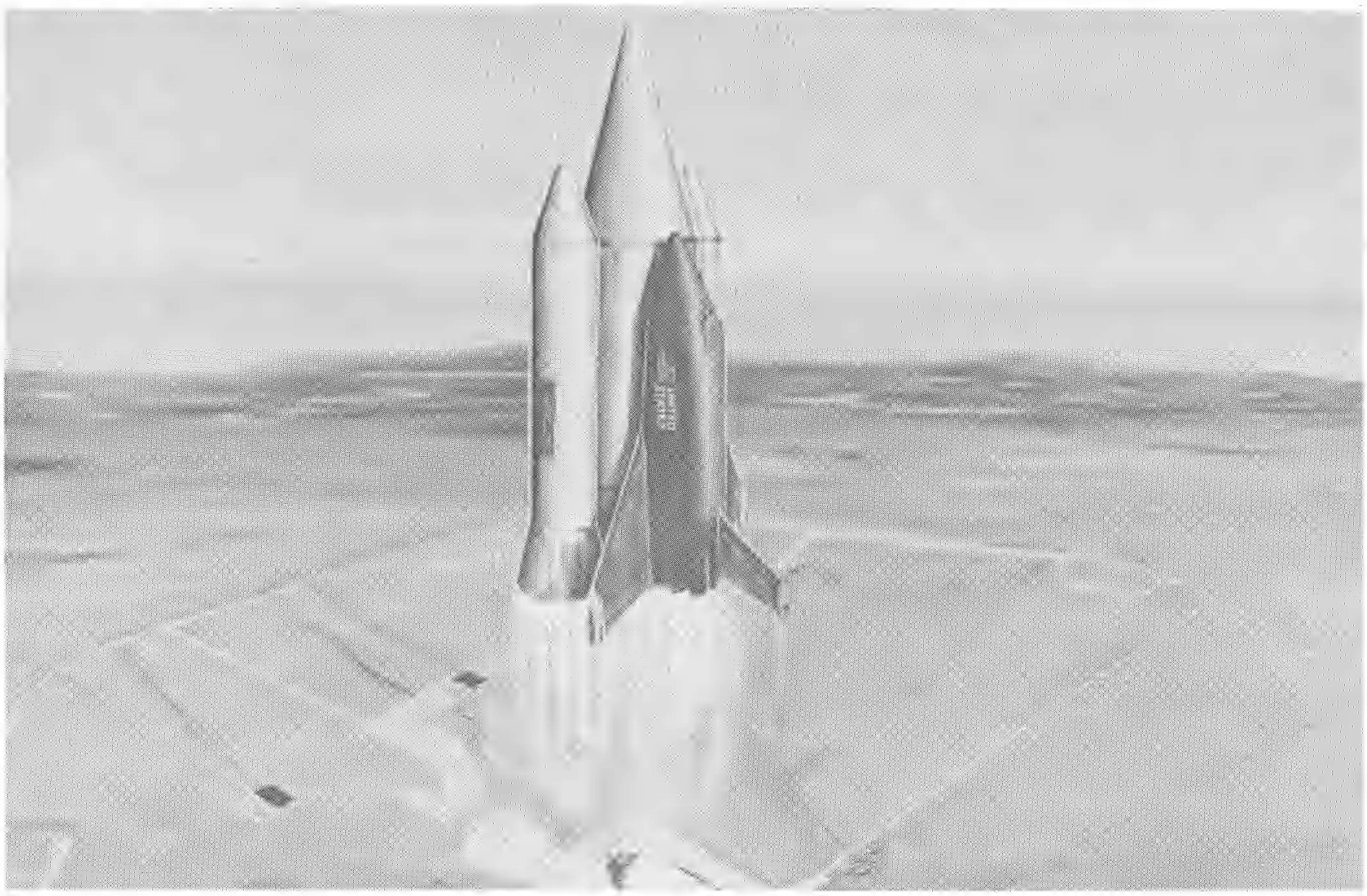
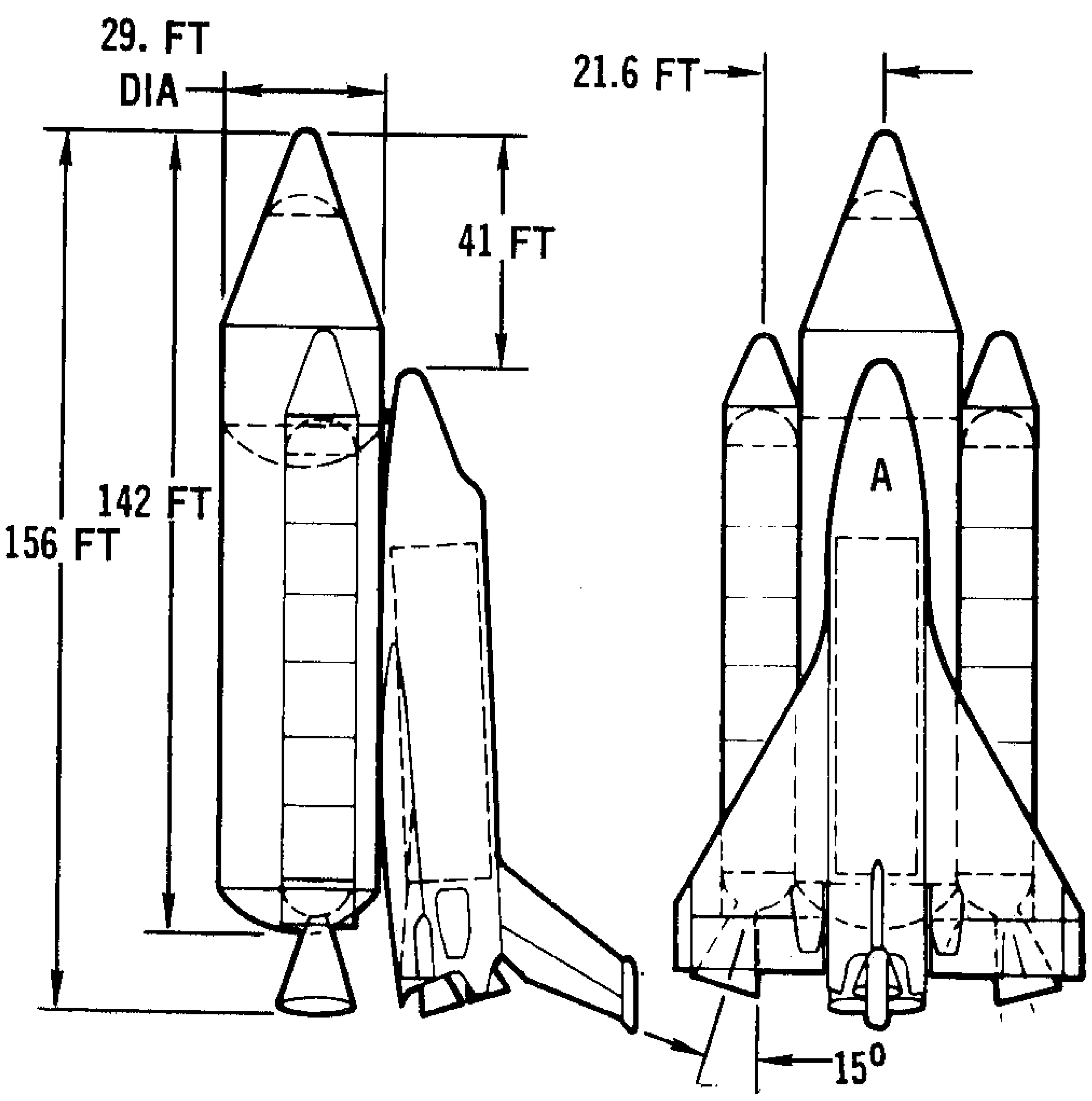


Figure 1

RECOMMENDED SYSTEM
Twin – 156 In. Parallel Burn SRMs

- OFF-THE-PAD ABORT CAPABILITY
- RECOVERABLE BOOSTERS
- LARGE ORBITER, 15 X 60 BAY
- ORBITER ENGINES: 3–470 K LB



• WEIGHTS (K LB)	65
PAYLOAD (EAST)	
GLOW	4,617
OLOW	1,968
BLOW	2,649
• REL STAGING VEL (FPS)	4,000
• COSTS ($M)	
TOTAL PROGRAM	9,816
RDT&E	4,313
PER FLIGHT (TOTAL)	9.49

Figure 2

2 CANDIDATE SYSTEMS

In the four Shuttle systems selected for main emphasis, economic and technical considerations were evaluated and compared. The prime parameters which bounded these considerations were configuration characteristics, ascent trajectories, performance and control, Orbiter selection and Booster selection. Comparisons were also made with the two 120-inch SRM parallel burn systems which were rejected because of high program and pre-flight costs.

2.1 **Configuration Characteristics** – A comparison of the four main Booster system candidates is summarized in Figure 3. The parallel burn twin 156-inch SRM vehicles are the lightest and the smallest systems but requires large HO tanks. Here the parallel burn system weights are based on expendable SRM's. Figures 4 and 5 illustrate the 156-inch mated configurations and compare the effect of two different sizes of payload bay dimensions.

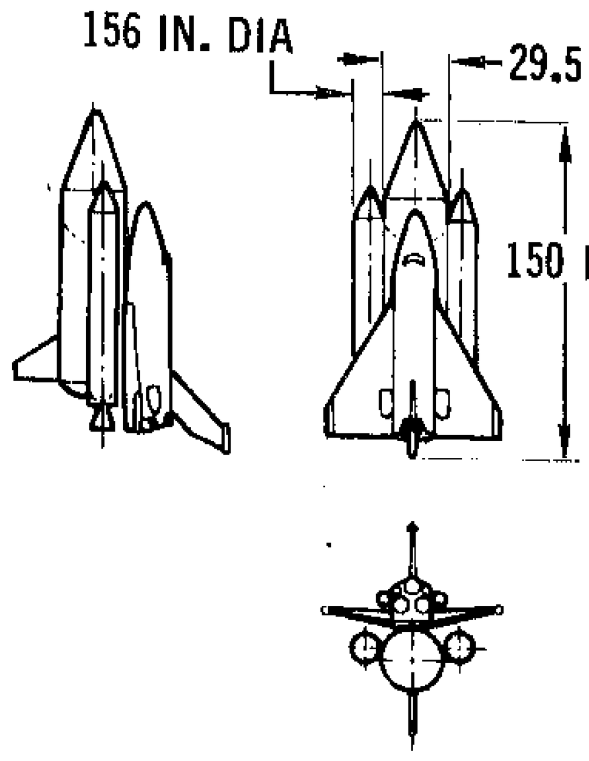
COMPARISON OF MAIN EMPHASIS SYSTEMS

BURN MODE	SERIES		PARALLEL	
ENGINE	PRESS. FED	F–1	2–156" SRM	2–156" SRM
BAY SIZE	15 x 60	15 x 60	15 x 60	14 x 45
STAGING VELOCITY (FPS)	4000	5000	4000	4000
WEIGHTS (K LB)				
PAYLOAD (EAST)	65.0	65.0	65.0	45.0
GLOW	5777.6	4458.3	4315.2	3617.4
OLOW	1512.2	1341.7	1898.6	1691.2
USABLE PROPELLANT	1196.6	1031.2	1577.9	1407.7
ORBITER DRY	148.8	148.8	148.8	139.1
TANK DRY	58.1	53.5	64.8	60.3
BLOW	4265.4	3116.5	2416.5	1926.2
USABLE PROPELLANT	3436.9	2687.3	2 x 1,064.5	2 x 848.5
BOOSTER DRY	734.9	372.3	(2 x 143.8)	(2 x 114.6)
$(T/W)_1/(T/W)_2$	1.35/0.934	1.35/1.053	1.405/0.927	1.405/1.071
COSTS ($\bar{M}$)				
TOTAL PROGRAM	10,798	10,002	10,207	9,681
RDT&E	5,676	5,018	4,126	4,006
PEAK FUNDING (YEAR)	1,263(76)	1,135 (76)	946 (76)	917 (76)
PER FLIGHT (TOTAL)	7.50	7.56	10.49	9.83
OPERATIONS	4.65	5.17	3.80	3.78
TANKS	1.37	1.28	1.49	1.41
EXPENDED BOOSTER HARDWARE	1.48	1.11	5.20	4.64

ALL CONFIGURATIONS HAVE 3–470 K HiPc ORBITER ENGINES

Figure 3

PARALLEL BURN
TWIN 156 IN. SRM BOOSTER
14 x 45 Payload Bay



- DESIGN CHARACTERISTICS

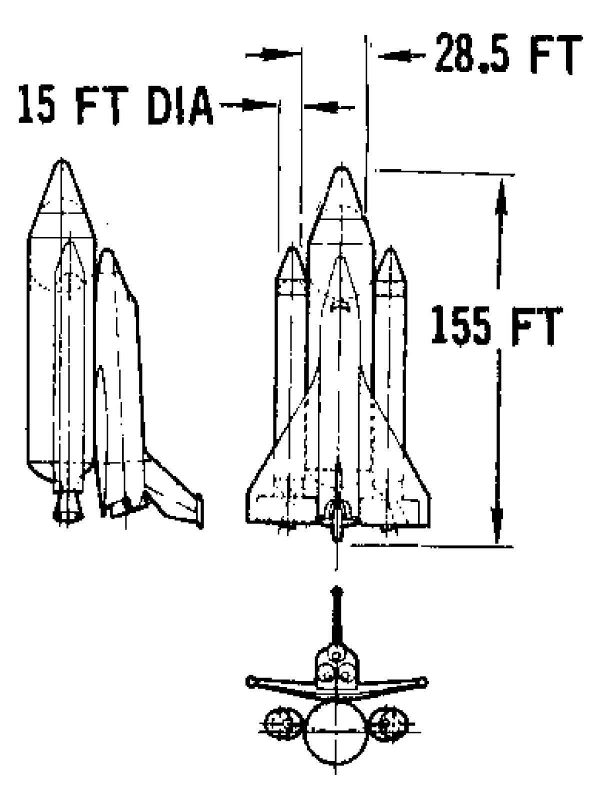
GLOW	3617 K LB
OLOW	1691 K LB
BLOW	1926 K LB

- BOOSTER DATA (PER SRM)

PROPELLANT WT	848 K LB
BURNOUT WT	115 K LB
MASS FRACTION	0.881
I_{SP} (VAC)	264 SEC
S.L. THRUST	2068 K LB
GIMBAL ANGLE	NONE

Figure 4

PARALLEL BURN
TWIN 156 IN. SRM BOOSTER
15 x 60 Payload Bay



DESIGN CHARACTERISTICS:

GLOW	4,315K LB
OLOW (INCL P/L)	1,898.6K LB
BLOW	2,416.5K LB

BOOSTER DATA:

USABLE PROPELLANT	2 x 1064.5K LB
BURNOUT WGT	2 x 143.8K LB
MASS FRACTION	0.881
I_{SP} (VAC) - MIN	264 SEC
S.L. THRUST	2 x 2,575K LB
GIMBAL ANGLE	NONE

Figure 5

2.2 **Ascent Trajectories** – Extensive ascent trajectory analyses performed for the Shuttle system concepts have provided performance verification of the sizing studies and insight into thrust shaping techniques required to satisfy q and g constraints. From this, reference trajectories were defined for control, separation, abort and Booster entry analyses, and for mission planning purposes. Figures 6 and 7 show altitude as a function of downrange distance and relative velocity for the preferred Parallel Burn Twin SRM 156-inch configuration. Figure 8 summarizes staging conditions and Booster impact range for the system configurations of main interest. All configurations were sized for 4000 fps staging velocity except for the Series Burn Pump-Fed F-1, which was sized for 5000 fps.

FLIGHT PROFILE
PARALLEL BURN – TWIN SRM (2 x 156 IN.)

- EASTERLY LAUNCH
- 15' x 60' P/L BAY
- MAX q = 650 PSF

	t	h	v	γ	q
	SEC	FT	FPS	DEG	PSF
A. ASCENT MAX q	70	39,160	1424	56.2	650
B. SRM REGRESSION	80	51,805	1774	49.3	590
C. SRM CUTOFF	120.3	118,590	4099	28.5	118
D. SRM APOGEE	180	172,800	3220	–0.9	–
E. ENTRY MAX q	330	23,300	584	–83.6	192
F. SRM IMPACT	369	0	–	–	–
G. ORBIT INJECT	515.7	303,980	24,545	0	–

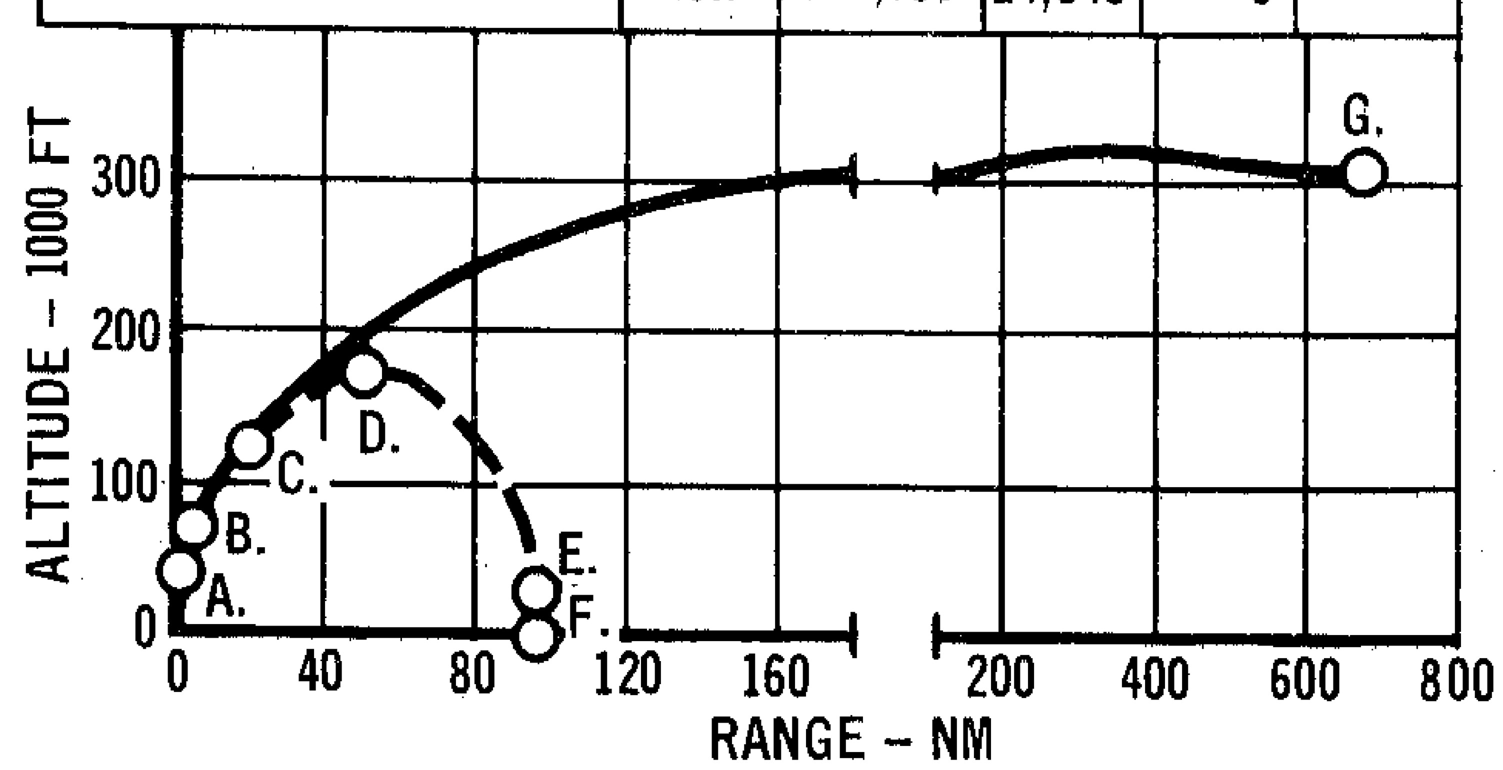


Figure 6

FLIGHT PROFILE
PARALLEL BURN – TWIN SRM (2 x 156 IN.)

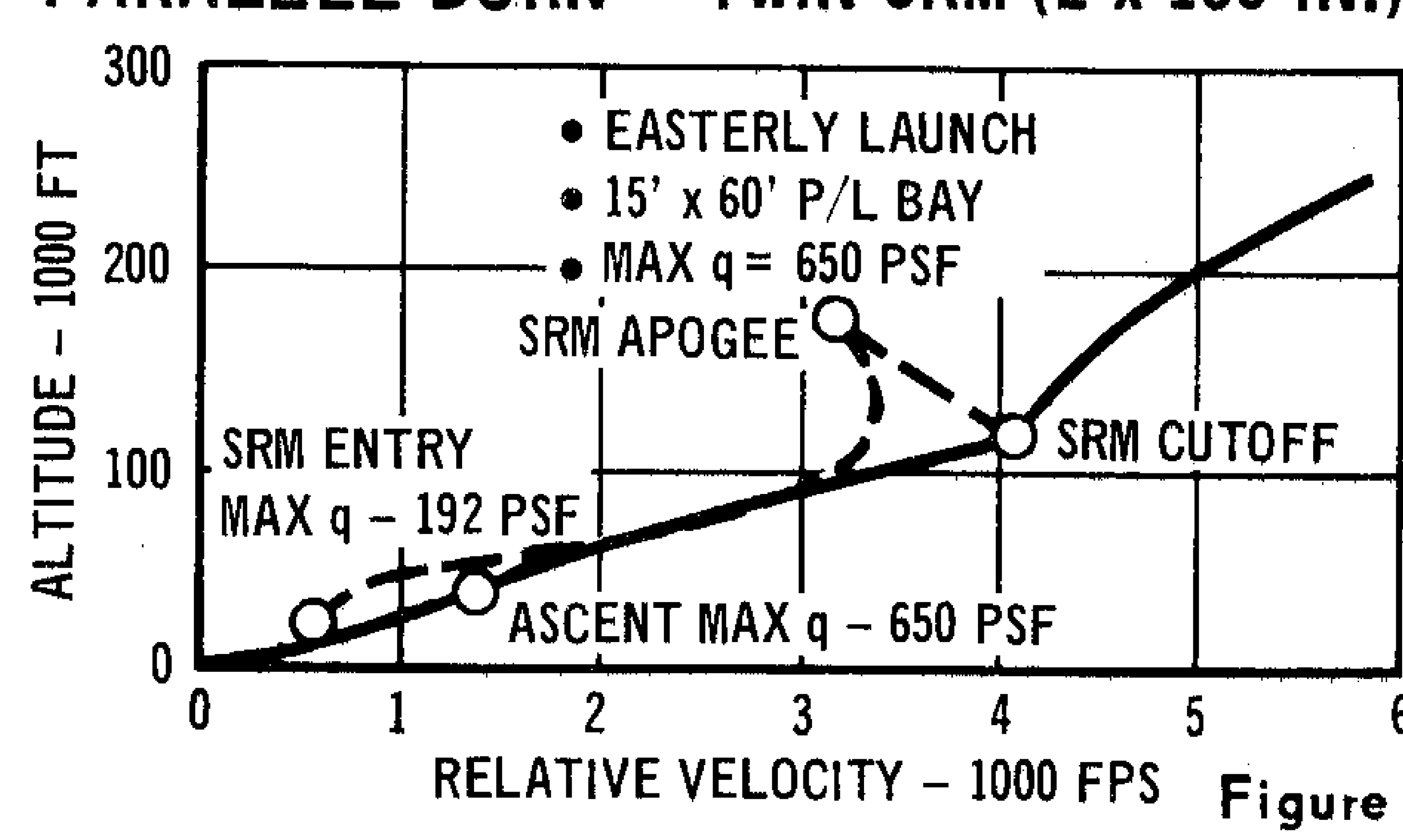


Figure 7

CONFIGURATION STAGING COMPARISON

- MAX q = 650 PSF

CONFIGURATION • KSC LAUNCH SITE • EASTERLY MISSION (i = 28.5°)	NOMINAL STAGING CONDITIONS						BOOSTER IMPACT
	TIME (SEC)	H (FT)	V_R (FPS)	γR (DEG)	R (NM)	Q (PSF)	DOWNRANGE (NM)
PARALLEL – SRM (2x156") 15' x 60' PAYLOAD BAY	120.3	118,590	4099	28.5	18.7	118	96
SERIES – PRESSURE FED 15' x 60' PAYLOAD BAY	112.0	112,798	3963	32.2	16.3	143	100
SERIES – F-1 15' x 60' PAYLOAD BAY	137.8	144,410	4913	21.5	27.9	55	156
PARALLEL – SRM (2x156") 14' x 45' PAYLOAD BAY	122.3	113,558	4069	25.0	19.3	145	83

Figure 8

2.3 Ascent Control – Response analyses of the ascent control systems have been performed for the parallel burn configuration with two 156-inch SRM's and for two series burn configurations, the Pressure-Fed Booster and the F-1 Pump-Fed Booster.

Control of the parallel burn system is derived from Orbiter thrust vector control (TVC), augmented by Oribiter aero surface control during the high dynamic pressure flight regime. This control authority was found to be adequate without resorting to the additional cost of TVC in the twin SRM Booster elements. Mated series burn control authority is provided by Booster TVC and Orbiter aero surfaces.

A comparison of the significant control characteristics of the parallel burn configuration versus the series burn configuration is presented in Figure 9. The most significant feature is that the parallel burn configuration is stable in pitch and yaw whereas the series burn concepts are either unstable or neutrally stable about these axes.

2.4 Abort – Analyses have concentrated on intact recovery of the Orbiter following an ascent phase abort, and methods for eliminating abort gaps have been determined. The abort forcing event was assumed to be failure of a Booster or Orbiter main engine. Aborts for other causes requiring immediate separation during the mated portion or the ascent phase were also considered. The abort technique to be used depends on flight conditions at the time of abort as shown in Figure 10. Abort separation modes for the parallel burn and series burn configurations are shown in Figures 11 and 12.

ASCENT CONTROL COMPARISON PARALLEL BURN SOLID (2–155 IN.) vs SERIES BURN

PARALLEL BURN SOLID (2–156 IN.) BOOSTER	SERIES BURN BOOSTER
• CONTROL WITHOUT TVC ON BOOSTER	• CONTROL REQUIRES TVC ON BOOSTER
• ROLL CONTROL REQUIRES 200 FT^2 VENTRAL FIN OR FREE ROLL MODE	• ROLL CONTROL REQUIRES YAW LOAD RELIEF WITH β FEEDBACK. ALTERNATES ARE A VENTRAL FIN, OR FREE ROLL MODE
• PEAK ROLL ANGULAR ACCELERATION (REGARDLESS OF CONTROL TECHNIQUE) IS 10 DEG/SEC^2 (0.14 g's AT PILOT'S STATION)	• PEAK ROLL ANGULAR ACCELERATION IS 25 DEG/SEC^2 (0.4 g's AT PILOT'S STATION) WITHOUT VENTRAL FIN.
• SENSITIVE TO MISALIGNMENT AND CG TOLERANCES	• RELATIVELY INSENSITIVE TO MISALIGNMENT AND CG TOLERANCES
• STABLE AIRFRAME IN PITCH AND YAW	• UNSTABLE AIRFRAME IN PITCH AND NEUTRALLY STABLE IN YAW

Figure 9

ORBITER ABORT CAPABILITY SUMMARY

- 3 HiPc ENGINES – 1410 K LB TOTAL THRUST
- EASTERLY MISSION
- 105–109%

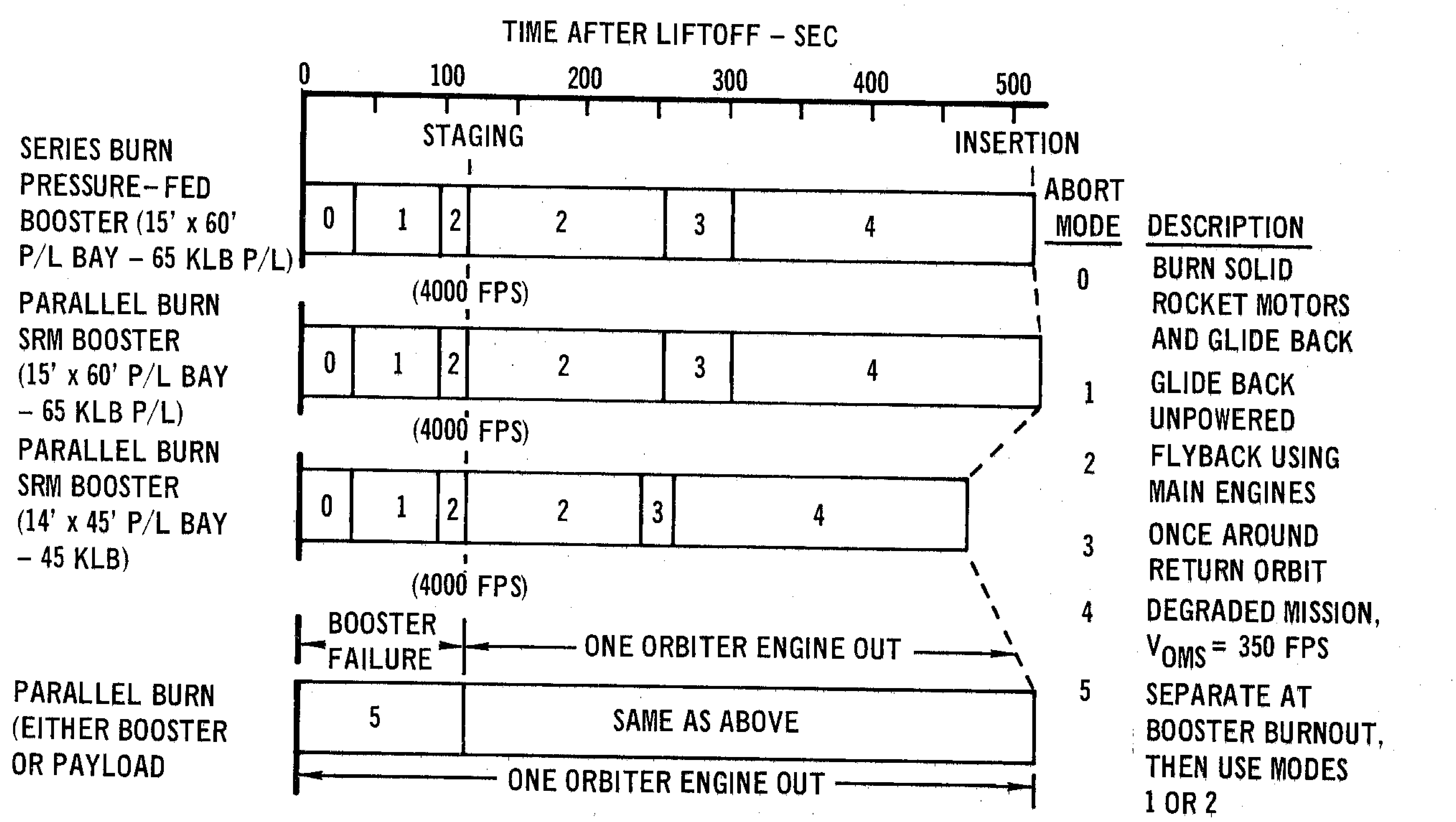


Figure 10

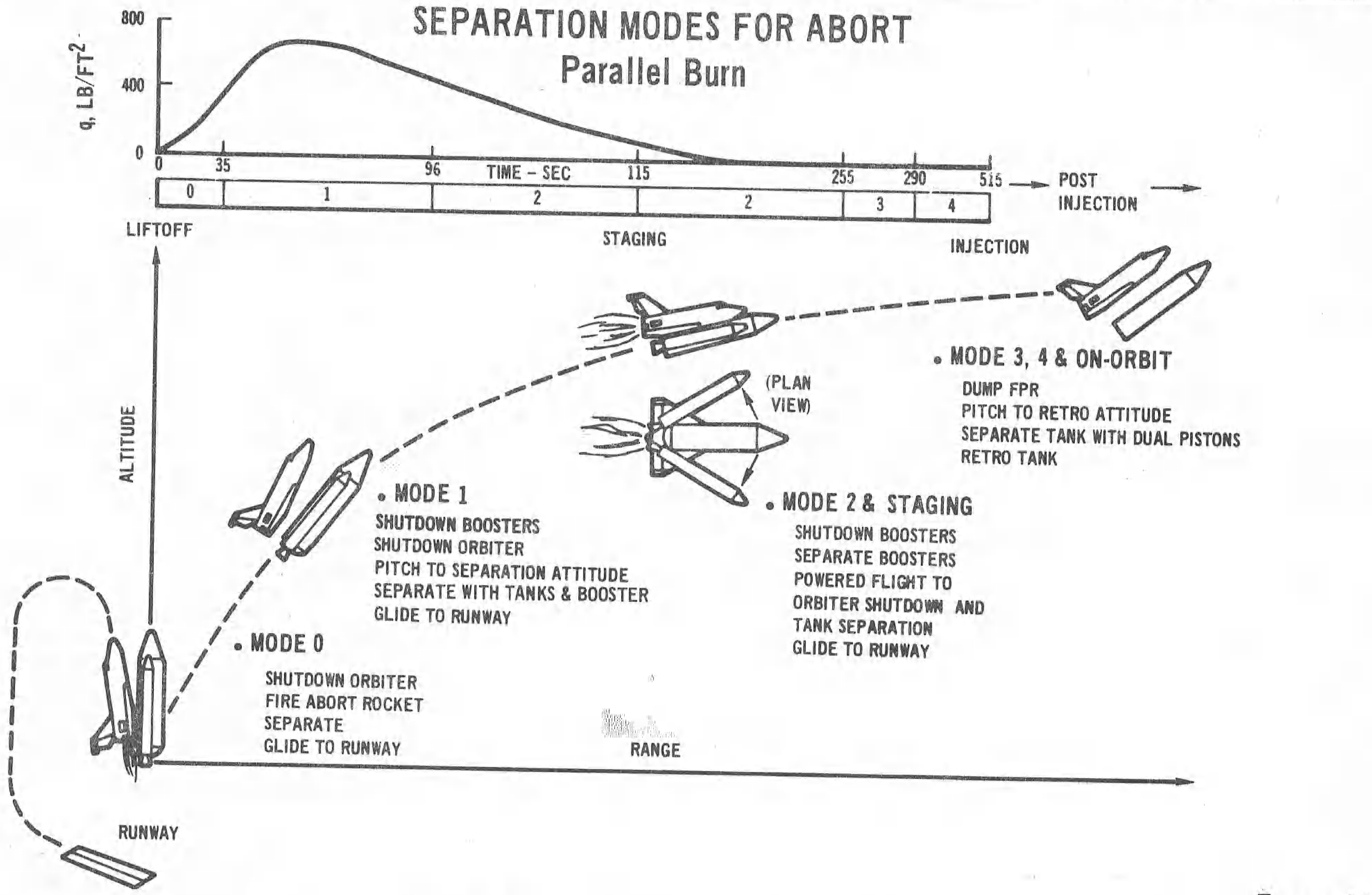


Figure 11

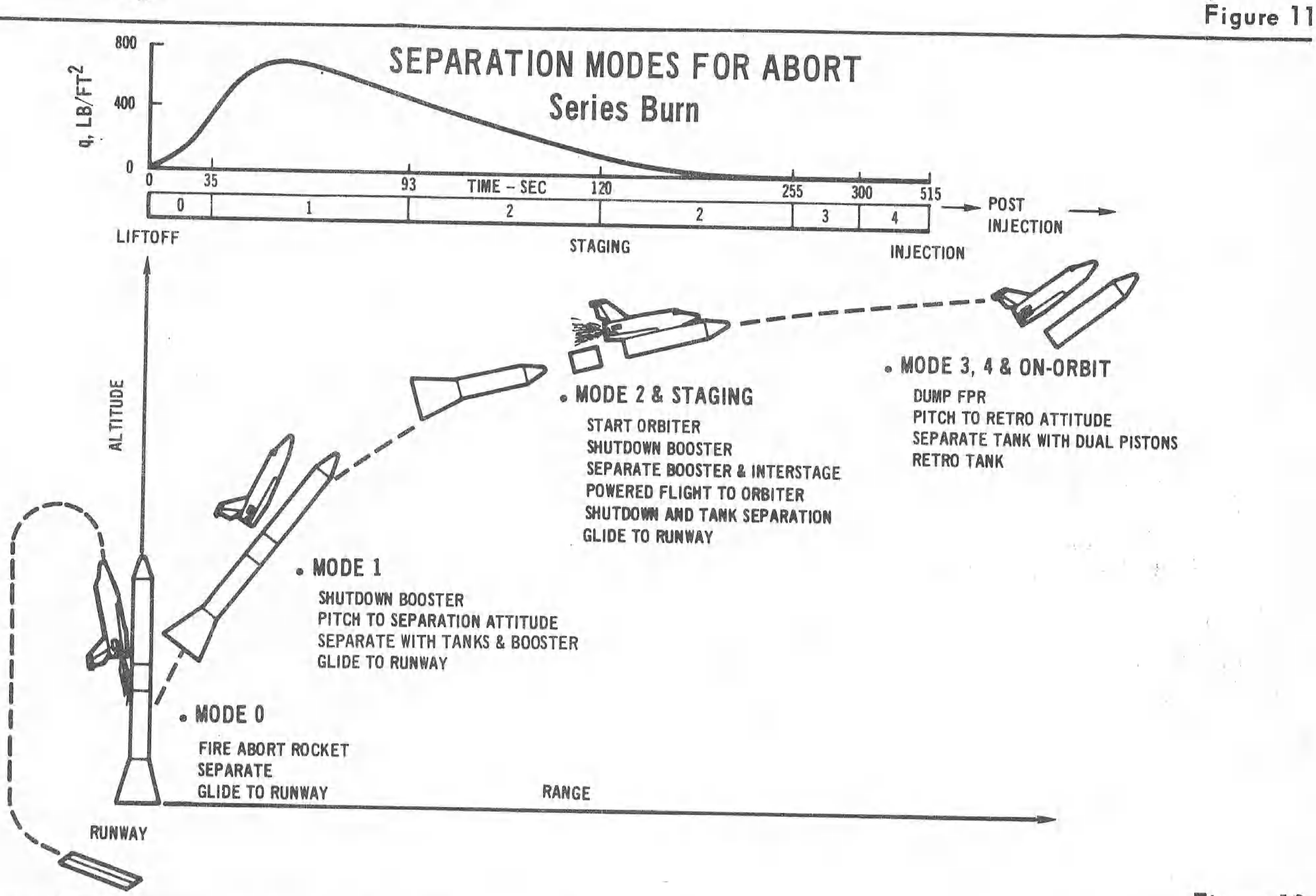


Figure 12

3 ORBITERS

Two basic orbiters were used in the extension studies for system analysis. The first evolved from the MSC 040A configuration with modifications needed to accommodate the change from 4-306K J2-S main engines to 3-470K HiPc engines. This configuration, referred to as the 068B and shown in Figure 13, has the large payload capability and was the subject of most of the detailed Orbiter analysis during this study period. The second configuration is the small payload Orbiter which is essentially the 068B shrunk to the smaller payload bay and retaining the 3-470K HiPc engines. This configuration, referred to as the 085A, has been analyzed only insofar as necessary to support system studies. Geometric characteristics for both the 068B and 085A Orbiters are shown in Figure 14. Several other Orbiter engine combinations have been examined to provide parametric data used in various trade studies.

3.1 **Aerodynamics and Performance** – Orbiter configuration 040A generic wind tunnel tests

ORBITER GENERAL ARRANGEMENT
15 x 60 Payload (068B)

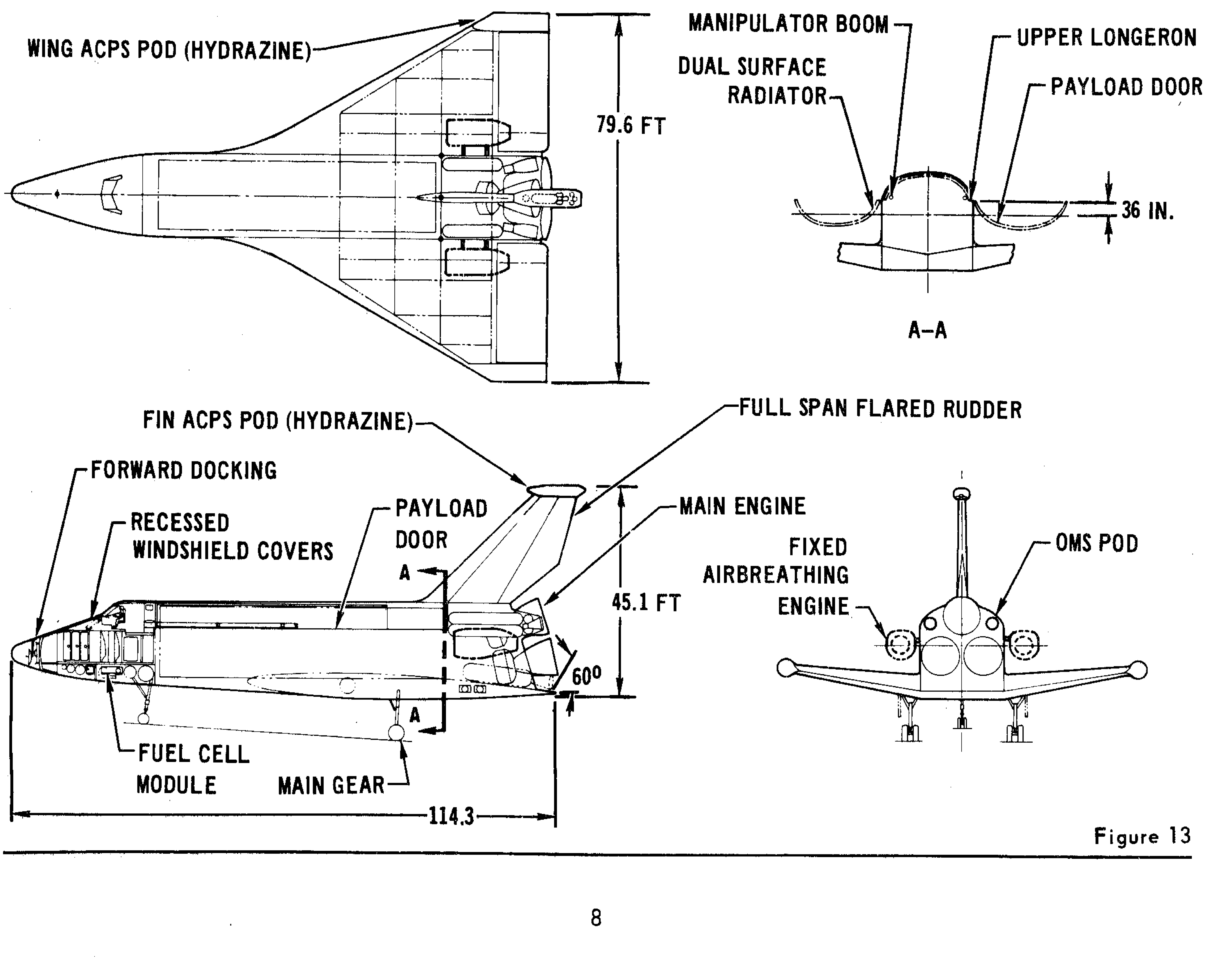


Figure 13

ORBITER GEOMETRIC DATA

	15 x 60 PL (068B)	14 x 45 PL (085A)
OVERALL CONFIGURATION		
LENGTH – FT	123.0	107.8
HEIGHT (LESS LANDING GEAR) – FT	45.1	44.0
ANGLE, MAXIMUM TAIL DOWN – DEG	17	17
FUSELAGE		
LENGTH, TO END OF BODY – FT	114.3	99.2
BASE AREA – FT^2	314	310
WING		
AREA, THEORETICAL – FT^2	3720	3310
SPAN – FT	79.6	75.2
ASPECT RATIO	1.7	1.7
DIHEDRAL – DEG	7	7
INCIDENCE – DEG	1.5	1.5
LEADING EDGE SWEEP – DEG	60	60
AIRFOIL SECTION – NACA	0008–64	0008–64
VERTICAL TAIL		
AREA, THEORETICAL – FT^2	434	418
ASPECT RATIO	1.43	1.43
LEADING EDGE SWEEP – DEG	45	45
AIRFOIL SECTION – NACA	0012–64	0012–64

Figure 14

ORBITER ENTRY TRAJECTORY
068B Configuration

100 NM POLAR ORBIT
V_0 INERTIAL = 25,655 FT/SEC
γ_0 INERTIAL = –0.83 DEG
h_0 = 400,000 FT
i = 90 DEG

1100 NM CROSSRANGE
$\alpha = 34^0$

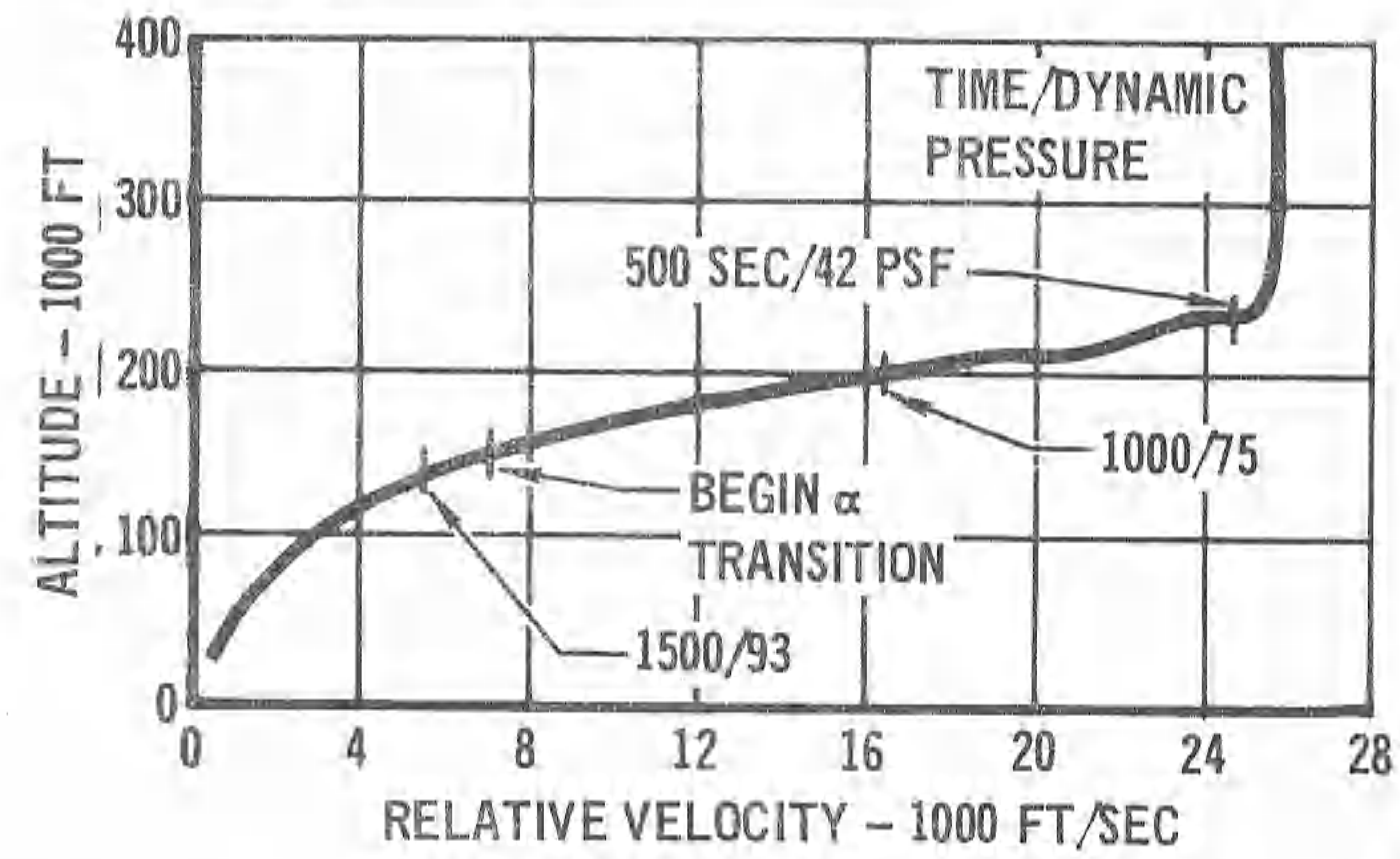


Figure 15

ORBITER PERFORMANCE CHARACTERISTICS

ENTRY (100 NM POLAR ORBIT)	
ANGLE OF ATTACK – DEG	34
CROSSRANGE – NM	1,100
INITIAL TRANSITION – MACH NO.	7
TIME – 400,000 FT TO 50,000 FT, MINUTES	32
SUBSONIC – UNPOWERED	
DESIGN LANDING WEIGHT – LB	192,659
APPROACH SPEED – KTS	245
LANDING SPEED – STANDARD DAY – SEA LEVEL, KT	175
TOUCHDOWN ANGLE OF ATTACK – DEG	12.2
FIELD LENGTH – FAA DRY – FT	9,550
SUBSONIC – POWERED (HOT DAY)	
GLIDE EXTENSION CAPABILITY – 3 ENGINES – NM	55
FUEL REQUIRED – LB	4,500
GLIDE EXTENSION CAPABILITY – ENGINE OUT – NM	15
FUEL REQUIRED – LB	1,100
FERRY MISSION (HOT DAY – 4 ENGINES AND JTO	
TAKEOFF WEIGHT – LB	208,000
FIELD EVALUATION – FT	4,000
BALANCED FIELD LENGTH – FT	13,000
INITIAL CRUISE ALTITUDE – ALL ENGINES – FT	15,500
INITIAL CRUISE ALTITUDE – ENGINE OUT – FT	6,400
AVERAGE CRUISE SPEED – ALL ENGINES – KTS	316
DESIGN CRUISE RANGE – NM	440

Figure 16

were used as the basis for estimating the aerodynamic characteristics and flight performance of the 068B and 085A configurations. The 068B Orbiter entry trajectory is shown in Figure 15 and performance characteristics are tabulated in Figure 16.

3.2 **Flight Control System** – The Orbiter post-staging stability, control, handling and performance characteristics required throughout its flight regimes are derived through combinations of main engine thrust vector control, reaction jet and aerodynamic surface control.

Orbiter ascent attitude control is provided by gimbaled main engines responsive to control logic commands. Information from the inertial navigation system is used to calculate a constant rate steering profile which achieves the desired injection state vector. Attitude error commands are converted to rate commands and summed with corresponding rates to form control commands.

On-Orbit transfer maneuvers, rendezvous sequence maneuvers, landing site selection and the timing for initiation of deorbit maneuvers are calculated by the guidance system using computer state vector information updated by the ground network and star tracker and horizon sensor measurements. Attitude control is provided by the Attitude Control Propulsion System thrusters in response to automatic and manual commands. Major ΔV maneuvers are performed with the Orbital Maneuvering System engines.

The reentry control system controls the vehicle to the entry flight path and responds to guidance commands for landing site acquisition. The guidance is targeted for a point approximately 750 nm from the runway and utilizes closed form range predictions to obtain the required L/D. Elevons and reaction jets are employed for pitch axis control with the jets deactivated when the dynamic pressure reaches 40 psf. Jets remain in use for lateral/directional control.

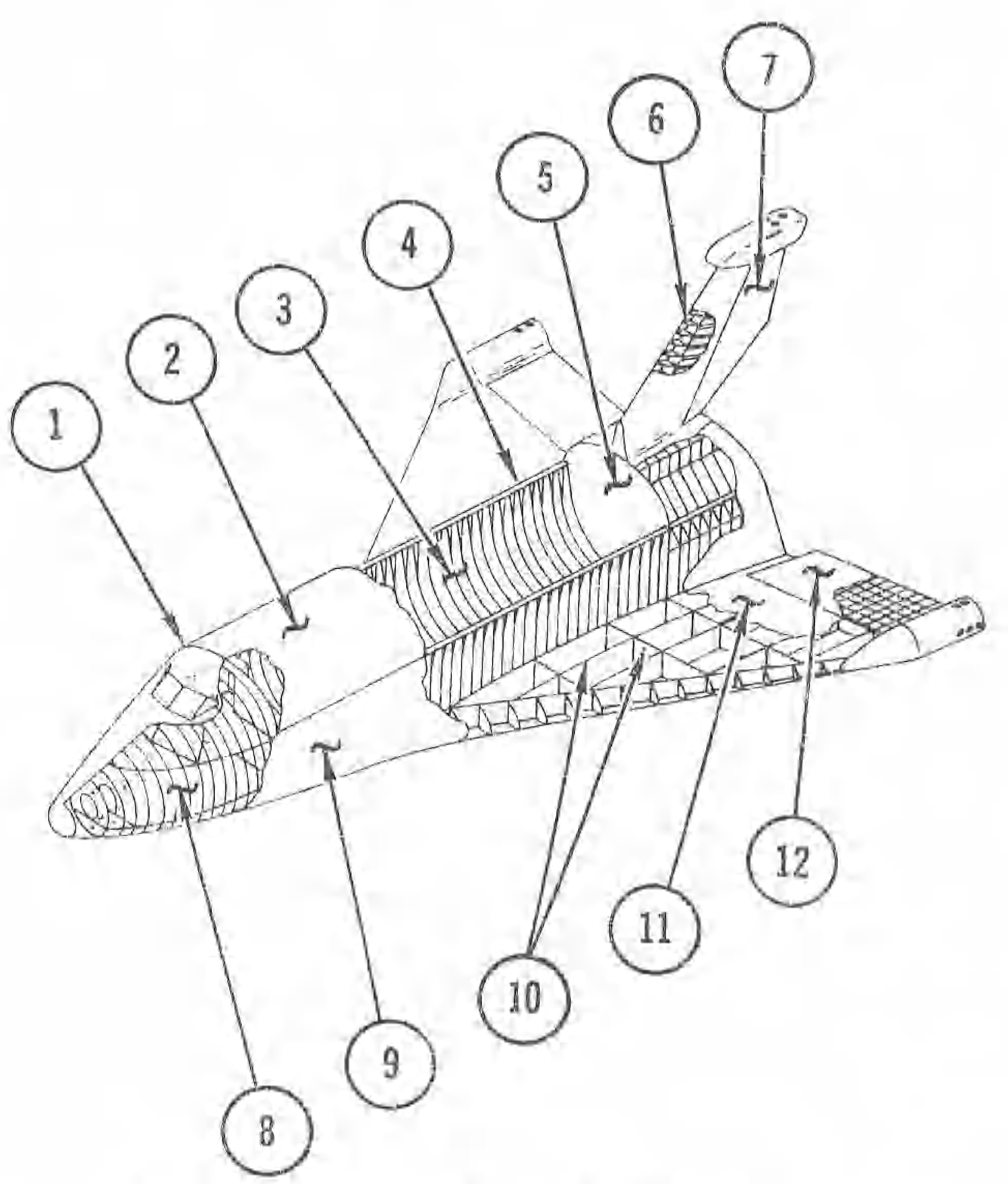
During the transition phase, the vehicle is maneuvered from the back side to the front side of the L/D curve. The vehicle responds to commands from the terminal guidance system during these pitch over maneuvers.

The pitch control system is unchanged for the transition and subsequent flight phase. However, the lateral-directional control concept is changed just prior to transition from all reaction jet to blended aero/reaction jet control.

Terminal phase energy management is accomplished through angle of attack, bank angle, and speed brake modulation. The vehicle is controlled entirely with aerodynamic controls during this phase of flight. Manual pitch and roll commands are introduced through a two axis control stick and yaw commands through the rudder pedals with automatic turn coordination provided. Automatic speed brake modulation provides the essential speed control for the autoland system. The autoland system utilizes the two step approach path which combines the high energy approach with the conventional final approach.

3.3 **Structures** – The primary structure selected for the Orbiter is an all-aluminum, stiffened shell, airplane type structure illustrated in Figure 17. A combination of non-metallic reusable insulation and ablative Thermal Protection System (TPS) is used. The basic fuselage shell is formed by zee-stiffened skins protected by TPS which forms the aerodynamic moldline. The forward fuselage consists of a pressurized integral cabin structure formed by the stiffened moldline fuselage shell and the cabin floor. The center fuselage consists of the payload bay enclosed by a one-piece structural door, payload attach points, and wing carry-through structure. The aft fuselage is an extension of the basic fuselage shell and contains the engine thrust structure. Interconnect structure between the Orbiter and the external tank is provided at one forward and one aft location, each consisting of three attachment points.

STRUCTURAL DEFINITION



1. FORWARD PAYLOAD BAY BULKHEAD
2. STRUCTURAL PAYLOAD BAY DOOR
3. INTERMEDIATE AND MAJOR FUSELAGE FRAMES
4. DOOR SILL LONGERON
5. AFT PAYLOAD BAY BULKHEAD
6. VERTICAL TAIL RIBS AND SPAR
7. RUDDER
8. INTEGRAL CABIN
9. STIFFENED FUSELAGE SKIN
10. WING SPARS AND MAIN RIBS (INTERMEDIATE RIBS NOT SHOWN FOR CLARITY)
11. STIFFENED WING SKIN
12. ELEVON

Figure 17

The exposed portion of the wing is a four-spar torque box structure with ribs at 25-inch spacing and angle-stiffened cover skins. Wing carry-through structure located under the payload compartment is integrated with the fuselage lower shell structure and major fuselage frames. The vertical tail is a one cell, two spar torque box employing zee-stiffened skin for distributed bending material.

3.4 **Thermal Protection** – The Orbiter TPS is passive and consists of: reusable surface installation (RSI), used where surface temperatures are between 750°F and 2500°F (maximum temperature for extensive reuse); SLA-561 ablator, used where temperatures are below 750°F (no charring) and above 2500°F; and ESA-3560-II A ablator, used on the nose cap and leading edges, where temperatures are above 2500°F and where protection from rain erosion is needed. Figure 18 shows the distribution of TPS material on the Orbiter and Figure 19 illustrates the entry temperature distribution.

3.5 **Propulsion** – The Orbiter propulsion group consists of the main propulsion, reaction control, orbit maneuver, airbreathing, and solid propellant

TPS MATERIAL DISTRIBUTION

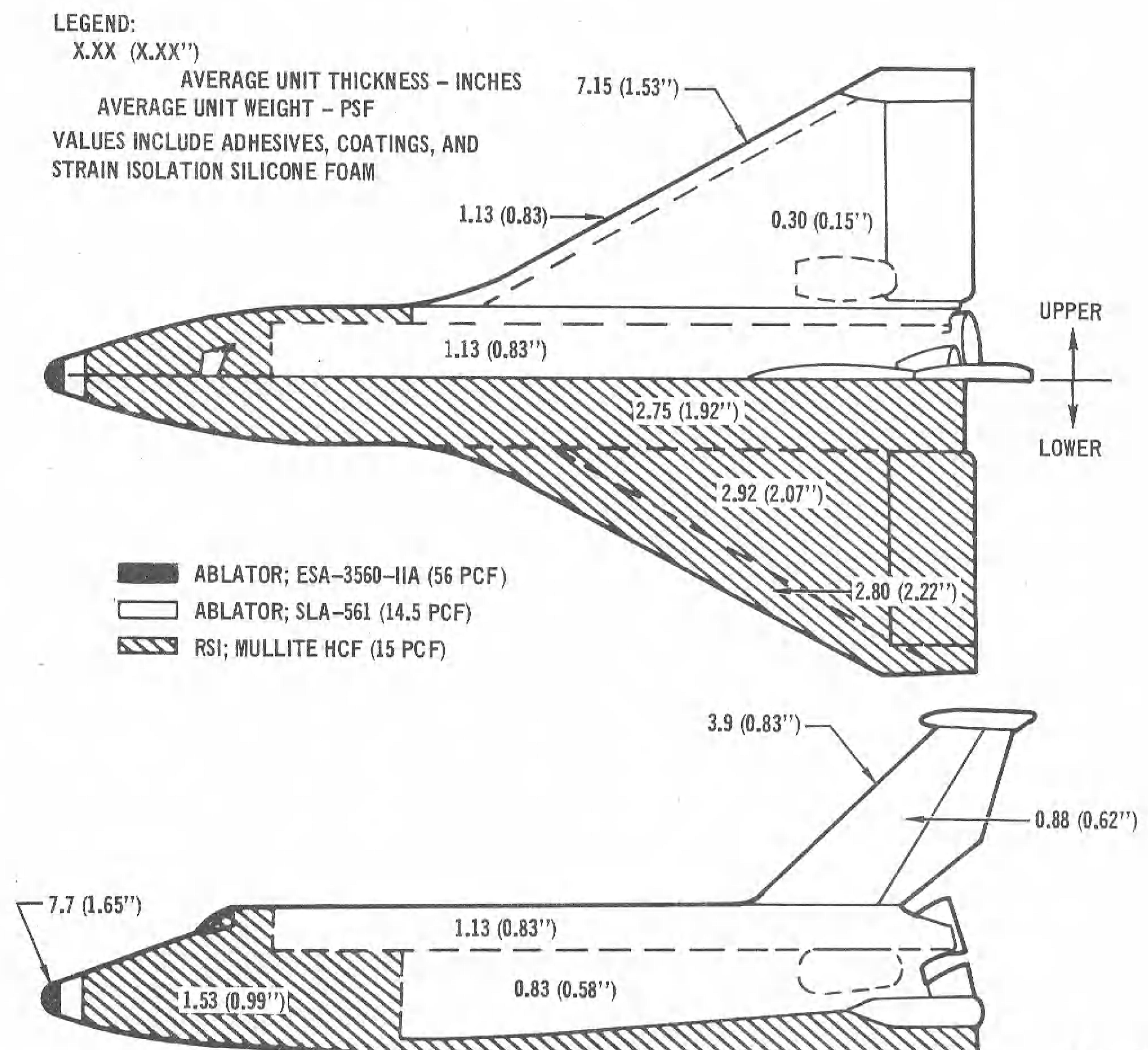


Figure 18

BASELINE ENTRY ISOTHERMS

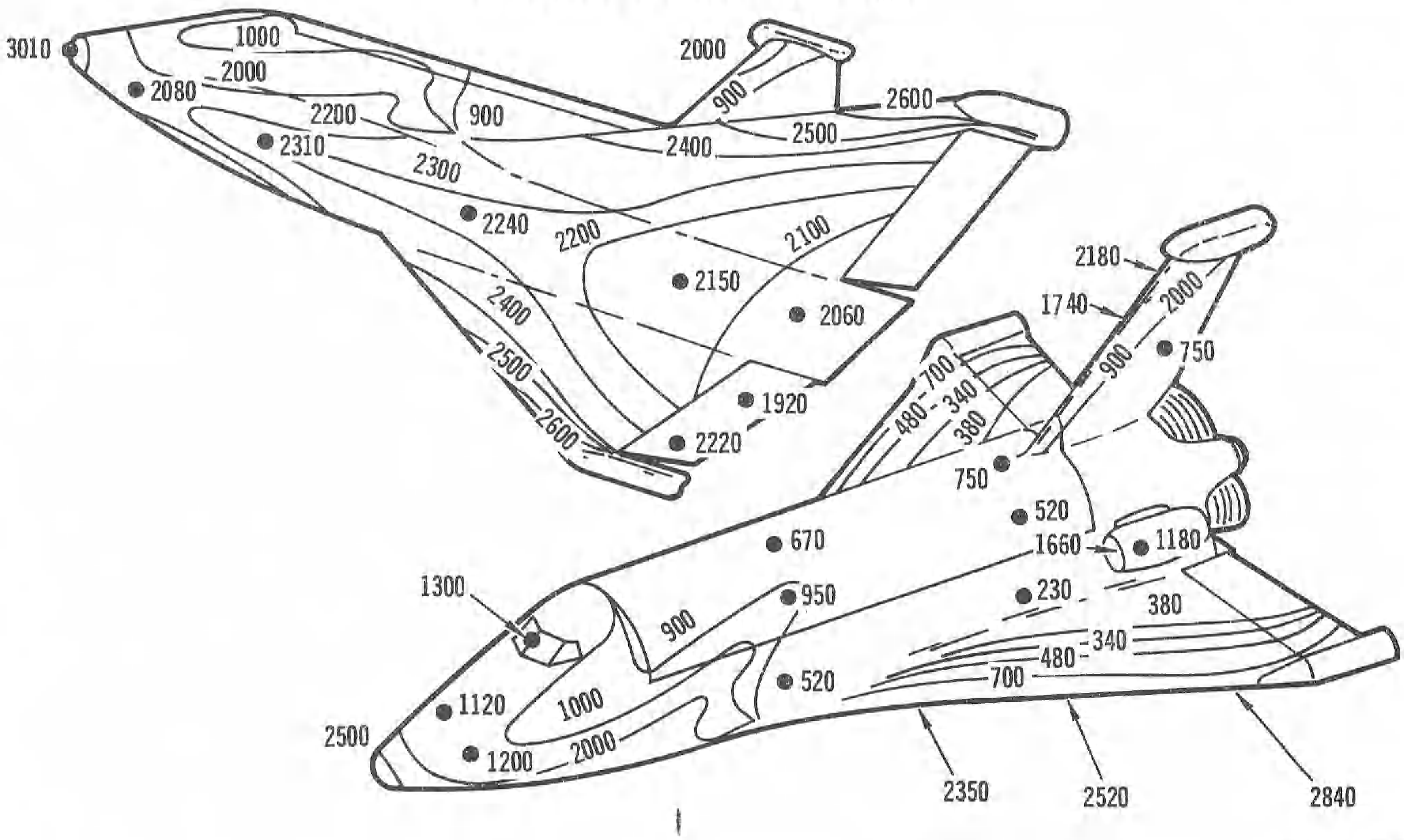


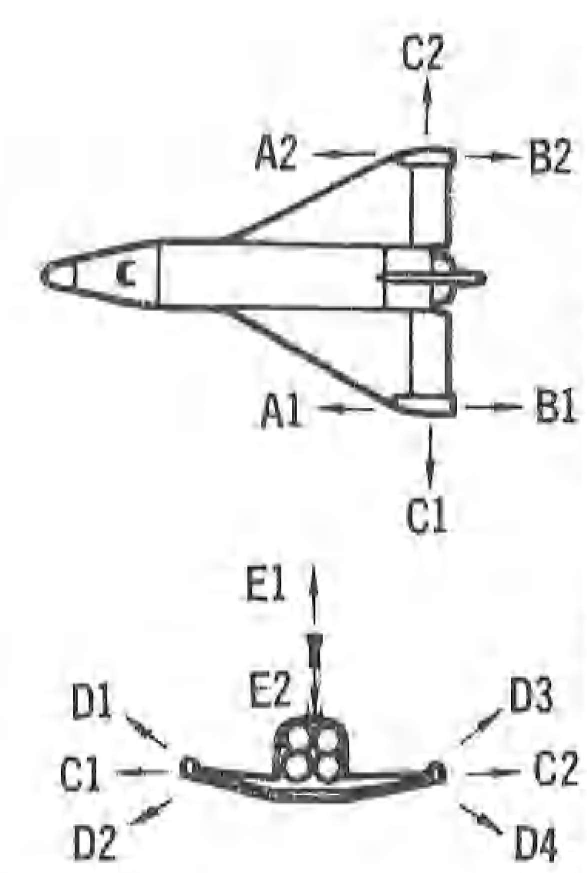
Figure 19

abort systems. These systems provide the propulsive energy required for Orbiter ascent, on-orbit and entry reaction control, landing assist, and off-the-pad abort.

The Main Propulsion Subsystem consists of three 472K lb (vac) thrust high Pc engines, an external tank containing liquid oxygen and liquid hydrogen propellants, and ancillary propellant supply and control subsystems. This system is designed to provide propulsive velocity to the Orbiter and contain, feed and regulate main propellant, oxygen and hydrogen, and inert fluids during all mission phases.

The Reaction Control Subsystem (RCS) controls the attitude of the Orbiter and also provides three axis vernier translation velocity. The system consists of three independent modules, each containing a group of engines, propellant, pressurant and associated controls. One module is located at the tip of the vertical tail and the two others are mounted one on each wing tip. The module location, number of engines, engine arrangement and function are shown in Figure 20.

BASELINE RCS POD & ENGINE ARRANGEMENT



CONTROL AXIS		ON ORBIT	REENTRY
PITCH	UP	E1	E1
	DOWN	E2	E2
YAW	RIGHT	A2, B1	A2, B1, C2
	LEFT	A1, B2	A1, B2, C1
ROLL	CW	D2, D3	D2, D3
	CCW	D1, D4	D1, D4
AXIAL TRANSLATION	FWD	B1, B2	–
	AFT	A1, A2	–
LATERAL TRANSLATION	RIGHT	C1	–
	LEFT	C2	–
VERTICAL TRANSLATION	UP	D2, D4	–
	DOWN	D1, D3	–

ARRANGEMENT		ENGINE NO./LOCATION				
THRUST	NUMBER	A	B	C	D	E
400	32	6	6	6	8	6

Figure 20

The Orbit Maneuvering Subsystem (OMS) provides the propulsive energy to preform all major velocity changes of the Orbiter from orbit insertion to retrograde. The system consists of two independent modules, located within the moldline of the Orbiter's base area, one on each side of the vehicle's centerline, each containing pressurant and propellant supplies, controls and plumbing, and a single OMS engine. The system uses storable bipropellants, nitrogen tetroxide and Aerozene-50. Two 3500 lb thrust modified Lunar Module Ascent Engines provide the total impulse required.

The Airbreathing Propulsion Subsystem is provided on the initial orbital flights and on the resupply orbital missions for glide range extension after deorbit. With 2000 lb of JP-4 the glide range can be extended about 24 nm. Airbreathing engines are also used during the horizontal flight tests and for ferry missions. With 40,000 lb of JP-4 the nominal ferry range is about 440 nm with a 25% fuel reserve. Ferry mission take-off is assisted with solid rockets to attain balanced field capability for hot day conditions.

The Solid Propellant Abort Subsystem for the 068A Orbiter consists of two solid propellant rocket motors producing a total of 800,000 lb sea level thrust. The motors burn for 6.5 seconds and loft the Orbiter to about 4000 feet off the launch pad. They are employed if an abort is necessary during the initial 40 seconds of ascent. After that time the abort motors are burned to furnish additional ascent velocity.

3.6 Avionics – The baseline avionics system architecture is summarized in Figure 21. Redundancies are noted in the lower right area of each equipment block. Components are grouped generally with the following subsystems:

- Aeroflight Control
- Spaceflight Guidance, Navigation and Control
- Communications and Tracking

SPACE SHUTTLE ORBITER AVIONICS SUBSYSTEM

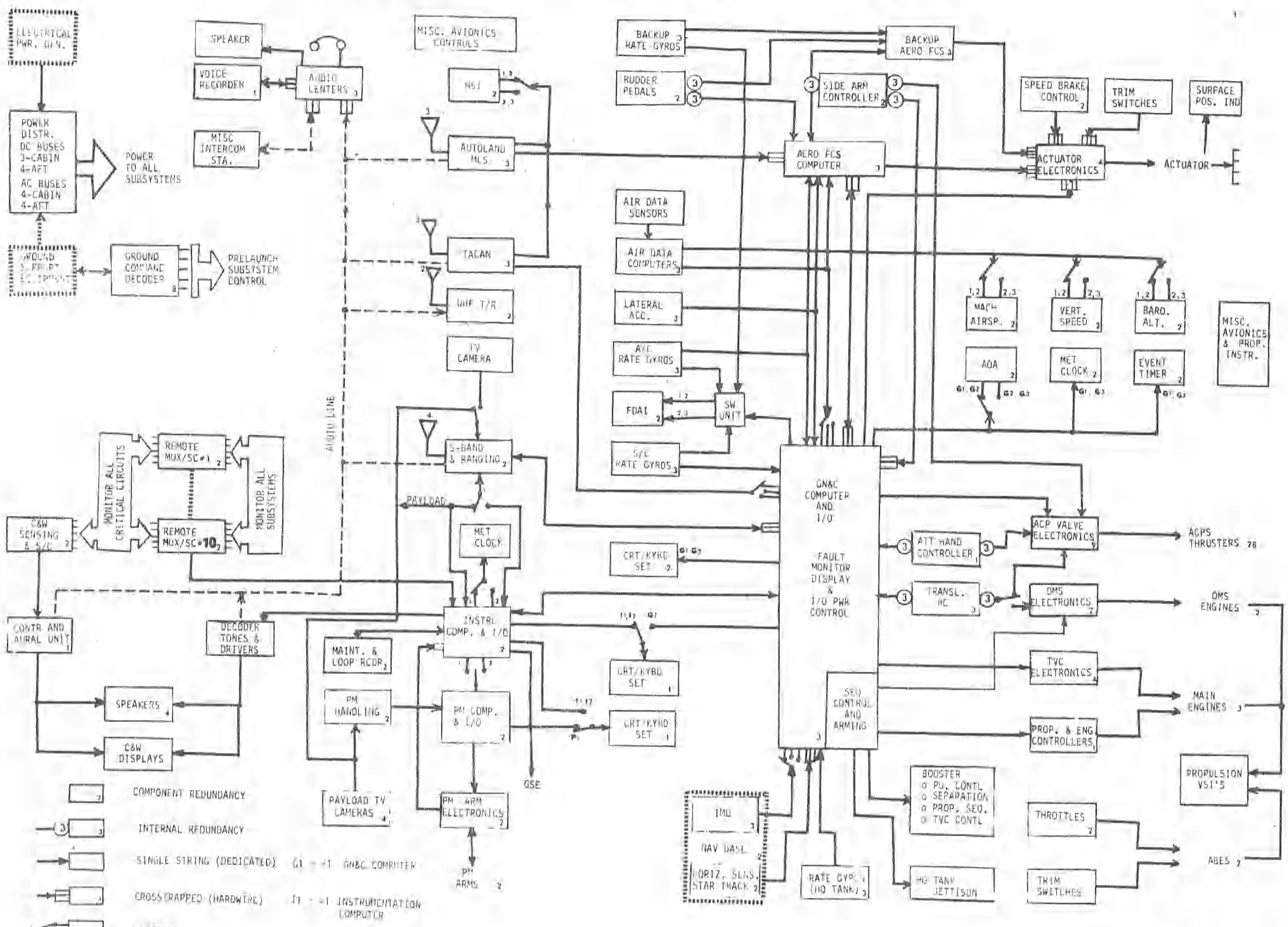


Figure 21

- Operational Instrumentation
- Displays and Controls, Caution, and Warning
- Payload Handling

The avionics system implements each of the functions with separate dedicated subsystems with minimal interfaces. Each subsystem design is multiply redundant, with hardwired functional strings, and management at the subsystem level.

3.7 **Crew Station** – The crew station includes the life support environment, crew/passenger furnishings and accommodations, and vehicle subsystem status monitoring and control required by the flight crews.

The Environmental Control and Life Support System (ECLSS) provides a shirtsleeve environment, water management, atmosphere revitalization, waste management, and equipment thermal control through launch, ascent, on-orbit, entry and landing for a flight crew of two pilots and two cargo handlers. The ECLSS design makes maximum use of aircraft philosophies and practices to provide convenient maintainability.

The crew subsystems include crew furnishings and equipment, controls and displays. The flight station provides for two pilots in a conventional side-by-side arrangement in the forward part of the flight deck. The general layout of the forward instrument panel, center console, side consoles and overhead panel is essentially the same as the DC-10. Side stick controllers are used in lieu of a center stick. The rudder pedals are retained for yaw control, braking, and nose wheel steering. Seat adjustments and floor spacing beneath the seats meet launch and orbital operations requirements.

3.8 **Power Systems** – Power systems for the Orbiter consist of the electrical and hydraulic subsystems.

The Electrical power subsystem consists of three 28 volt fuel cells and four AC generators. Two batteries provide power for restart of the fuel cell modules and the AC generators. Static inverters provide AC power during periods when the AC generators are not running; transformer-rectifier units provide DC power from the AC generators. Power is transmitted fore and aft in the vehicle over 115/200 volt, 3 ϕ, 400 Hz AC transmission lines to reduce the weight.

The Hydraulic system consists of four independent systems operating at 3000 psig. Each system is powered by a hydrazine auxiliary power unit which drives the system pump. All four hydraulic systems are distributed to flight critical subsystems to provide the vehicle with a highly reliable design.

3.9 **External Tanks** – The Orbiter mounting arrangement and tank geometry of the external HO tank for the recommended twin 156-inch SRM parallel burn configuration is illustrated in Figure 22. The external tank design for the series burn configuration is similar except that it contains separate bulkheads and holds about one third less propellant.

TANK GEOMETRY – PARALLEL BURN

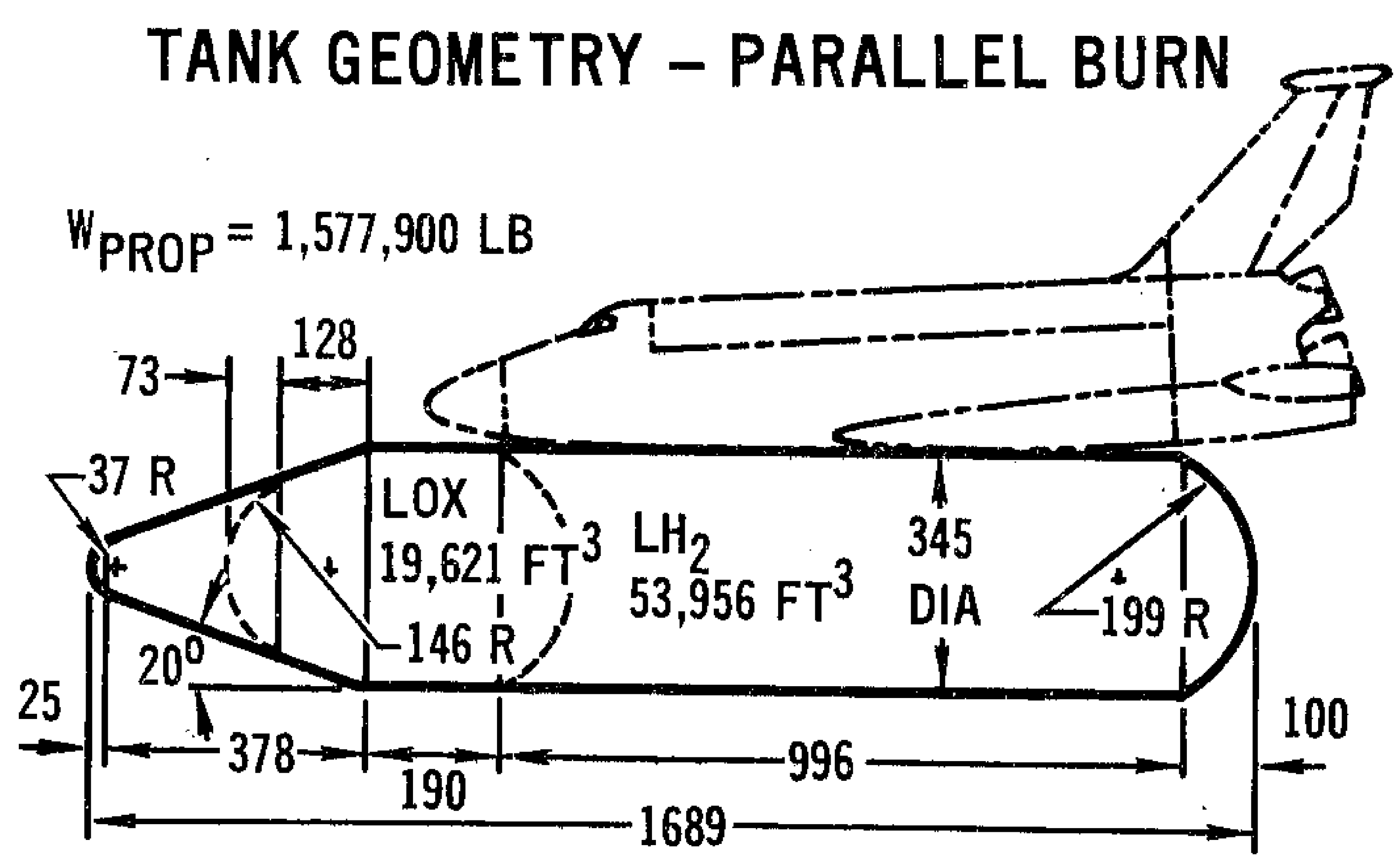


Figure 22

4 BOOSTERS

A wide spectrum of Booster concepts was explored in search of a balance between the dual objectives of low RDT&E cost and low cost per flight. Flyback Booster concepts were discarded because of excessive RDT&E costs. Subsequent effort emphasized water-recoverable liquid Boosters, using pressure-fed or pump-fed (F-1) engines, and 156-inch diameter segmented solid rockets.

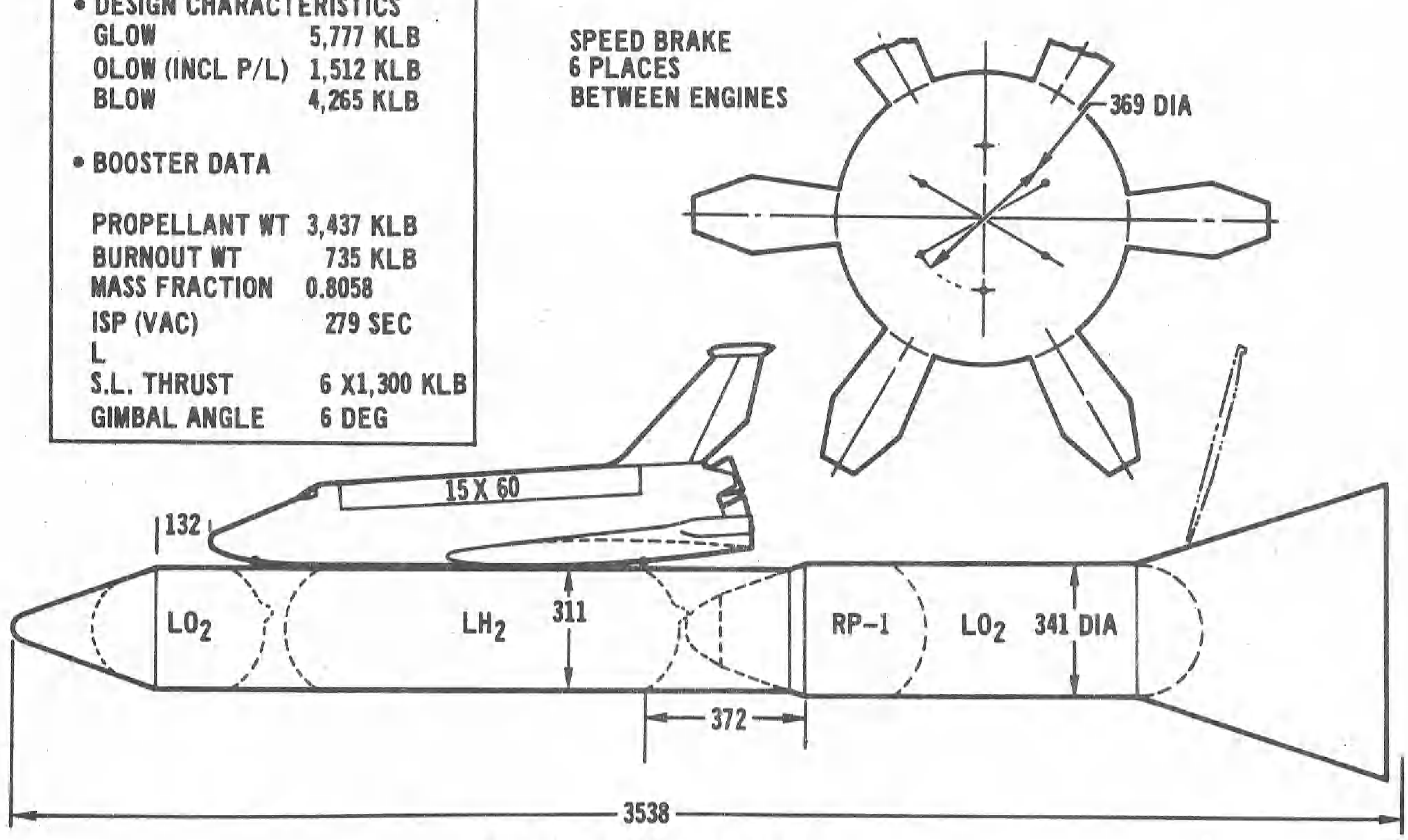
4.1 **Pressure-Fed Booster** – The final pressure-fed Booster system shown in the ascent configuration in Figure 23, is series burn, tandem, and staged at ~ 4000 fps relative velocity. Its size is comparable to Saturn V in GLOW, thrust and total length, as are the propellants, LO_2/RP-1 in the Booster and LO_2/LH_2 in the expendable Orbiter Tankage. The Booster is about 152 ft long, with tankage 341-inch dia, and the base flaring to 756-inch dia.

Additional detail is shown in the inboard profile, Figure 24. The entire maraging steel tank is reinforced with ring frames to resist water impact loads. The common bulkhead is integrally stiffened against full reverse pressure. The fuel tank is pressurized by catalytically decomposed hydrazine, also used to warm the helium gas which pressurizes the LO_2 tank.

The rugged 2219 aluminum base flare protects the engine system against seawater impact. Six

ASCENT CONFIGURATION
Series Burn Pressure-Fed (Resized)

• DESIGN CHARACTERISTICS	
GLOW	5,777 KLB
OLOW (INCL P/L)	1,512 KLB
BLOW	4,265 KLB
• BOOSTER DATA	
PROPELLANT WT	3,437 KLB
BURNOUT WT	735 KLB
MASS FRACTION	0.8058
ISP (VAC)	279 SEC
L	
S.L. THRUST	6 X1,300 KLB
GIMBAL ANGLE	6 DEG



DIMENSIONS GIVEN IN INCHES

Figure 23

INBOARD PROFILE
Series Burn Pressure – Fed Booster

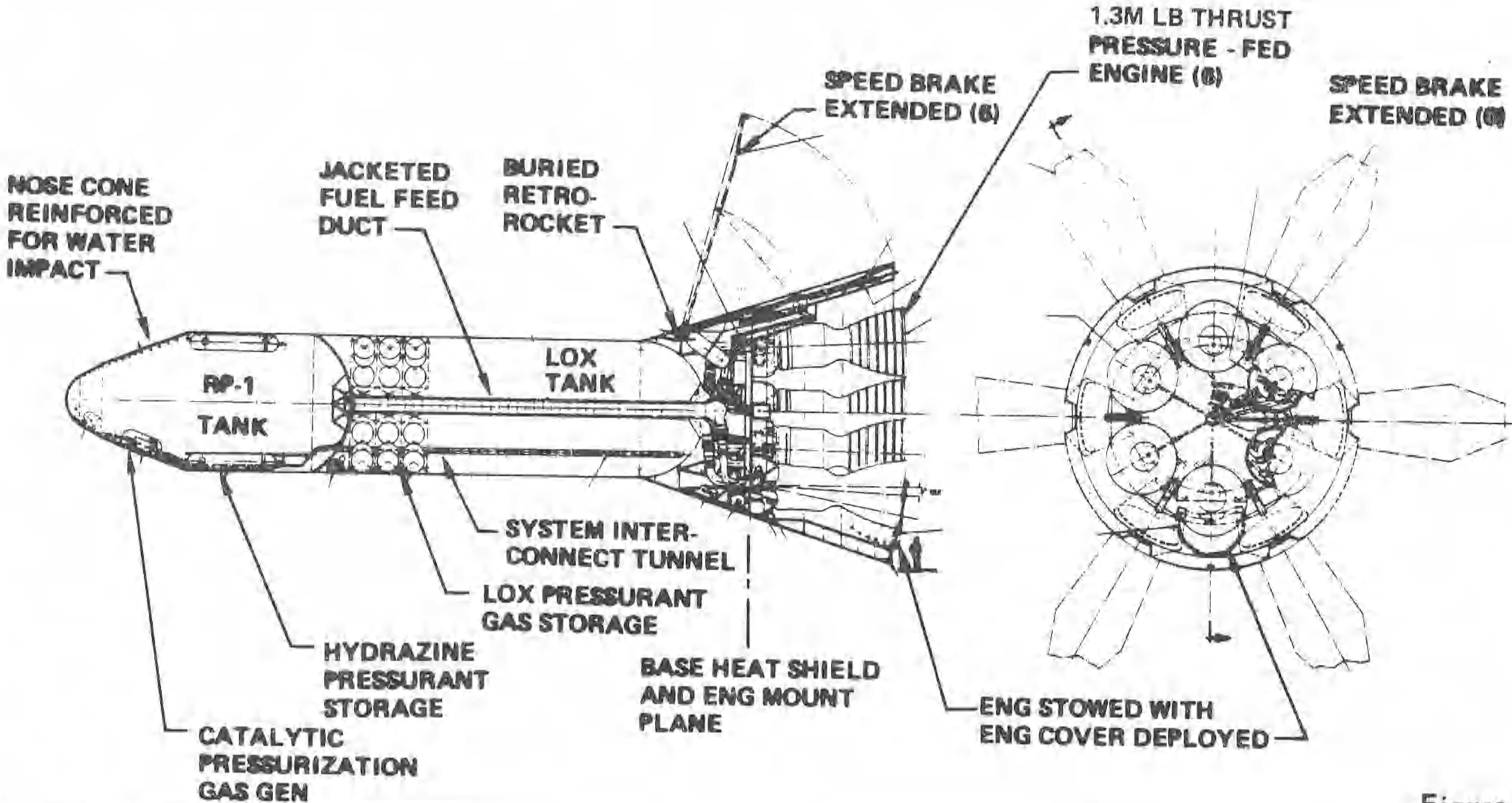


Figure 24

braking flaps, flush mounted on the outside of the flare, are opened after burnout to provide drag and stability during descent. Six retrorockets are fired during the last few seconds of descent, reducing the water entry velocity to 100 fps. The braking flaps are retracted and the engine nozzle covers deployed prior to water entry.

This pressure-fed Booster evolved via numerous trade studies. Figure 25 compares a liftoff thrust/weight (T/W) ratio of 1.35 with the 1.25 originally baselined. Lower inflight losses resulting from T/W = 1.35 produce net size reductions worth about $20 million in program cost and simultaneously double the growth accommodation margin. Booster engine-out abort margin is also increased.

Three classes of pressure-fed engines were initially advocated, as shown in Figure 26. Ablatively-cooled engine featuring ruggedness, good performance and minimum technical risk were disqualified because of excessive refurbishment costs. The duct-cooled engine, which provides film cooling penalty because of the poorer specific impulse of the coolant fuel. The high-performance regeneratively-cooled engine is more complex to develop and fabricate, but represents the greatest backlog of relevant experience. Performance trade study results dictated selection of a LO_2/RP-1 regenerative engine.

Thrust Vector Control (TVC) trade studies investigated liquid secondary injection, high performance hot-gas main-chamber-bleed and gimbaled main engines. Three candidates are illustrated in Figure 27. Although the water-recovery mode made protection of conventional gimbaled engines difficult, it was the selected concept. Figure 28 indicates a performance penalty of about 7% in Booster size for either secondary injection approach over an adequately-protected gimbaled engine.

Several other subsystem trade analyses were performed. Of the pressurization system concepts considered the baseline was preferred to the next best candidate using autogenous GOX to pressurize the LO_2. Tank material selection was found to be a critical issue. Of three conditionally satisfactory candidate LO_2 tank materials, maraging steel was baselined, though it requires process development for cryogenic fracture toughness. Thick, high-pressure 2219 aluminum tank-

IMPACT OF $(T/W)_1$ ON GROWTH AND COST

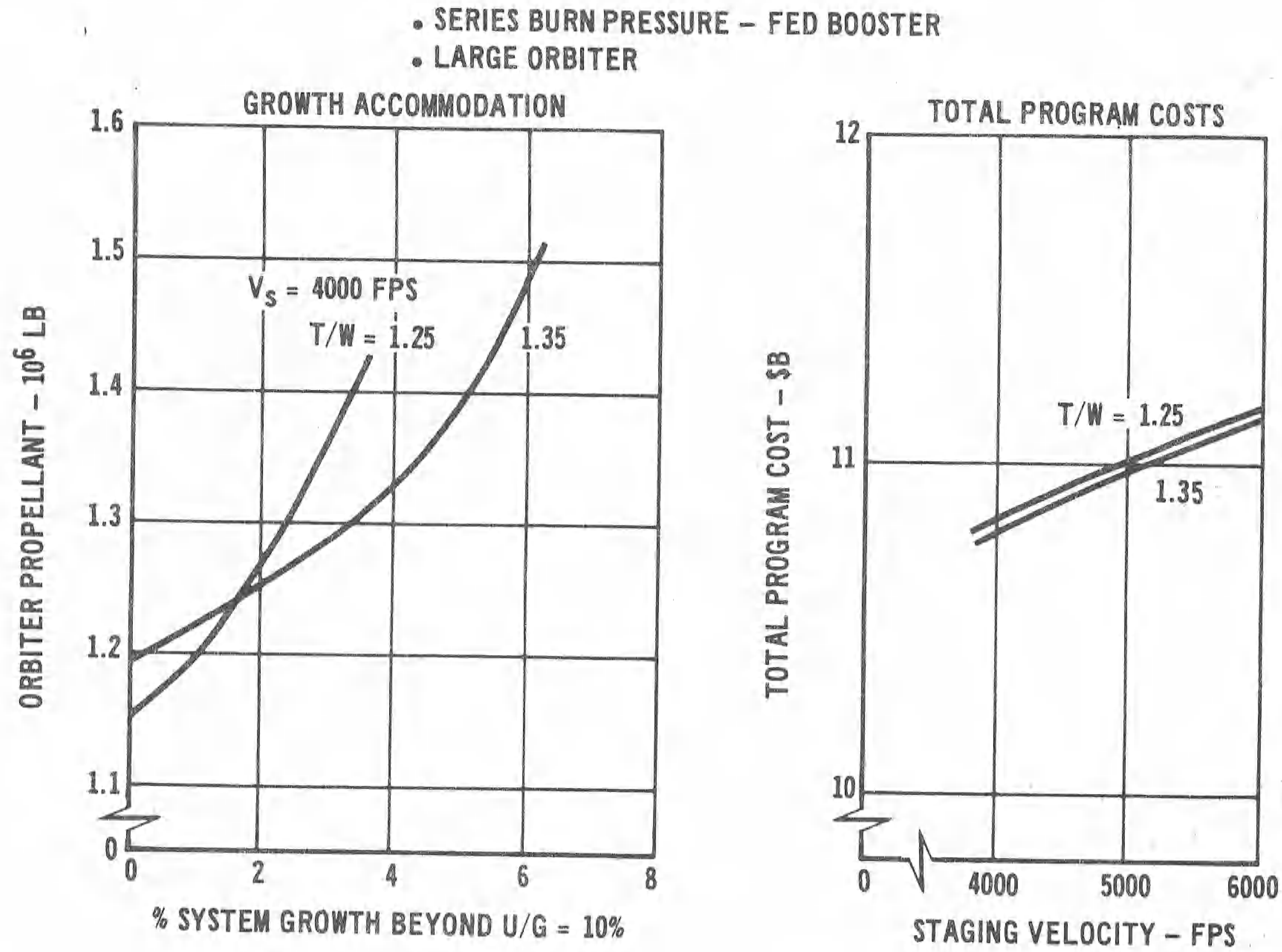


Figure 25

PRESSURE-FED ENGINE CONCEPTS

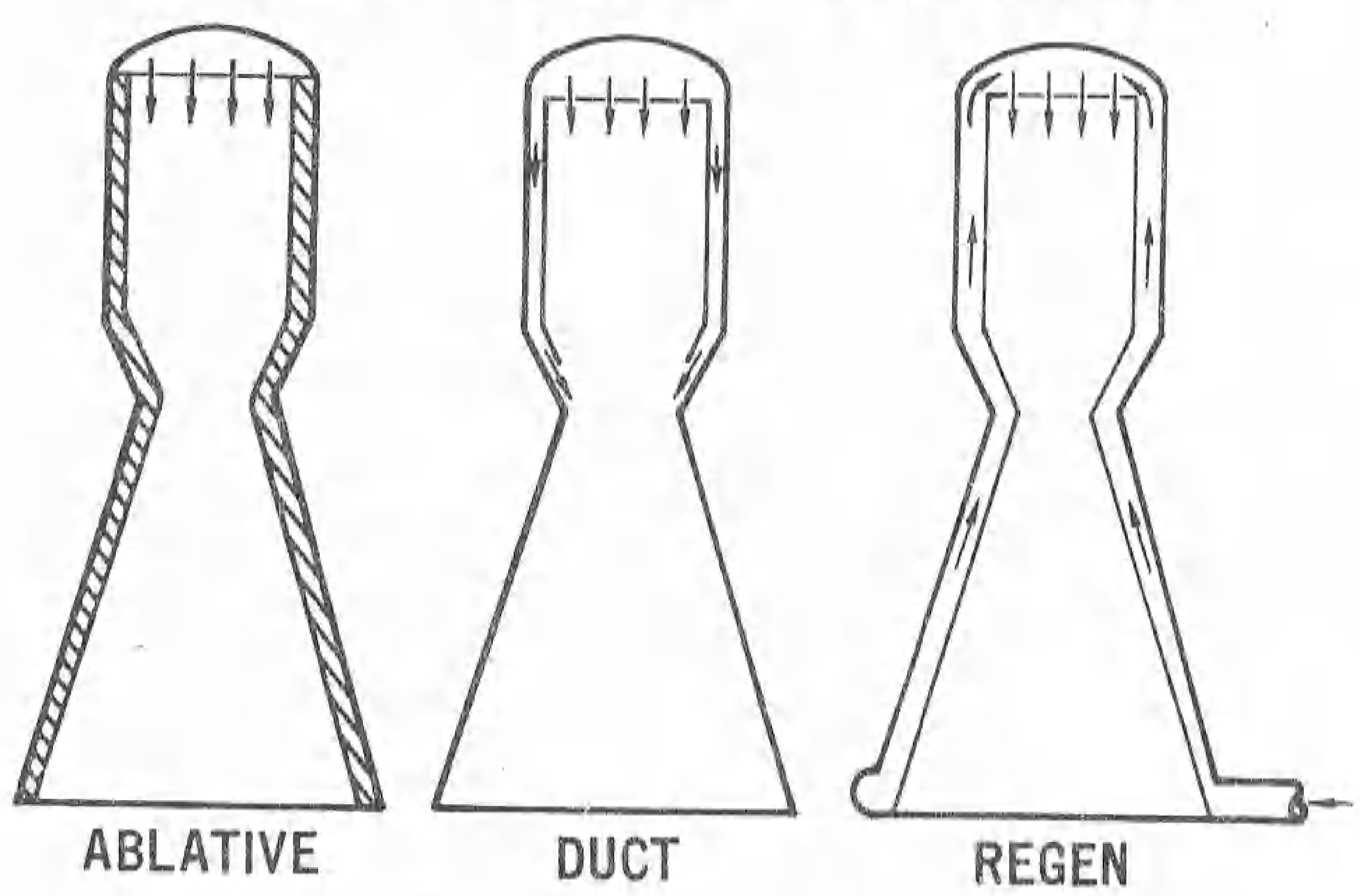


Figure 26

age was discarded because it would require thick weld bands and excessive welding time. Bolted joints are a promising alternative to welding thick plates. Inconel 718 is strong and corrosion resistant, but machining costs appear prohibitive.

Recovery System alternatives illustrated in Figure 29, considered high-α and low-α entry concepts. High-α entry is sensitive to configuration details and requires a continuously active ACPS till parachute deployment. The baselined mode is low-α entry which is a low-risk, relatively passive approach. Retrorockets, in lieu of parachutes, were selected for terminal deceleration for reasons of weight and cost as well as for minimum technical risk. This comparison is shown in Figure 30.

4.2 Series Burn Pump-Fed Booster – The primary pump-fed Booster candidate is one incorporating F-1 engines and significant portions of Saturn S-IC tankage and plumbing. This series burn system is illustrated in Figure 31. The inboard profile, Figure 32, shows the LO_2 tank forward, and multiple individual feed lines through the fuel tank to simulate Saturn S-IC pump inlet conditions and minimize modifications to the F-1 engines. The heat shield is a watertight bulkhead

CANDIDATE TVC CONFIGURATIONS
Series Burn Pressure-Fed Booster

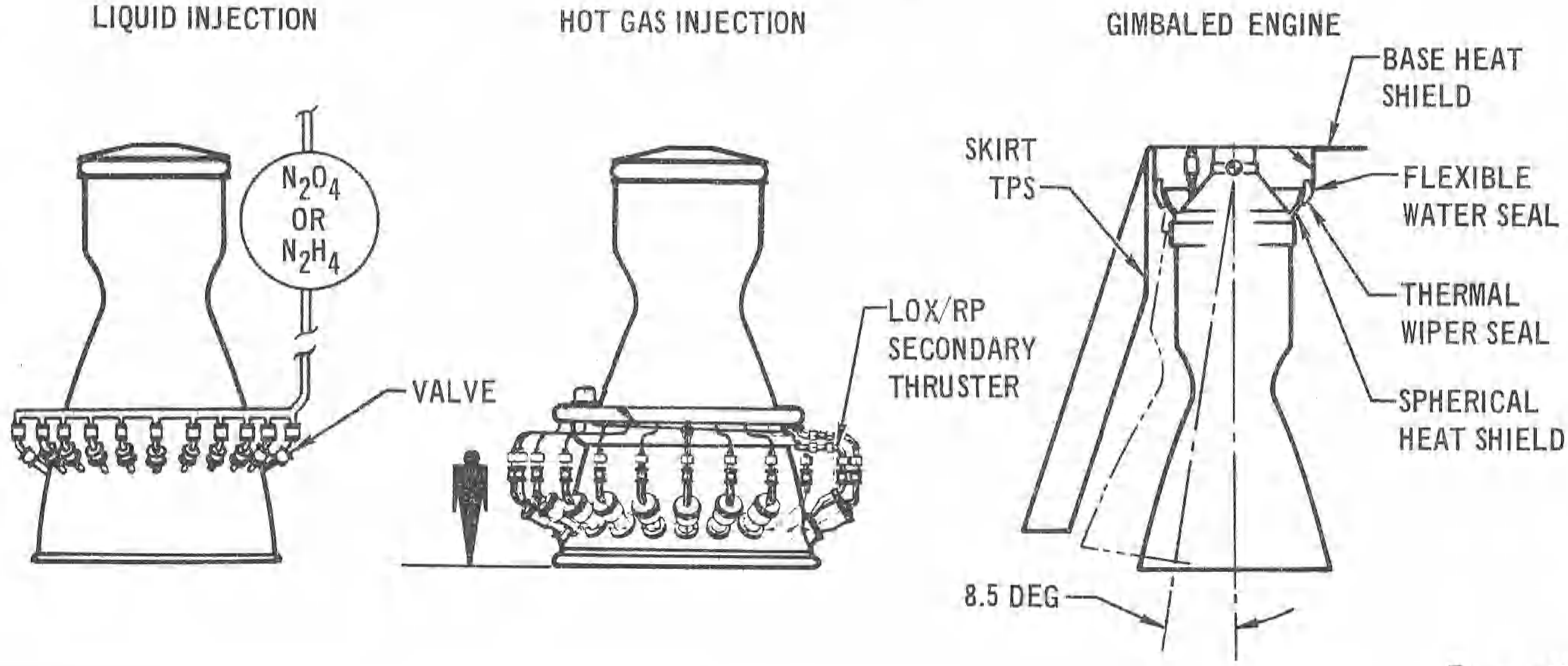


Figure 27

PERFORMANCE OF CANDIDATE TVC SYSTEMS
Series Burn Pressure-Fed Booster

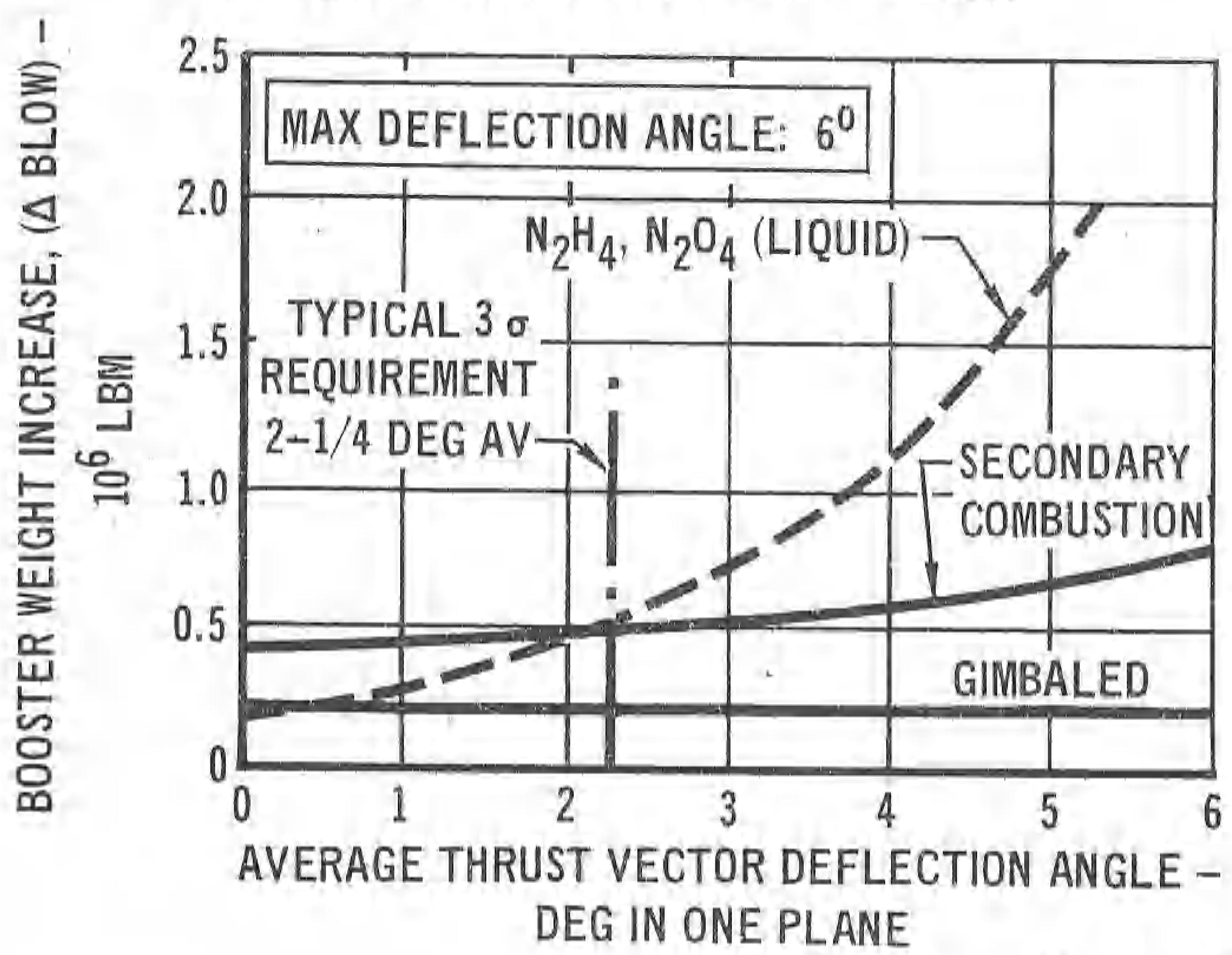


Figure 28

with sliding spherical seals that permit $\pm 6^0$ gimbaling. Retrorockets for this vehicle are sized for a maximum water-impact velocity of 35 fps.

Among the trade analyses leading to this design is thrust shaping during ascent. With no thrust modulation, an optimum trajectory would experience a maximum dynamic pressure (q) of 905 psf as shown in Figure 33, whereas lofting to reduce q max below 800 psf severely penalizes performance. Shutting down on F-1 early to comply with the 650 psf q max constraint costs 12K lb in payload. 13% step throttling, which requires no new engine development also complies with this q constraint with half the penalty and is the recommended approach. Figure 34 summarizes a comparison of the pressure-fed and F-1 versions of a fully-reusable LO_2/RP-1 water-recoverable shuttle Booster.

4.3 Parallel Burn Solid Booster (2-156-inch SRM) – The baseline parallel burn, SRM ascent configuration (Figure 2) consists of the 068B Orbiter, external HO tank, and two 156-inch SRM's Staging velocity is 4000 ft/sec, at ~ 130 sec. SRM data are summarized in Figure 35.

Additions to the motor that are necessary to adapt it for this application are: nose fairing and tunnel, necessary avionics, thrust termination device, SRM attachments (for mating to the Orbiter propellant tank), forward and aft skirts, and launch support structure. The avionics needed are minimal since all commands are provided by the Orbiter and the only sequencing signals needed are for SRM ignition, thrust termination, and separation. SRM thrust termination is provided by two ports which are located in the forward dome of each SRM. Attachments for mating the

RECOVERY SYSTEM ALTERNATIVES

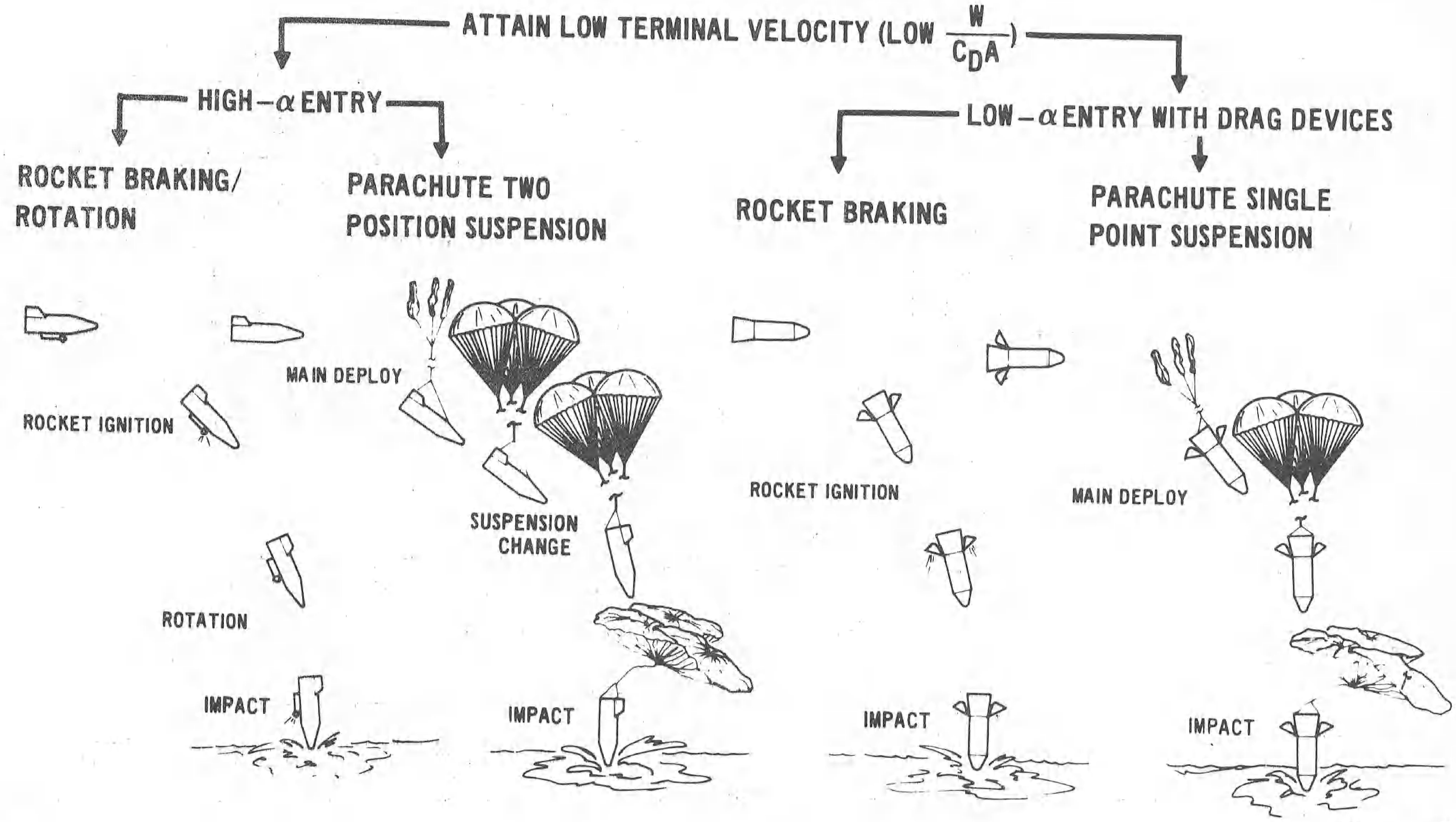


Figure 29

ROCKET vs PARACHUTES

SERIES BURN PRESSURE - FED
RECOVERY WEIGHT = 748,800 LB

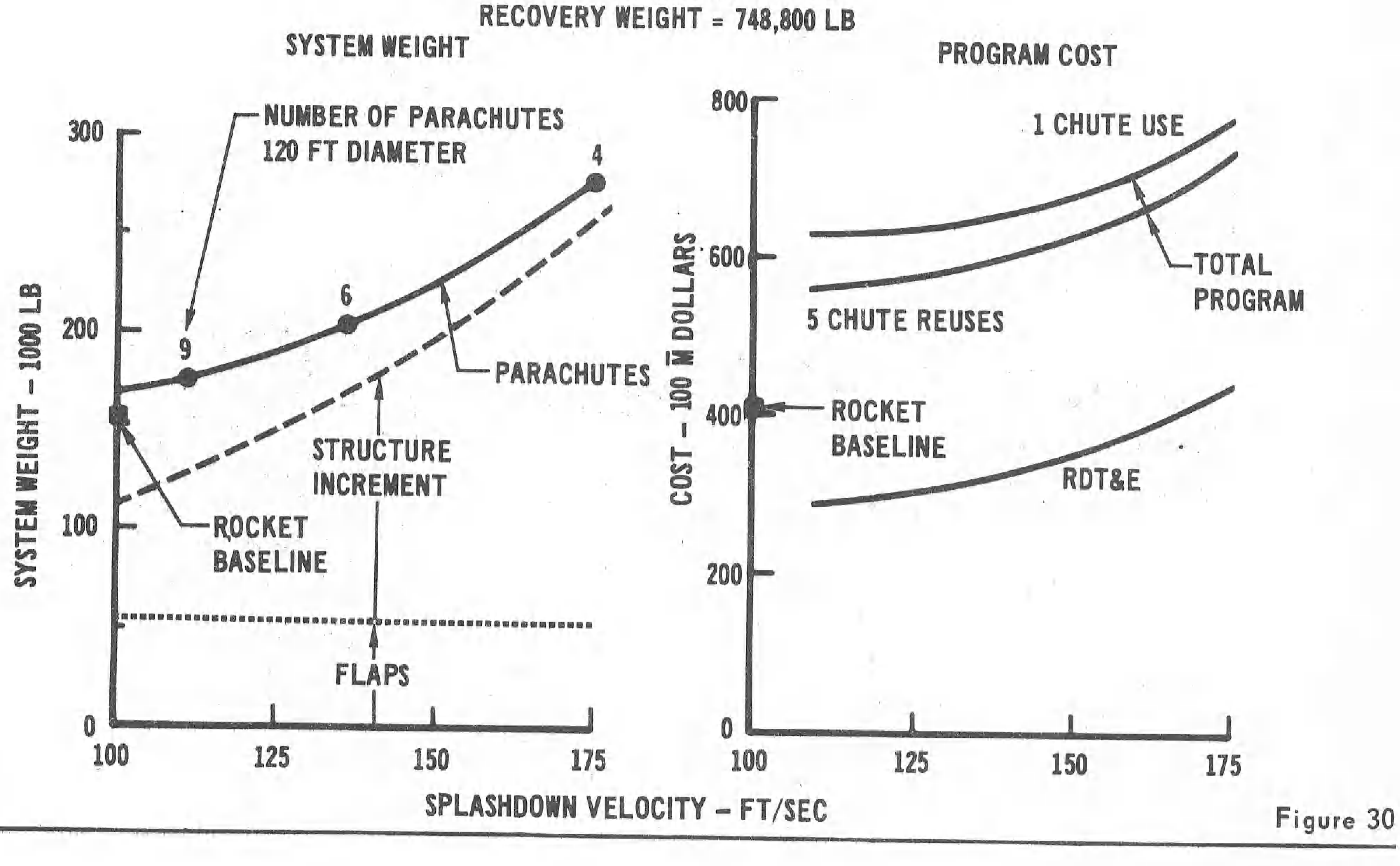


Figure 30

SERIES BURN F-1 BOOSTER

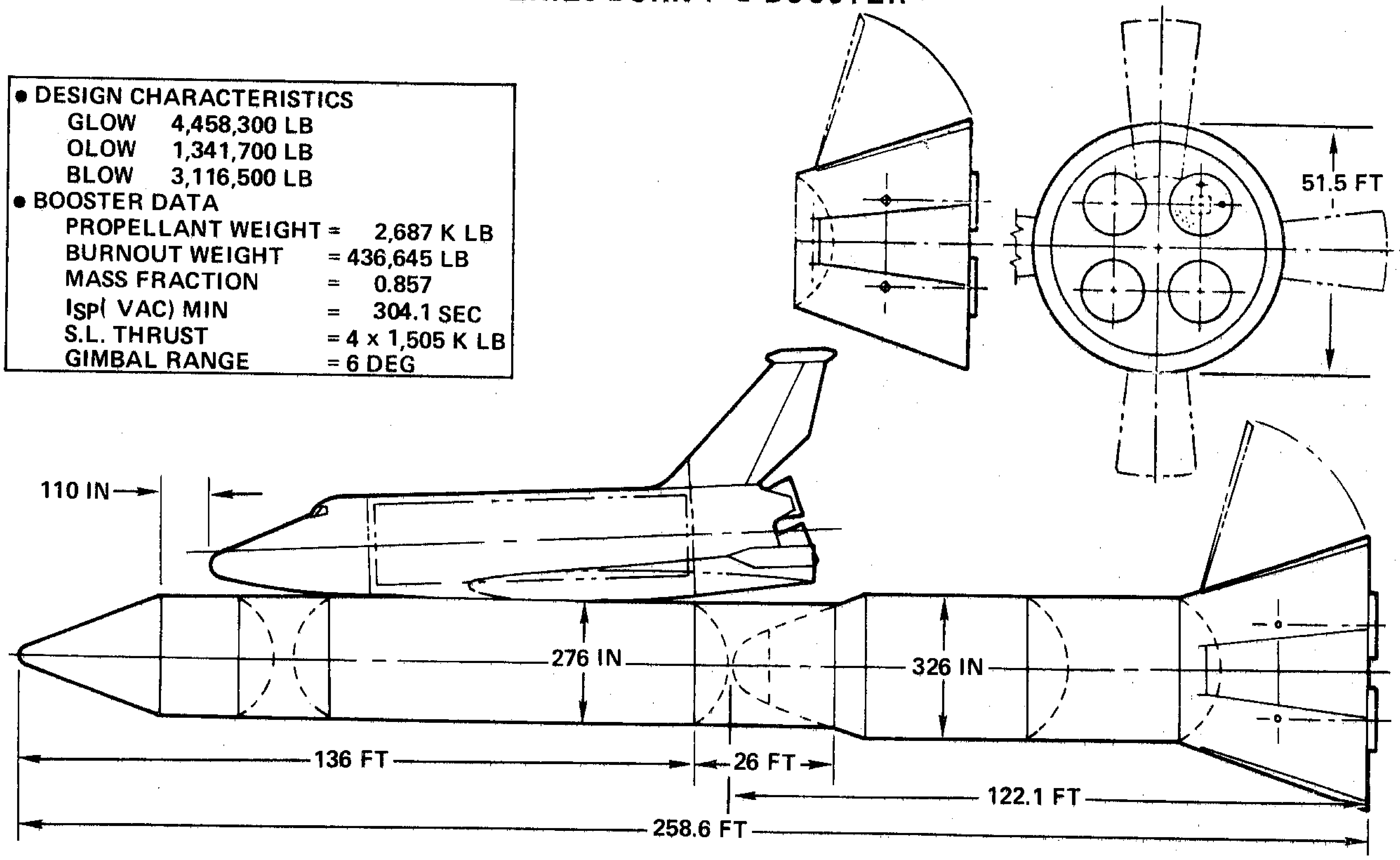


Figure 31

INBOARD PROFILE
Series Burn Pump-Fed (F-1) Booster

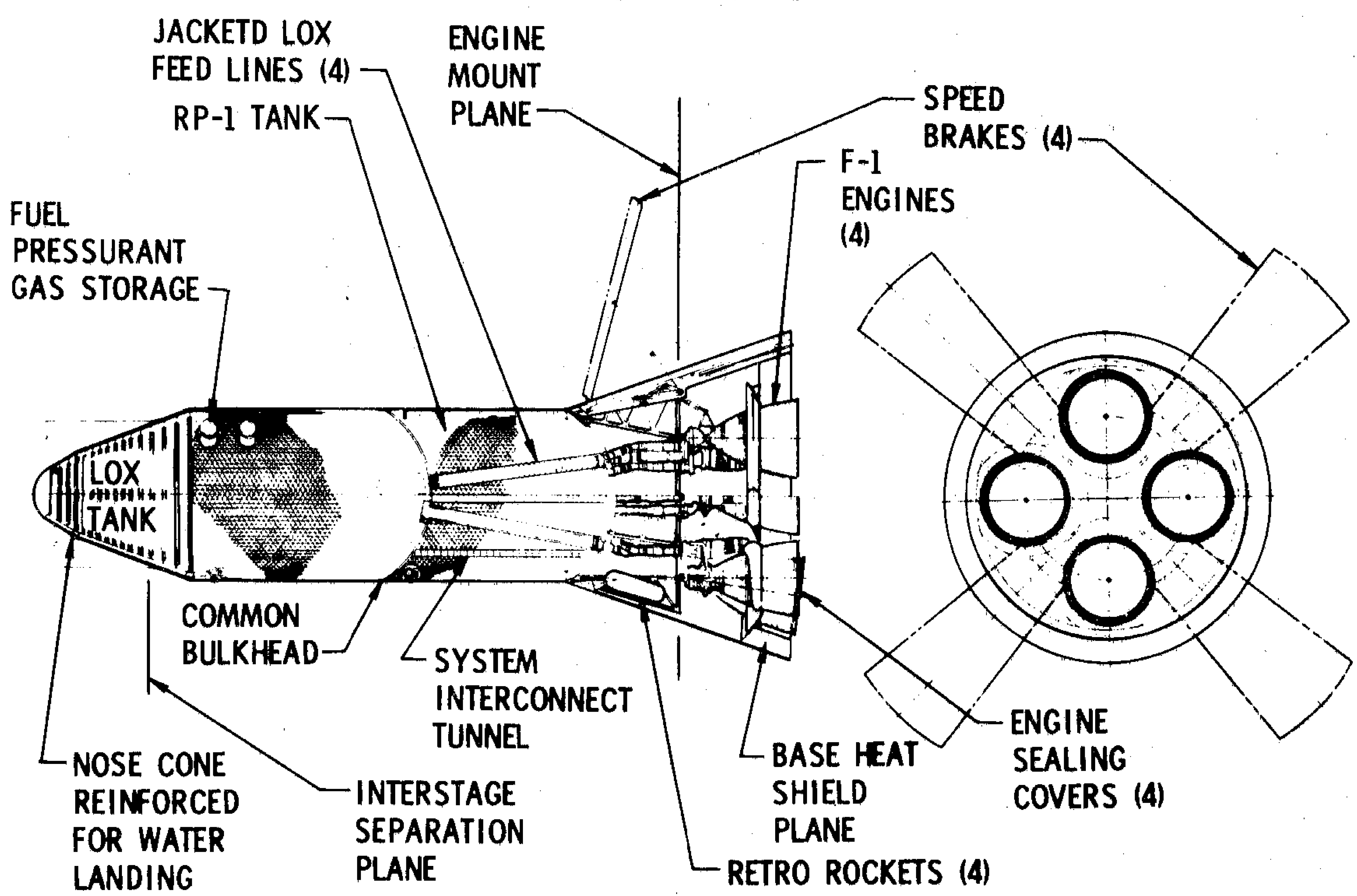


Figure 32

F–1 SERIES BURN (TEL) DYNAMIC PRESSURE STUDY

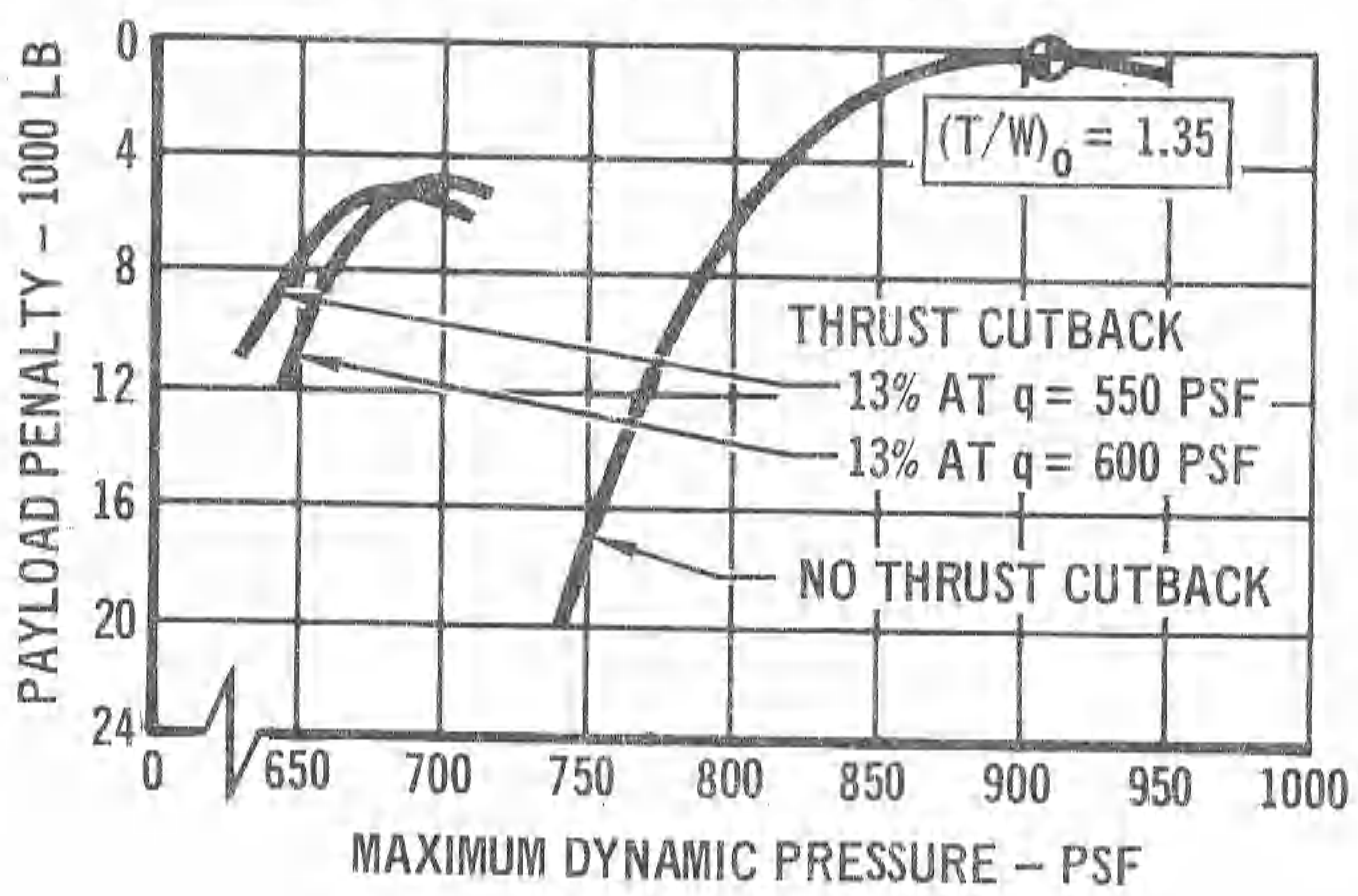


Figure 33

SRM to the tank are located in the forward and aft skirts. The aft skirt also supports the total shuttle system on the launch pad.

No SRM recovery is provided for in the baseline system; however, recovery and reuse of the SRM's has been recommended to further reduce total program and pre-flight costs.

During mated Booster operation, the SRM nozzles are fixed and control of the vehicle is achieved by gimbaling Orbiter engines and deflecting Orbiter aerodynamic surfaces. The Booster motor nozzles are canted outboard 15 degrees to reduce moments caused by any thrust imbalance between the two motors.

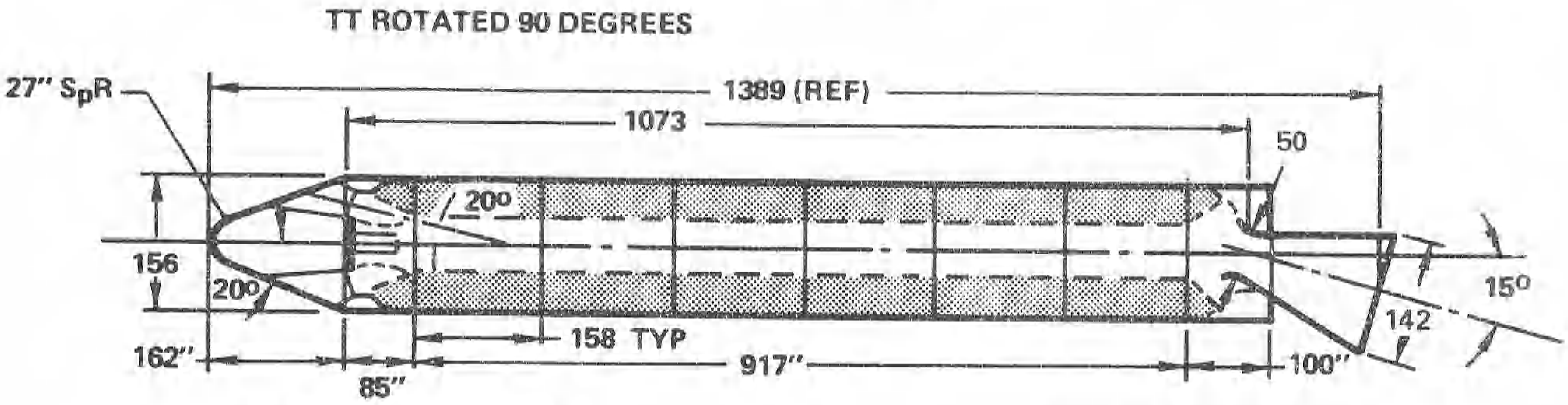
Analyses of ascent controllability without Booster TVC was a key element of this study. The resultant Orbiter pitch gimbal angle requirements, as

PUMP vs PRESSURE-FED COMPARISON
Series Burn

CRITERIA	PRESSURE FED	F–1
WEIGHT (K LB) GLOW DRY	 5778 735	 4458 372 ✓
ENGINE DEVELOPMENT	NEW DEVELOPMENT – POTENTIAL STABILITY PROBLEMS	MINOR CHANGES ✓
EASE OF SEALING ENGINES	6 ENGINES	4 ENGINES SIMPLER ✓
BEEF–UP FOR WATER IMPACT	MODEST DUE TO RUGGED PRESSURE FED SHELL ✓	MORE FRAGILE REQUIRING MORE SOPHISTICATED RECOVERY SYS OR MORE STRUCT–
ENGINE OUT PERFORMANCE	ADEQUATE ✓	MARGINAL FOR T/W & CONTROLLABILITY
PRESSURIZATION SYSTEM	SIMPLE He PLUS HYDRAZINE DECOMPOSITION. MUCH LARGER REQUIREMENTS	SIMPLE He PLUS AUTOGENOUS OXYGEN ✓
MATERIALS	PREFERRED 2219 REQUIRES 4 TO 5 INCH WELDS RESULTING IN SUBSTANTIAL DEVELOPMENT & PRODUCTION PROBLEMS. BOLTED JOINTS LOOK PROMISING.	STRAIGHT FORWARD 2219 ✓
MAX Q CONTROL	CUT ONE ENGINE OR BLEED PRESSURE ✓	USE 2 POSITION ORIFICING TO THROTTLE ✓
ENTRY HEAT PROTECTION	SHELL HEAT SINK ADEQUATE ✓	MODEST WEIGHT ADDITION FOR HEAT SINK
RECOVERY SYSTEM	FLAPS & ROCKETS ΔWT = 155,000 LB	FLAPS AND ROCKETS ΔWT = 66,000 LB ✓
COSTS ($M) TOTAL PROGRAM RDT&E	 10,798 5,678	 10,002 5,018 ✓
DEPTH OF ANALYSIS	SUBSTANTIAL ✓	MODERATE

Figure 34

BOOSTER CHARACTERISTICS
Parallel 156 In. SRM Configuration



SRM CHARACTERISTICS	
PROP WT	1,058,700 LB
THRUST AT SL	2,575,000 LB
SPECIFIC IMPULSE, VAC	264 SEC (MIN)
NOZZLE EXPANSION RATIO	8
WEB/ACTION TIME	120/130 SEC
THRUST REGRESSIVITY	40%
MEOP	1,000 PSIA
TVC	NONE
CASE AND DOME MATERIAL	D6AC STEEL
YIELD STRENGTH	195,000 PSI
CASE YIELD/MEOP RATIO	1.25
PORT AREA/THROAT AREA	1.3

SRM STAGE WEIGHTS	
SOLID ROCKET MOTOR	118,470
AFT SKIRT AND LAUNCH STRUCTURE	8,630
TANK ATTACH AND SEPARATION	11,850
NOSE FAIRING & TUNNEL	900
THRUST TERMINATION	2,110
AVIONICS	310
GROWTH/UNC	7,280
TOTAL INERT WT	149,550
PROPELLANT	1,058,700
TOTAL WT	1,208,250
EXPENDED PROP AND INSUL	1,064,500
BURNOUT WT	143,756
$\lambda' = \frac{\text{EXPENDED MASS}}{\text{BLOW}}$	0.881

Figure 35

shown in Figure 36, include a one-degree allowance for attitude maneuvering as well as thrust and cg effects and the gominant influence of winds. Since the parallel burn vehicle is stable, the tailwind attitude-loop saturation of q_{max} will merely impart minor trajectory dispersions on extreme-wind missions. Though yaw-roll data suggested the desirability of adding a ventral fin of about 200 sq ft, a roll-rate control mode permitting free roll at high q was selected.

Though substantial data and analysis support the feasibility of control without SRM a contingency SRM TVC design was studies. SRM TVC alternates are summarized in Figure 37. Risk and cost considerations dictate selection of the flexible-joint fully gimbaled nozzle should SRM TVC ever prove to be desirable.

Environmental issues reduce to two concerns: noise and atmosphere pollution. Fairfield noise analyses indicate that the twin 156-inch SRM shuttle sound pressure level will be about 10 db less than that of Saturn V. Noise level should thus be acceptable. Worst-case wind-distributed concentrations of the main SRM combustion products are shown in Figure 38. Concentrations of carbon monoxide and alumina are negligible. The HCl concentrations appear acceptable except in the event of launches in rain or mist. Direct precipitation of HCl could be a crop hazard and might impose launch humidity constraints.

Figure 39 illustrates a high-confidence SRM development schedule. It allows about 18 months from shuttle ATP till SRM ATP, which not only permits funding deferral, but also permits high-confidence requirement definition.

MAXIMUM PITCH THRUST VECTOR DEFECTION
Parallel Burn Solid (2–156 In.)

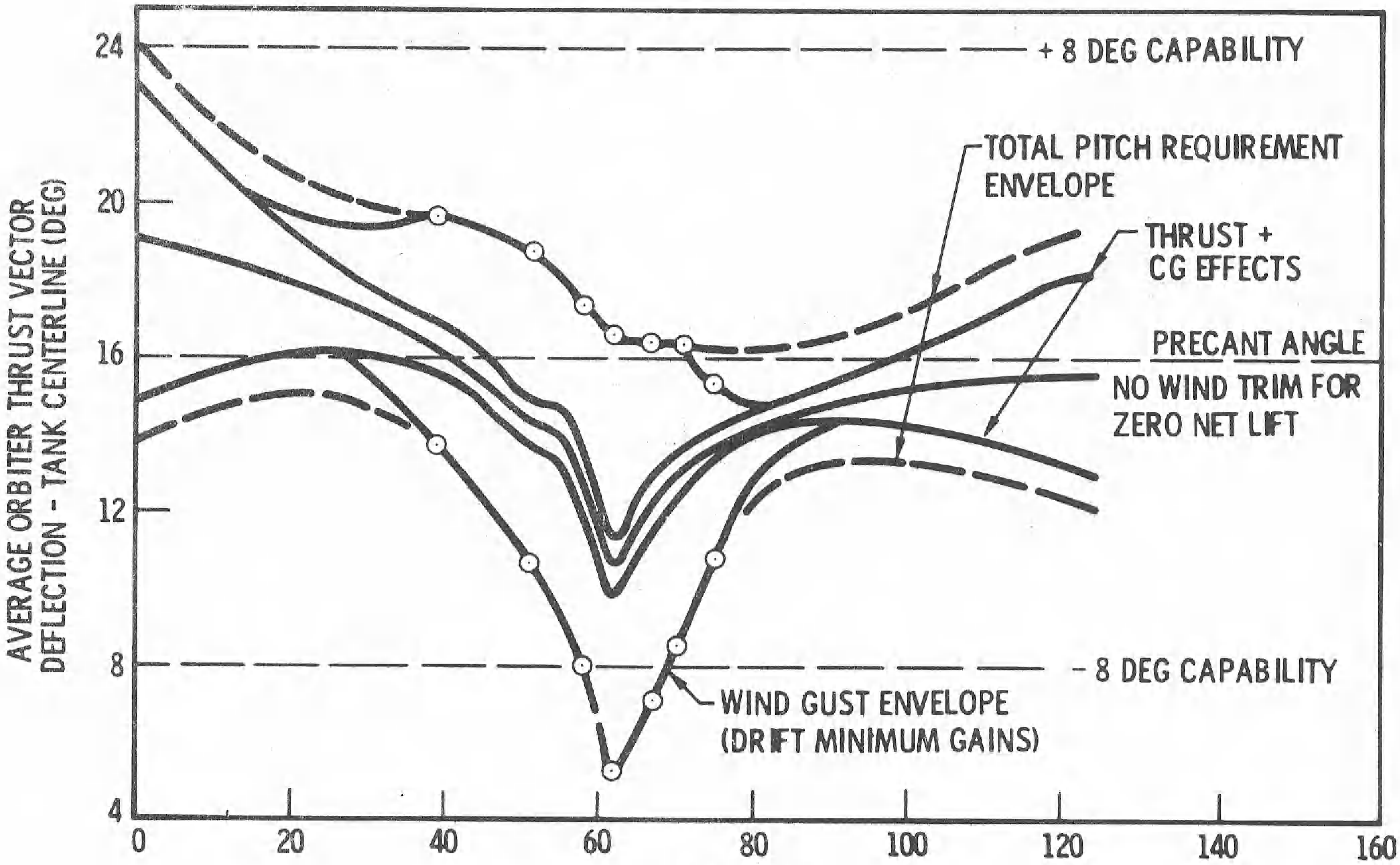
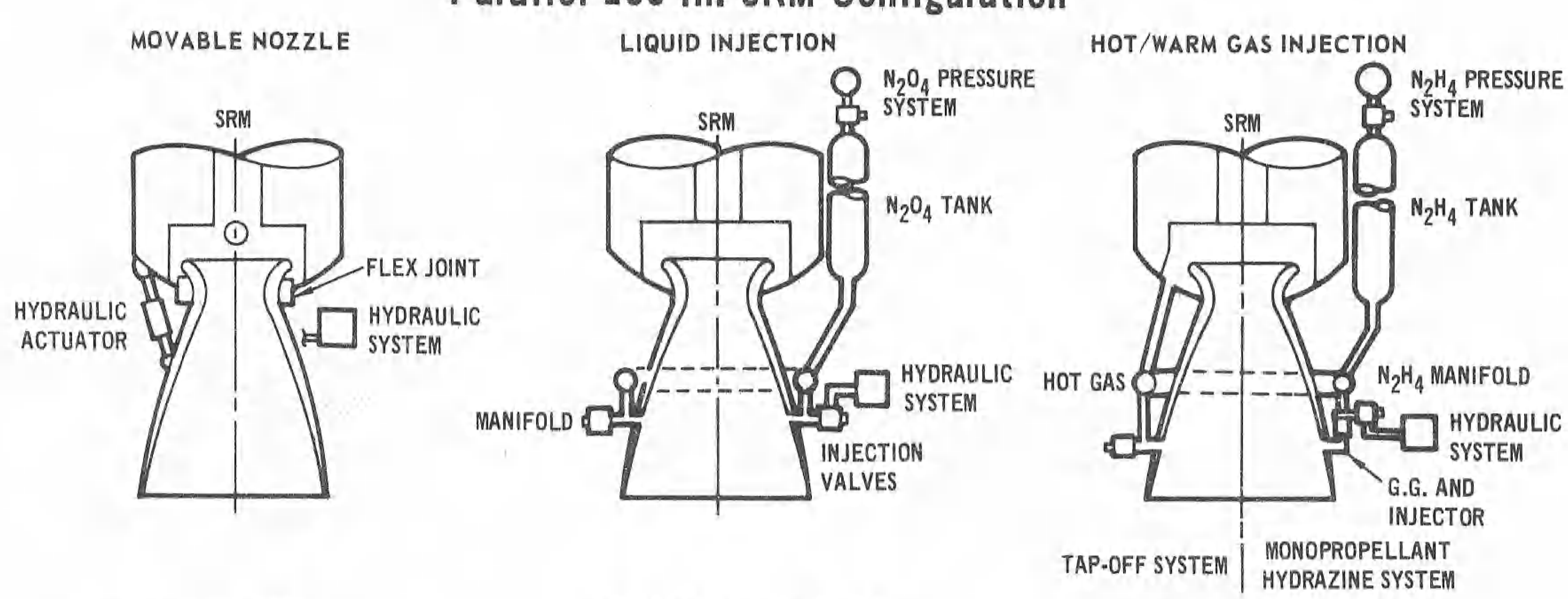


Figure 36

TVC CANDIDATE SYSTEMS
Parallel 156 In. SRM Configuration



Movable Nozzle
- STATE-OF-THE ART
- LARGE FLEXIBLE JOINT NOZZLES FIRED
- HYDRAULIC/APU EQUIPMENT DESIGNS NEEDED EXIST
- MINIMUM COMPLEXITY AND LOSS

Liquid Injection
- STATE-OF-THE-ART, SIMILAR TO UTC 1205 SYSTEM
- ADDITIONAL PROPELLANT IMPACTS
- HYDRAULIC ACTUATED VALVES
- HIGH COST

Hot/Warm Gas Injection
- HIGH RISK APPROACH
- COMPLEX
- LARGE PERFORMANCE PENALTY
- HIGH COST

Figure 37

PEAK CONCENTRATION (HC1, CO, and Al_2O_3) Downwind, Pad Abort

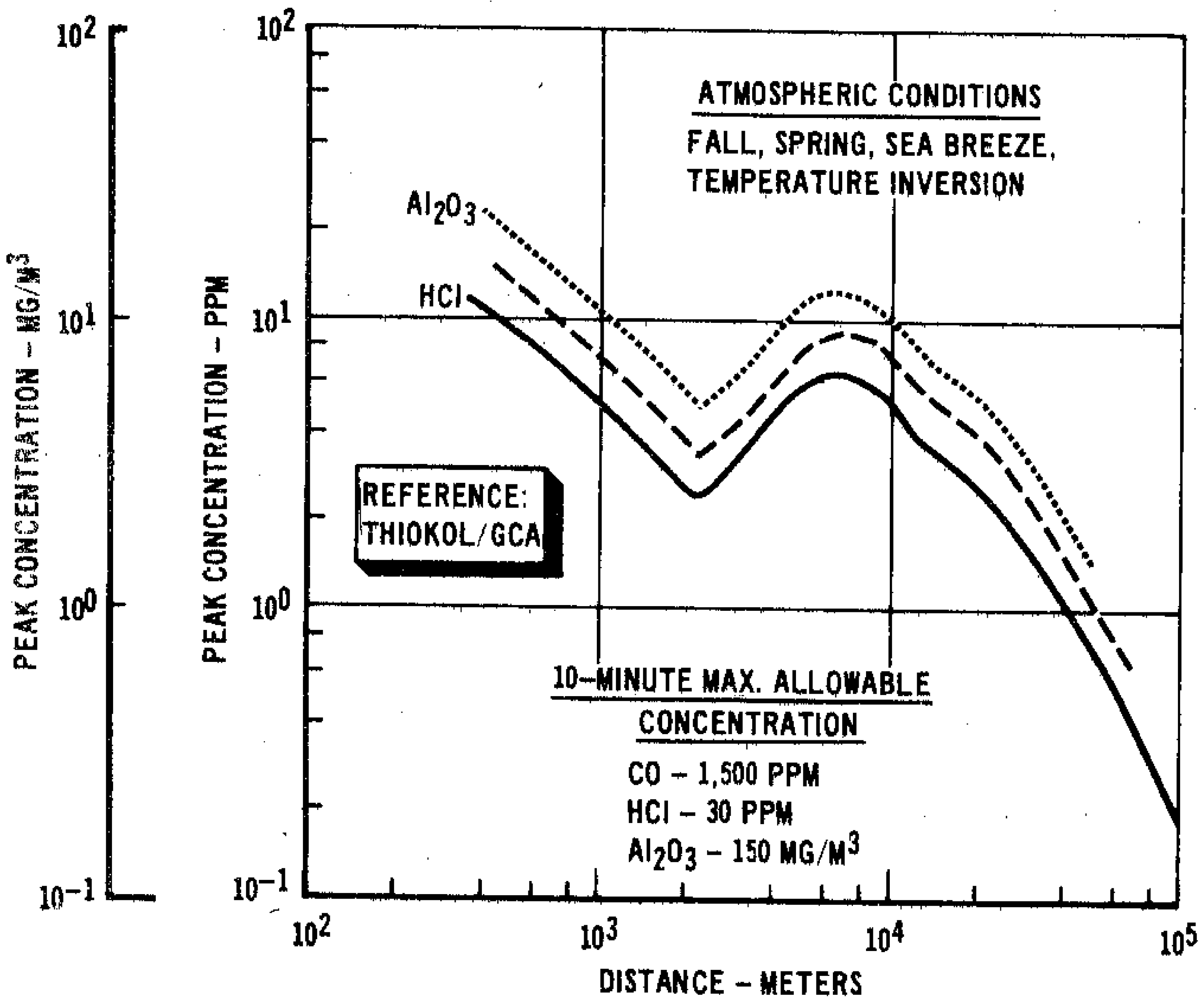


Figure 38

DEVELOPMENT SCHEDULE Parallel 156 In. SRM Configuration

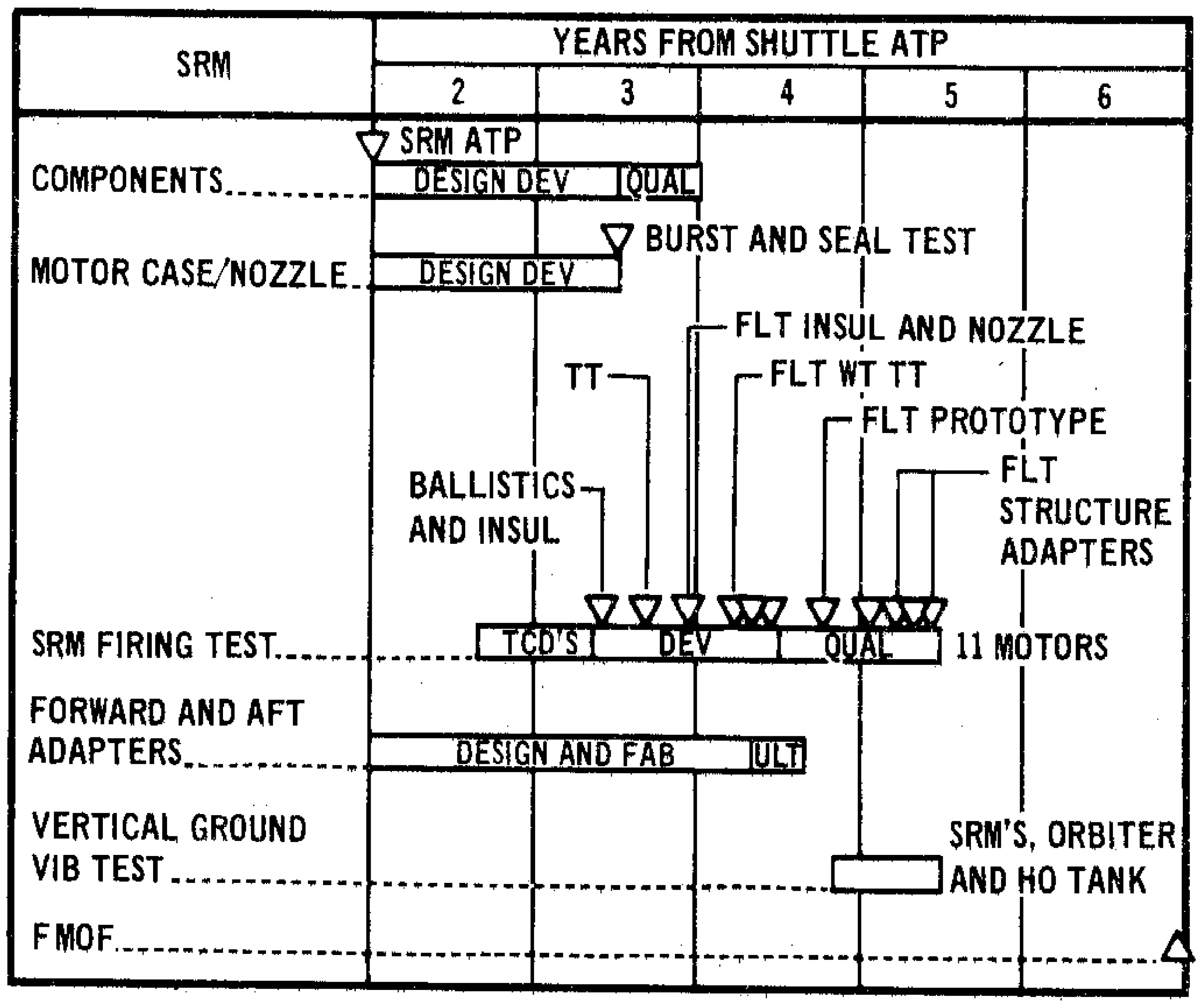


Figure 39

5 SYSTEM EVOLUTION

Two four-month extensions to the Phase B were conducted to define Shuttle configurations which would reduce the RDT&E costs and peak annual funding requirements defined during Phase B. Some of the major configurational variations investigated were Mark I (limited technology) Orbiters that could grow to Mark II (advanced technology) vehicles, parallel versus series burn sequence with various Booster concepts, Orbiter engine size and numbers, and payload bay size.

5.1 **Requirements Evolution** – During the two study extensions many of the original program constraints were varied in order to develop more economical approaches and concepts. The major variations in the system requirements are outlined in Figure 40. Only Orbiter crossrange and use of the airbreathing engine concept remained essentially unchanged throughout the extensions.

5.2 **Mark I – Mark II Approach** – The possibility of phasing system requirements to reduce

SHUTTLE REQUIREMENTS EVOLUTION

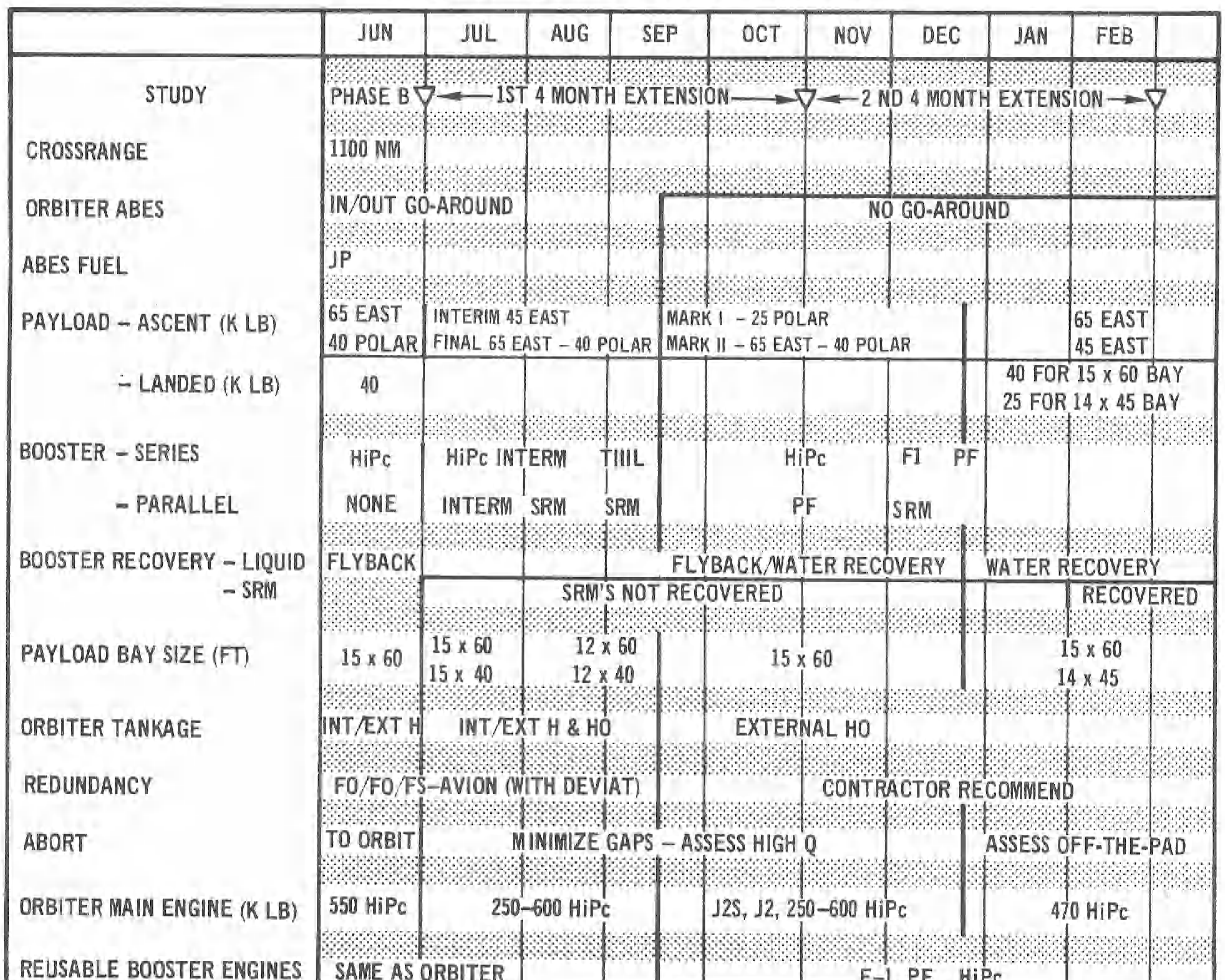


Figure 40

costs was investigated, utilizing reduced performance capability for the first 125 flights of the 445 flight program. This involved interim use of reduced payload requirements, an existing Orbiter engine (J2/J2-S), ablative TPS, simpler avionics, and an expendable Booster. Figures 41 and 42 compare the evolution time frame and risk of the two development approaches. Total program cost was found to be considerably lower if the Mark I phase was eliminated, primarily because Mark II required a single engine development, lower operational costs, and no retrofit costs.

5.3 Parallel versus Series Burn and Booster Selection – During the study, selection of the Booster and mated vehicle configuration was emphasized with particular attention to costs and technical risk. Evaluation of technical risk indicates parallel burn SRM's have the least risk, series burn HiPc Boosters have the most risk and series pressure-fed and series F-1 pump-fed are intermediate and about equal in risk. This technical risk assessment, along with the cost comparisons summarized in Figure 43, led to the selection of the twin 156-inch parallel burn design as the prime choice and the pump-fed F-1 series-burn as the second choice.

5.4 Orbiter Engine Selection – A series of trade studies were conducted in which various J-2 derivative engine designs as well as the HiPc engine were evaluated for variations in engine number and thrust level. The recommended main engine configuration for the Orbiter is three HiPc, oxygen/hydrogen engines each delivering 470K lb (vac) thrust. The evaluation is summarized in Figure 44. While the recommended configuration was three 450K lb (vac) thrust engines, the thrust

MARK I – MARK II EVOLUTION

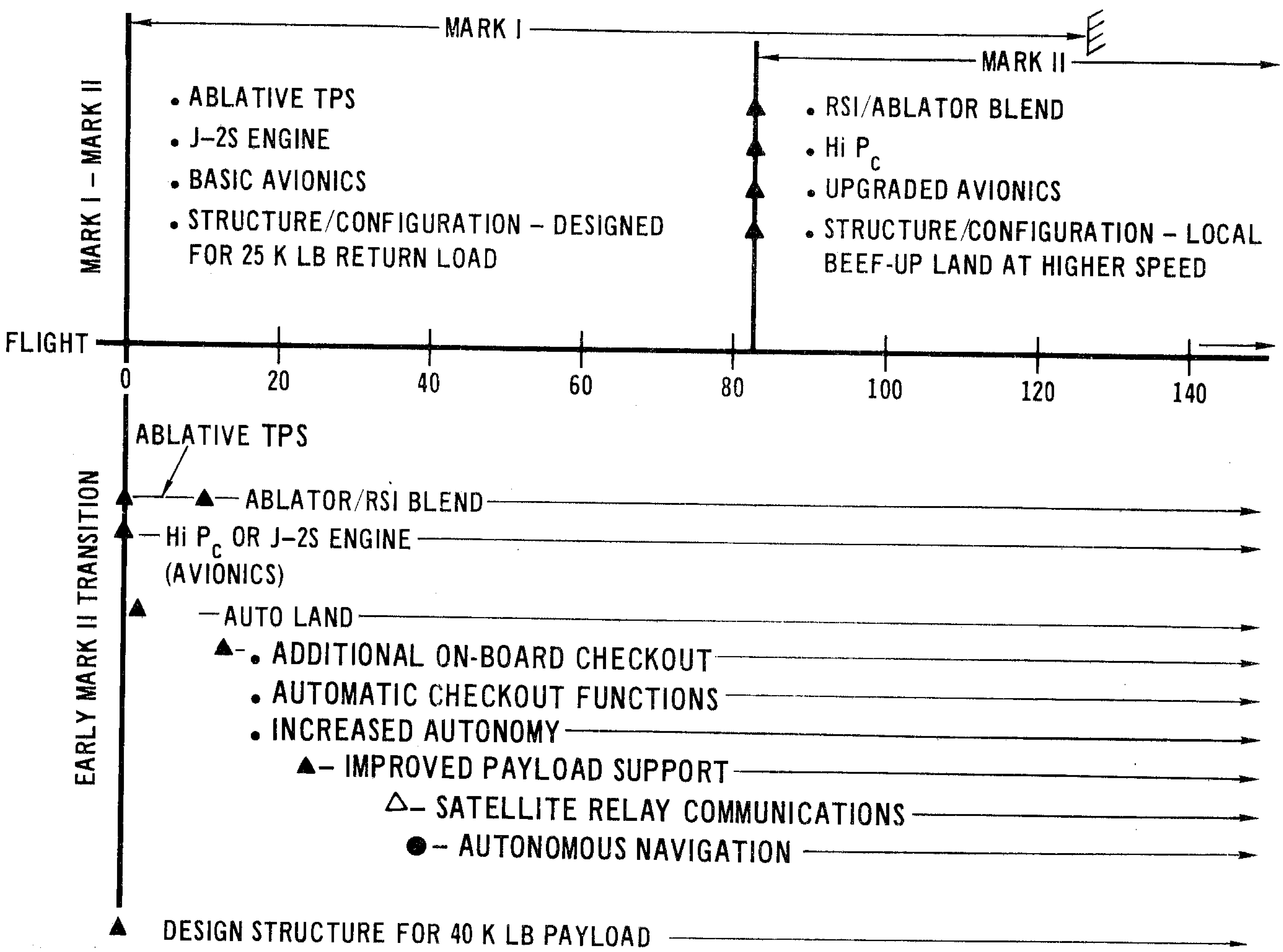


Figure 41

value was revised to 470K lb to be compatible with subsequent vehicle design developments and with the 415 lb reference sea level thrust of the

with the 415L lb reference sea level thrust of the powerhead in the original ICD engine used as an early point design.

5.5 Payload Bay Size – Three Orbiter concepts were investigated early in this study extension to determine their sensitivity to payload bay size. These concepts were the single HO tank, the twin H tanks and the Orbiter with integral tanks. The study was directed toward a two phase program with the Orbiter being stretched later in the program. It was found that a swept-wing Orbiter was amenable to stretching but that the blended wing-body design of a delta-wing precluded stretching. No advantage in cost was found in using a small (12 x 40 ft) payload bay over the larger (15 x 60 ft) bay, but a significant (55%) portion of the anticipated NASA and DOD payload traffic opportunities was lost.

Late in the study a detailed comparison was made between the baseline 15 x 60 ft bay Orbiter with a 65,000 lb easterly payload (40,000 lb payload landed) and a 14 x 45 ft bay Orbiter with a 45,000 lb easterly ascent payload (25,000 lb payload landed). Results of the study shown that bay size reduction is extremely restrictive to mission accommodation: use of the 14 x 45 ft bay would preclude about one sixth of the NASA missions,

PROGRAM RISK COMPARISON

	MARK I ——→ MARK II	EARLY MARK II TRANSITION
PAYLOAD REQUIREMENT (SOUTH POLAR)	DESIRED: 25 K LB ——→ 40 K LB MINIMUM – 10 K LB • INABILITY TO CAPTURE TRAFFIC FOR (4) YRS	40 K LB
ORBITER PROPULSION	J-2S ——→ Hi Pc • USE OF MODIFIED EXISTING ENGINE INITIALLY REDUCES RISK OF SCHEDULE DELAY	J-2S • LEAST RISK HiPc • SOME RISK IF DEVELOPMENT PROBLEMS OCCUR • GREATEST COST SAVINGS
TPS	ABLATOR ——→ ABLATOR & RSI BLEND • EASES RSI DEVELOPMENT TIME TABLE • USE OF ABLATOR FOR 125 FLIGHTS INCREASES COSTS	ABLATOR ——→ ABLATOR & RSI BLEND • FIRST FLIGHTS UTILIZE ALL – ABLATOR SYSTEM DESIGN • EARLY FLIGHTS WILL VERIFY ENVIRONMENT & FLIGHT TEST RSI (IN NON-FLIGHT SAFETY CRITICAL AREAS) • USE OF EXISTING ABLATOR SYSTEM FOR BACKUP ELIMINATES RISK • REDUCED COST
AVIONICS	MINIMUM SYSTEM ——→ UPGRADED SYSTEM • MINIMAL RISK • LOWERS PEAK ANNUAL FUNDING INITIALLY • REDUCES ORBITER CAPABILITY	INTRODUCE MARK II REQUIREMENTS IN STEP SEQUENCE COMMENSURATE WITH PEAK FUNDING CONSTRAINTS • MINIMAL RISK • PROVIDES INCREASED CAPABILITY
STRUCTURE	DESIGN FOR 25 K LB PAYLOAD ——→ STRETCH DESIGN PHILOSOPHY • LIGHTER VEHICLE BUT SOME RISK • LIMITED GROWTH POTENTIAL	DESIGN FOR 40 K LB PAYLOAD • MINIMUM RISK • PROVIDES GROWTH POTENTIAL
CONFIGURATION	LAND AT ≈150 KT WITH 25 K LB PAYLOAD ——→ LAND FASTER WITH 40 K LB PAYLOAD • MINIMUM WEIGHT • LIMITED GROWTH POTENTIAL	LAND FASTER WITH 40 K LB PAYLOAD AND MK I WING • MINIMUM WEIGHT • LIMITED GROWTH POTENTIAL

Figure 42

SYSTEM COST COMPARISON

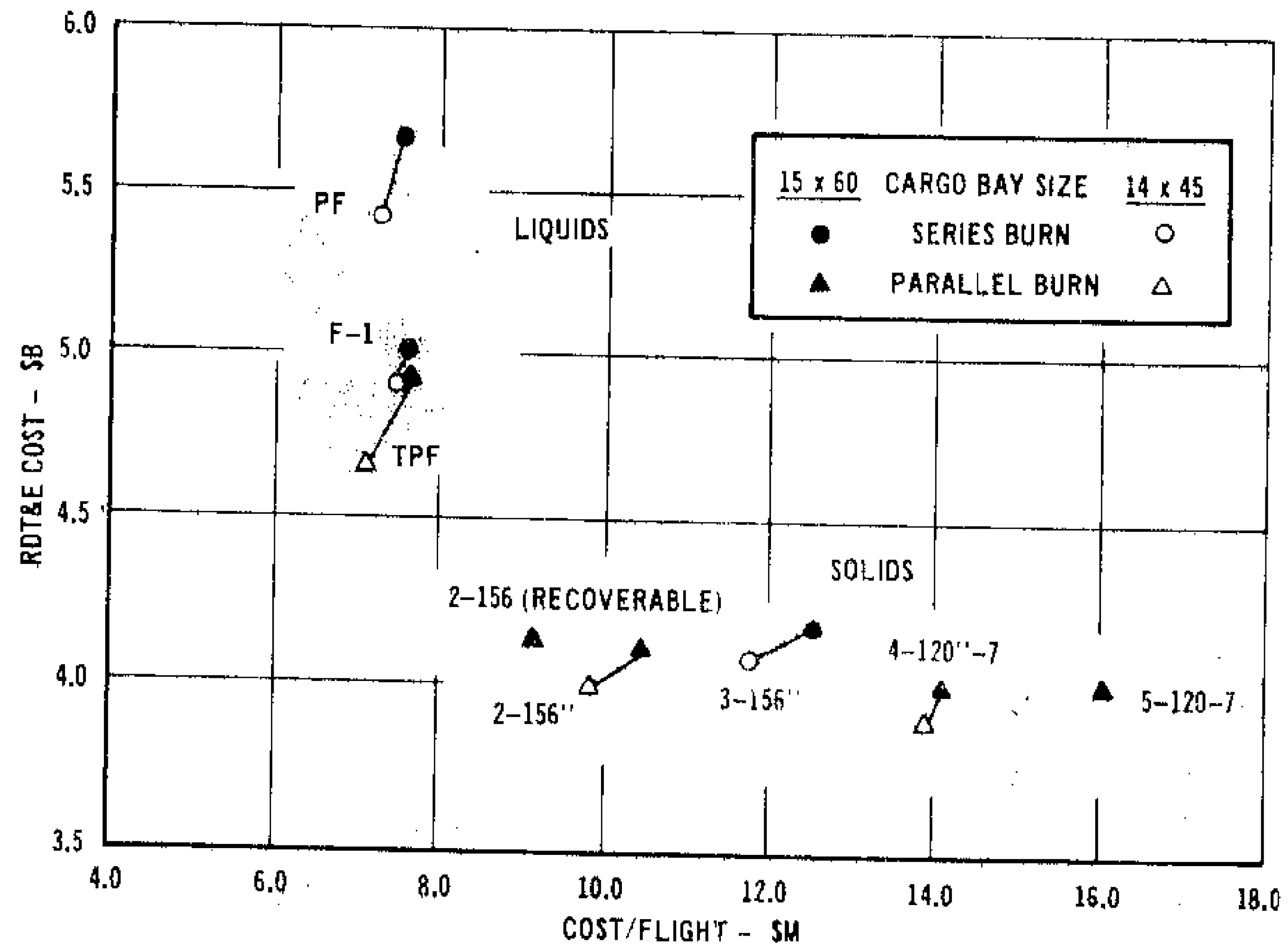


Figure 43

and while about two thirds of the DOD missions could be accomplished with the smaller bay, most would require dual launches. It was also determined that program cost savings with the small bay (ranging from \$202 M to \$576 M) were attributable primarily to reduction in payload weight. Only about one-fourth of the savings is attributable to reduction in Orbiter size.

ORBITER MAIN ENGINE SYSTEM SELECTION

Engine Selection Cost Comparison

BASELINE CONCEPTS	BASELINE J-2S→HiPc	DELTAS FROM BASELINE			
		J-2S (ALONE)	HiPc (ALONE)		
			4-306 K LB (VAC)	3-450 KLB (VAC)	4-450 KLB (VAC)
FLYBACK SERIES BURN					
RDT&E	7,398	-430	-209	-131	-85
PER FLIGHT	6.16	+0.06	0	-0.01	-0.04
TOTAL PROGRAM	12,863	-493	-287	-214	-172
PAF	1,319 (76)	-12	+37	+49	+51
SINGLE RAO SERIES BURN					
RDT&E	5,726	-262	-209	-199	-180
PER FLIGHT	5.46	+0.03	0	0	+0.02
TOTAL PROGRAM	10,894	-330	-287	-320	-300
PAF	1,046 (78)	+22	+37	+36	+32
TWIN RAO PARALLEL BURN					
RDT&E	5,199	-213	-209	-208	-258
PER FLIGHT	6.87	+0.10	0	-0.01	-0.01
TOTAL PROGRAM	10,469	-120	-287	-347	-407
PAF	939 (78)	+32	+37	+25	+16

- ORBITER THRUST OPTIONS
  - 4-6 265K J-2S
  - 4 320K J-2S UPRATED
  - 3-4 300K - 450K Hi Pc
- STUDY PARAMETERS
  - ORBITER GEOMETRY WEIGHT EFFECTS
  - $V_{STAGING}$ VS ORBITER THRUST LEVEL
  - $V_{STAGING}$, GROWTH CAPABILITY, THRUST LEVEL INTERRELATIONSHIPS
  - PROGRAM COSTS

Three Engines Preferred to Four

- ASSUMING CONSTANT TOTAL ORBITER THRUST

CRITERION	NUMBER OF ENGINES 3	4	
SYSTEM RELIABILITY	✓		• SYSTEM RELIABILITY HIGHER WITH FEWER ENGINES PROVIDED
ABORT GAPS	—	—	• NO SIGNIFICANT DISCRIMINATOR BETWEEN 3 AND 4 ENGINE SYSTEM
DESIGN	✓		• FEWER ENGINES ALLOW VEHICLE INTERFACES TO BE SIMPLIFIED
MAINTENANCE	✓		• MANHOURS – 3 HiPc 96 MMH/FLT 4 HiPc 128 MMH/FLT 4 J-2S 160 MMH/FLT
	✓		• FIELD OPERATIONS (SUPPORT, SPARES, TURNAROUND TIME) FAVOR FEWER ENGINES
FLIGHT OPERATIONS	✓		• DATA MONITORING, INSTRUMENTATION, ELECTRICAL POWER CONSIDERATIONS FAVOR FEWER ENGINES
ORBITER WEIGHT		✓	• THREE ENGINE SYSTEM 1000 LB HEAVIER

- SINGLE ENGINE DEVELOPMENT PREFERRED
- 4 J-2S ENGINE SYSTEM FEASIBLE FOR BOTH SINGLE AND TWIN RAO (PRESSURE-FED)– $V_{STG} \geq 7000$ FPS
- HIGHER ORBITER THRUST ALLOWS REDUCED V_{STG} AND LOWER COSTS
- "UPRATED" J-2S BETTER THAN J-2S, BUT HiPc IS SUPERIOR
- 3 – 450 K ENGINES REPRESENTS GOOD COMPROMISE FOR BOTH SINGLE AND TWIN RAO (PRESSURE-FED) SYSTEMS
- 4–450K ENGINES WOULD SAVE MORE THAN \$100 M IN SYSTEM COST FOR PARALLEL BURN. EXPENSE OF ORBITER AERO/CG DESIGN PROBLEMS NOT ACCOUNTED FOR.
- 3 – 450 K ENGINES IS RECOMMENDED SELECTION

Figure 44

6 COST AND SCHEDULE SUMMARY

The cost data are based on a combination of detailed data (vendor budgetary quotes, equipment lists, etc.) and parametric data (use of cost and design scaling relationships based on historical data). The requirements for a management traceability, etc., have a significant impact on program costs, and while not specifically defined or costed, these effects are built into the cost data shown, by virtue of the historical data base from which the estimate was made (primarily Mercury, Gemini, and SIV-B).

The cost data are organized by RDT&E, Production, and Operations activities. RDT&E includes design, development, ground and flight test operations, ground and flight test hardware, and the support effort required for the development and qualification of the system. This covers all development related costs through completion of the flight test program. Production costs include 440 SRM Boosters, three Orbiters and the costs associated with refurbishment of the Orbiter flight test vehicles to make them operational. Operations costs include all recurring labor and material to support a 400 flight 10 year program (445 less 5 flights in RDT&E).

COST SUMMARY

Millions of 1970 Dollars (1)

15' x 60' PAYLOAD BAY ORBITER – PARALLEL BURN 2-156 IN. SRM BOOSTER

COST CATEGORY	RDT&E	PRODUCTION	OPERATIONS	TOTAL
ORBITER				
ENTRY VEHICLE	2,712	514		2,226
TANKS	123	707		830
TOTAL	2,835	1,221		4,056
BOOSTER	259	2,418		2,677
MAIN ENGINES				
ORBITER	497	90	125	712
BOOSTER	0	0	0	0
TOTAL	497	90	125	712
FLIGHT TEST				
ORBITER	226	83		309
BOOSTER	16	0		16
TOTAL	242	83		325
OPERATIONS				
ORBITER	47		1,683	1,730
BOOSTER	47 (2)		235	282
TOTAL	94		1,918	2,012
MAN. & INTEG				
ORBITER	183	79		262
BOOSTER	16	147		163
TOTAL	199	226		425
TOTAL				
ORBITER	3,788	1,473	1,808	7,069
BOOSTER	338	2,565	235	3,138
TOTAL	4,126	4,038	2,043	10,207

(1) EXCLUDES PRIME CONTRACTOR FEE
(2) LAUNCH FACILITIES

Figure 45

The cost of two complete Orbiter and five Boosters is included in the RDT&E cost. The Orbiters are used during the flight test program and then refurbished to become operational vehicles. Three additional Orbiters are purchased for the operational portion of the program, providing a total inventory of five Orbiters. The traffic model was provided by NASA and consists of 445 flights over 10 years with six flights in 1978 and building to 60 flights per year for the last five years for the 15' x 60' payload bay Orbiter and the parallel burn 156-inch SRM Booster.

The costs are summarized in Figure 45 for the Level 3 elements of the Work Breakdown Structure. The main engines are considered GFE. A major criterion throughout the Shuttle study has been to define and select systems and subsystems which result in a least-cost approach. Many other Booster concepts were evaluated as part of the Phase B extension requirements and are described in Report MDC E0558, Part IV.

Funding requirements are shown in Figure 46; it can be seen that peak funding of about $.95 billion occurs in FY 1976. These data are based on meeting a target date for the first horizontal takeoff flight in September 1976, and the first vertical takeoff mated flight in March 1978. The funding shown is for the total program, including main engine and launch facilities costs.

Figure 47, the Space Shuttle Master Program Schedule, summarizes the major activities required to accomplish the NASA program objectives. During the initial portion of Phase C/D, many pro-

FISCAL FUNDING REQUIREMENTS
Parallel Burn, 2-156 In. SRM Boosters

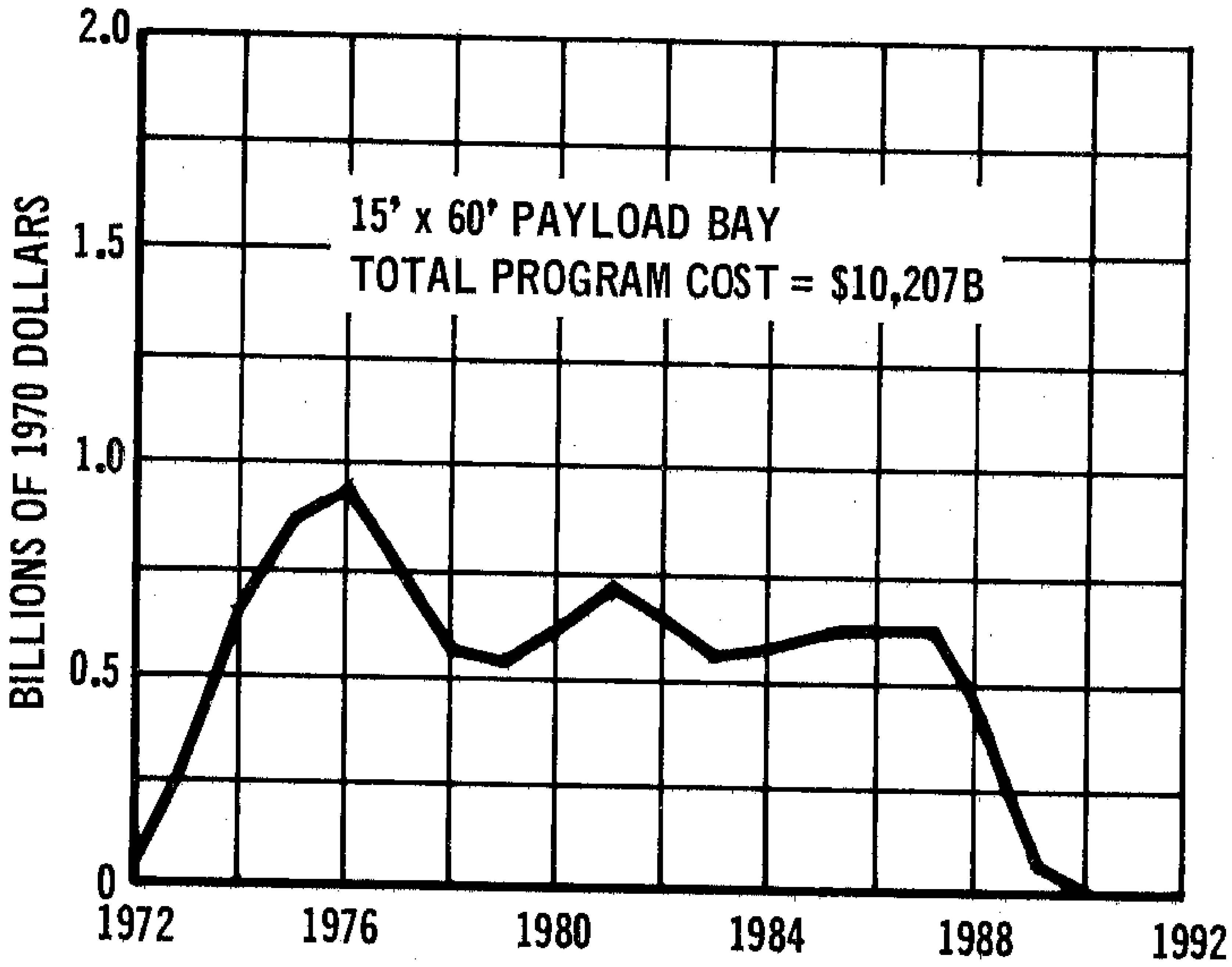


Figure 46

gram and engineering decisions must be made which are critical to the course of the entire program. A summary of these key decisions is shown on the schedule. Each major program element was developed not only to satisfy its own peculiar objectives but also to be integrated with the Space Shuttle objectives so that no one major element dominated or drove the entire program.

The Orbiter vehicles will be used to satisfy the development requirements and to also satisfy the operational requirements through 77 vertical flights. Orbiter No. 3 will be required in January 1982 to satisfy the 41 vertical launches during that year. A total of five Orbiters will be required. A single expendable tank links the Orbiter and the Booster.

A solid rocket motor (twin, parallel burn) Booster will be developed in parallel with the Orbiter. A total of 445 Boosters will be produced. The first five will be utilized to satisfy development requirements.

SPACE SHUTTLE MASTER PROGRAM SCHEDULE

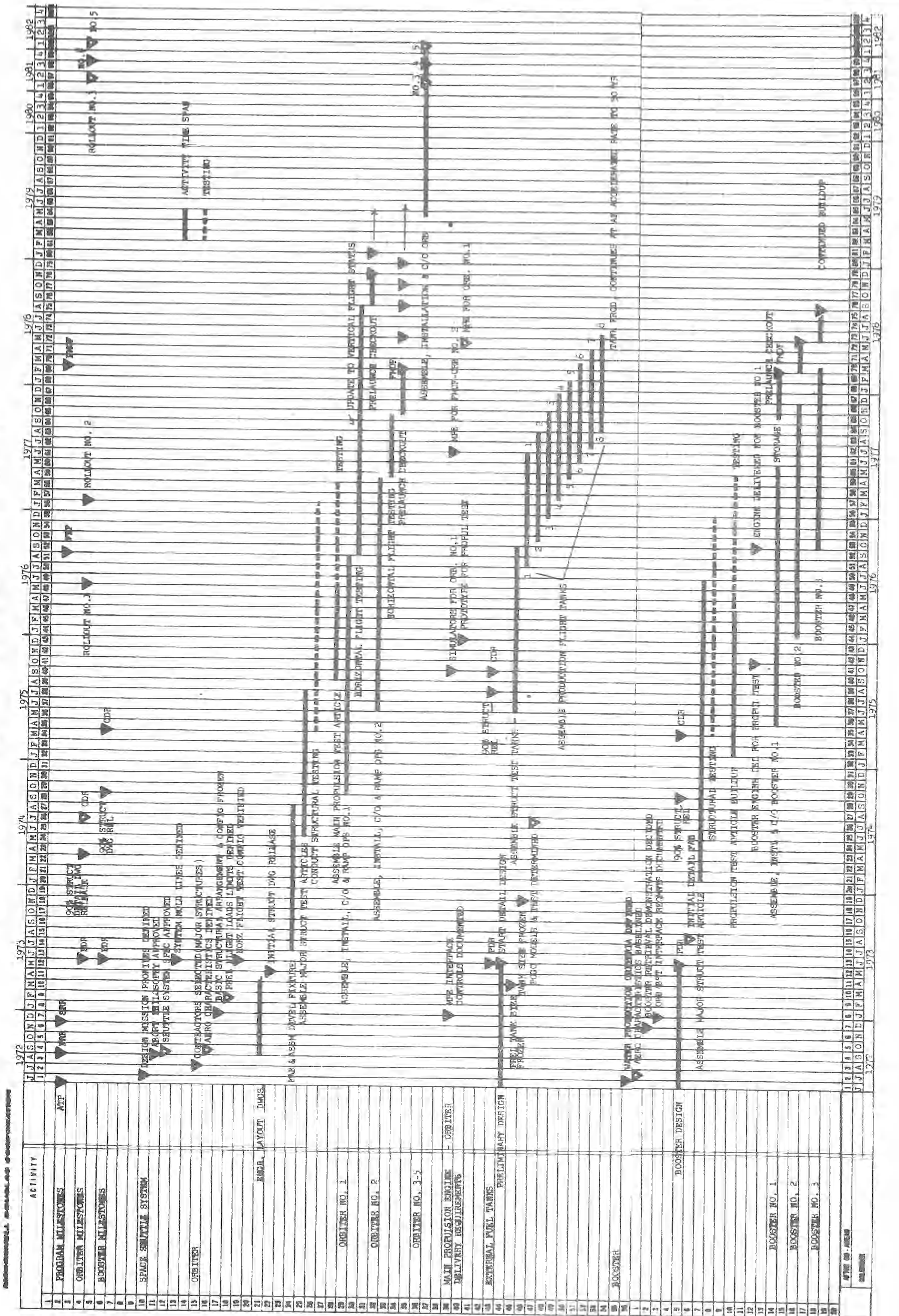


Figure 47