REPORT NO. GDC-DCB69-046 CONTRACT NAS 9-9207



# SPACE SHUTTLE FINAL TECHNICAL REPORT

VOLUME I + CONDENSED SUMMARY

GENERAL DYNAMICS

**Convair Division** 



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31 October 1969

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Prepared by CONVAIR DIVISION OF GENERAL DYNAMICS San Diego, California

#### FOREWORD

This volume of Convair Report No. GDC-DCB 69-046 constitutes a portion of the final report for the "Study of Integral Launch and Reentry Vehicles." The study was conducted by Convair, a division of General Dynamics Corporation, for National Aeronautics and Space Administration George C. Marshall Space Flight Center under Contract NAS 9-9207 Modification 2.

The final report is published in ten volumes:

Volume I	Condensed Summary
Volume II	Final Vehicle Configurations
Volume III	Initial Vehicle Spectrum and Parametric Excursions
Volume IV	Technical Analysis and Performance
Volume V	Subsystems and Weight Analysis
Volume VI	Propulsion Analysis and Tradeoffs
Volume VII	Integrated Electronics
Volume VIII	Mission/Payload and Safety/Abort Analyses
Volume IX	Ground Turnaround Operations and Facility Requirements
Volume X	Program Development, Cost Analysis, and Technology Requirements

Convair gratefully acknowledges the cooperation of the many agencies and companies that provided technical assistance during this study:

NASA-MSFC	Aerojet-General Corporation
NASA-MSC	Rocketdyne
NASA-ERC	Pratt and Whitney
NASA-LaRC	Pan American World Airways

The study was managed and supervised by Glenn Karel, Study Manager, C. P. Plummer, Principal Configuration Designer, and Carl E. Crone, Principal Program Analyst (all of Convair) under the direction of Charles M. Akridge and Alfred J. Finzel, NASA study co-managers.

#### ABSTRACT

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A study was made to obtain a conceptual definition of reusable space shuttle systems having multimission capability. The systems as defined can deliver 50,000-pound payloads having a diameter of 15 feet and a length of 60 feet to a 55-degree inclined orbit at an altitude of 270 n.mi. The following types of missions can be accommodated by the space shuttle system: logistics; propellant delivery; propulsive stage delivery; satellite delivery, retrieval, and maintenance; short-duration missions, and rescue missions.

Two types of reusable space shuttle systems were defined: a two-element system consisting of a boost and an orbital element and a three-element system consisting of two boost elements and an orbital element. The vehicles lift off vertically using high pressure oxygen/hydrogen rocket engines, land horizontally on conventional runways, and are fully reusable. The boost elements, after staging, perform an aerodynamic entry and fly back to the launch site using conventional airbreathing engines. Radiative thermal protection systems were defined to provide for reusability. Development programs, technology programs, schedules, and costs have been defined for planning purposes.

During the study, special emphasis was given to the following areas: System Development Approaches, Ground Turnaround Operations, Mission Interfaces and Cargo Accommodations/Handling, Propulsion System Parameters, and Integrated Electronics Systems.

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#### 1 INTRODUCTION

This volume contains a condensed summary of a conceptual definition study to define a reusable space shuttle system. The study was conducted by Convair division of General Dynamics for the NASA to support future space exploration activities.

The initial spectrum of vehicles investigated in this study included liquid and solid expendable two stage systems, a reusable first stage with an expendable second stage, and a fully reusable concept. Except for the reusable system all configurations also included 12 man reusable spacecraft and expendable cargo/propulsion modules. During the study NASA revised the spectrum to include only reusable concepts. The following sections therefore include only the results of the study or reusable space shuttle systems. The results of the work on the partially reusable systems are documented separately in Reference 1.

During the study special emphasis was given to: integrated electronics systems, propulsion system parameters, ground turnaround operations, mission interfaces and cargo accommodations, handling, and system development approaches.

#### 2 STUDY OBJECTIVES

The specific study objectives, paraphrased from the contract statement of work, were:

- 1. Derive conceptual designs, ius resources analysis identifying the development, manufacturing, procurement, testing, RDT&E and operations requirements for logistics space vehicle systems to support earth orbital programs in the post 1974 time period. The major design approaches will attempt to:
  - a. Achieve order of magnitude reductions in the recurring costs of the total logistic support operational system.
  - b. Achieve significant advances in the inherent safety of the space vehicle system.
- 2. Establish requirements for research and technology development.

While the baseline mission was to accomplish space station crew rotation and resupply, the space shuttle system has multi-mission capability. The general characteristics of alternate missions as supplied by NASA and within the capability of the space shuttle system described in this report are shown in Figure 1.

MENTICH	ORBIT AI TITUDE m.ml.)	ORBIT INCLINATION (deg)	MAIN PRO- PULSION ON-ORBIT &V (fps)	₹R-3 РАУ1/ОАО САРАВНИТУ ЛЬ ур)	NUMBER OF MISSIONS
SPACE STATION SPACE BARE LOOSTICS	270	58	(Ann	50,000	113
DELEVERY OF PROMUNE STADE AND CAYLOAD	100	29, 5 55	1010	87,000 65,000	41
PLACEMENT, "ETRIEVAL, SERVICE AND MADITENANCE OF SATELLITES	100 800	24,6 90	1010/2040* 2010/2346*	876ng/ <b>56</b> 620* g/0*	22
DELEVERY OF PROPELIANTS	200 300	28, A 88	1460	74,400 48,900	100
SHORT DURATION ORBIT	1 on 300	28, 5 P0	1820	87,800 22,900	22
SPACE RESCUE	278	54	3990**	~**	-
NOTE: MATELLITE BERVICE, MAIN ++RENDEZOIR WITHIN 24 HOL	TENANCE, O	N RETRIEVAL. PT OF REACUE A	ONAL.		

Figure 1. Mission Requirements

#### 3 RELATIONSHIP TO OTHER NASA EFFORTS

This is one of four concurrent NASA studies to define the conceptual design, performance, operations, and technology requirements of a space shuttle system. The other three studies are being conducted by Lockheed Missiles and Space Company, McDonnell Douglas Corporation, and North American Rockwell Corporation under contract NAS 9-9206, NAS 9-9204, and NAS 9-9205 respectively. Definition studies of a 12-man space station to be launched in the 1975 time period are being conducted by McDonnell Douglas and North American Fochewell under contracts NAS 8-2540 and NAS 9-9253. A conceptual design study is being conducted by the Convair division of General Dynamics Corporation, contract NAS 8-25051, to define common modules necessary to perform the NASA candidate experiment program at minimum cost.

#### 4 METHOD OF APPROACH AND PRINCIPAL ASSUMPTIONS

Certain vehicle characteristics were defined by NASA. These included:

Vehicle Concepts: Two and three-element two-stage systems with and without propellant crossfeed.

Engine Type and Thrust Level: High chamber pressure oxygen-hydrogen Bell nozzle engines with a sea level thrust rating of 400,000 pounds.

Reaction Control: An oxygen-hydrogen reaction control system.

Payload Weight and Size: 50,000 pounds, 15 feet in diameter and 60 feet long.

On-Orbit  $\Delta V$ : 2,000 feet per second.

Weight Contingency: 10 percent of vehicle design dry weight.

 $\Delta V$  Contingency: 3/4 of 1 percent of the ideal  $\Delta V$  to the point of injection for flight performance reserve.

Mission Orbit/Inclination: 270 n.mi./55 degrees.

Mission Duration: 7 days nominal; additional staytime is at expense of payload.

Maximum Acceleration: 3g with passengers; 4g without passengers.

The primary study task, therefore, was to optimize each candidate system to satisfy these requirements by:

- 1. Determining optimum  $\Delta V$  split between stages.
- 2. Determining optimum engine area ratios and mixture ratio.
- 3. Defining design loads and temperatures plus design studies to satisfy these requirements.
- 4. Providing for satisfactory vehicle control and stability characteristics.
- 5. Determining system inert weight and liftoff weights.
- 6. Defining alternate mission capability.

The vehicle concepts evaluated are fully reusable. No hardware is expended during the mission and vehicle has all-azimuth launch capability after the development program is completed and reliability established. All of the concepts have vehicles with decoupled hypersonic and subsonic modes. This decoupling is obtained by use of stowable wings on both stages, which permits independent optimization of the hyperscnic shape for reentry and the subsonic shape for terminal maneuvers, approach, and landing performance.

Aerodynamic shape of the first and second stage vehicles for the two and three-element systems were initially similar to minimize the cost of developing the aerodynamic vehicle. The studies indicate, however, that if emphasis is placed on maximum performance and low recurring costs rather than on low development costs, similar elements do not result in maximum performance, minimum weight, or minimum recurring cost. For the final two-element system (FR-3), the boost and the orbit elements were tailored to their mission requirements.

#### 5 BASIC DATA GENERATED AND SIGNIFICANT RESULTS

Two fully reusable space shuttle systems were assigned by the NASA to Convair for investigation during the final period of the Phase A study. The systems are:



FR-3. This is a two-element system. The configuration is necessarily asymmetric, with the booster element considerably larger than the orbiter to achieve minimum system weight. The final configuration has a gross liftoff weight of 4.33 million pounds, a total system dry weight of 683 thousand pounds, and an overall length of 235 feet.



FR-4. This is a three-element system. The central orbiter element is located between the two identical booster elements which are larger than the orbiter and have no commonality except for the main boost rocket engines and the turbofan flyback engines. The final configuration has a gross liftoff weight of 4.92 million pounds, a total system dry weight of 837 thousand pounds, and an overall length of 219 feet.

The FR-3 and FR-4 configurations are both sequentially staged. That is, the orbiter element engines are ignited at the staging point.

The FR-4 system is not competitive with the FR-3 at launch rates above 35 per year. It is recommended that the FR-4 system effort be discontinued. For this reason, only the FR-3 system will be covered in this condensed summary volume. Both systems are described in more detail in Volume II.

#### 5.1 FR-3 VEHICLE DESIGN

The operation of the system is as follows. The vehicle is launched vertically and pitches over into a ballistic trajectory to the staging point. After separation, the orbiter proceeds to a 43 n.mi. injection point and eventually to a 270 n.mi., 55 degree inclination mission orbit, to rendezvous and dock with the space station. After a staytime not exceeding seven days, for the nominal mission, the orbiter leaves the station, ret-







rofires, and reenters (Figure 2). After entry, the wings and turbofan landing engines are deployed and the orbiter arrives at the launch site making a conventional airplane type landing. After separation from the orbiter, the booster element proceeds through a series of energy management maneuvers to depress the downrange distance, and then cruises back subsonically to land at the launch site using turbofan engines (Figure 3). Landing is made on a 10,000 too' runway with a nominal touchdown velocity of 180 knots.

5.1.1 FR-3 LAUNCH CONFIGUR-ATION. The booster element and orbiter element of the FR-3 two stage sequential burn systems are

shown in Figure 4 in the launch configuration. The optimum staging velocity for minimum gross liftoff weight is approximately 11,000 fps resulting in a larger booster (gross volume 236K cu ft) than orbiter (gross volume 89K cu ft). Both elements have stowable wings, the only aerodynamic surfaces exposed in the launch configuration being the V tail stabilizers. As shown in the end view the elements have their flat surfaces adjacent. This minimizes the stage attach linkage length and provides a clean launch configuration. It eliminates interference problems by placing the booster



Figure 4. FR-3 Launch Configuration

and orbiter stabilizers 180 degrees apart. Also shown are the fifteen 400K pound sea level thrust two-position rocket engines in the booster and the three fixed nozzle engines in the orbiter. Because of the offset c.g. in this asymmetric launch arrangement it is necessary to cant the average thrust vector of the booster element engines to a 6-1/2 degree basic neutral position. By gimballing either side of this, the configuration is controllable from liftoff through staging. The elements are arranged in the side view in a tail-to-tail position, with the rocket engine exits aligned. This reduces loads on both elements, eliminates base effects of the orbiter on the booster (relative to a nose-to-nose arrangement) and facilitates ground handling and support by locating the orbiter base nearer the pad. The orbiter engines are not ignited at liftoff in the present concept, however, the arrangement would allow it if required.

5.1.2 <u>THE FR-3 ORBITER ELEMENT</u> (Figure 5). The body consists of a constant section with a tapered forebody terminating in a hemispherical nose. The constant section consists of a flat bottom with sides tapered inward at a 12 degree angle, and a full upper radius. The upper radius allows maximum use of the state-of-the-art cylindrical propellant tankage. The flat bottom improves the hypersonic L/D and provides convenient stowage space for the wing, used for subsonic approach and landing. The V tail provides longitudinal and directional stability across the hypersonic transonic and subsonic flight regimes.



Figure 5. FR-3 Orbiter Three-View

The stowable wings are shown deployed to the nominal subsonic position. Doors for installation and removal of the 15-foot-diameter by 60-foot-long payload (shown on the upper surface) are accessible throughout the launch operation. An extension of the lower surface protects the bell nozzle rocket engines during entry. This extension is in the form of a flap for hypersonic nose-down trim. The orbiter engines are the required 400K pound sea level thrust design with the nozzles extended to give an expansion ratio of 160 and a vacuum thrust of 472K pounds.

The general arrangement (Figure 6) consists of a nose compartment 14 feet long. This accommodates the pilot and copilot, instruments, controls, and consoles in a cabin with conventional side-by-side seating. Retractable visors protect the windshields during aerodynamic heating phases. Avionics equipment is also installed in the forward compartment. A subsystems compartment aft of the crew compartment pressure bulkhead



#### Figure 6. FR-3 Orbiter General Arrangement

accommodates power generation and distribution (hydraulic and electric) and environmental control and life support subsystems.

The flyback engine compartment is aft of the subsystems compartment. Three existing turbofan engines in the 21,000 pound sea level static thrust category are used. These engines are deployed via pivots for use during the subsonic approach. JP-4 fuel is provided for an approach and one go-around.

The main  $LO_2$  tank and the main  $LH_2$  tank form an integral part of the overall body structure. The space below the main  $LO_2$  tank accommodates the insulated tanks for on-orbit maneuver and reaction control system propellants. Long-term-storage  $LH_2$ tanks for on-orbit maneuver are stored below the payload envelope, along with additional main-impulse  $LH_2$  tanks for improved volumetric efficiency. Internal insulation is used in all  $LH_2$  tanks. The stowed wing is deployed via screwjacks, slaved together via reduction gear boxes and driven by hydraulic motors. Segmented doors in the body close as soon as the wing deployment is complete. The entire orbiter vehicle is protected at reentry by a radiation thermal protection system (TPS) using primarily cobalt alloy cover panels over microquartz and dynaflex insulation on the lower surface heat shield and 811 titanium over microquartz insulation on the upper surface. The stabilizers are similarly protected.

5.1.3 <u>THE FR-3 BOOSTER ELEMENT (Figure 7)</u>. The FR-3 booster design goal was to maximize the propellant volume on board relative to the total volume, while containing this volume in state of the art cylindrical tanks with conventional tank domes. The body cross-section consists of a flat bottom, inward sloping sides of 9 degrees nominally, and a full upper radius. This cross-section is held constant over most of the vehicle length, including the base. The blunt nose improves volumetric efficiency and reduces nose temperatures at entry. It increases hypersonic drag and reduces the downrange distance. The two-man crew compartment is installed above the nose. The wing arrangement is similar to that of the orbiter.

The propellant tanks and intertank section are the main load carrying body assemblies, and the lower heat shield and the sides are primarily fairing structure. The tanks are 33 feet in diameter and are of aluminum alloy with external frames and stringers. The lower heat shield is made up of nickel alloy cover panels over microquartz insulation. The upper surface and sides are faired with 811 titanium panels which stand off from the main structure and are insulated only as necessary. The internal arrangement consists of the spherical JP-4 tank, used for flyback, together with the four, 52, 500 pound sea level static thrust RB 211-56 flyback turbofan engines which occupy the nose



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compartment. The nose compartment also houses the power generation and distribution subsystems and the avionics equipment. The nose landing gear is of conventional aircraft type high-heat-treat alloy steel, oleo shock, with 36 by 11 inch type VII dual tires. The main gear consists of four 56 by 16 inch type VII tires per main strut bogey. The tricycle gear arrangement is designed for the maximum landing weight of 565K pounds. Four  $LO_2$  lines lead aft where they are manifolded in the thrust structure area to feed the fifteen 400K pound sea level thrust, two-position bell nozzle high pressure rocket engines. The engines are protected against reentry heating by an extension of the lower surface. Dimensional data for the FR-3 system is shown in Figure 8.

5.1.4 FR-3 MISSION PROFILE AND VELOCITY REQUIREMENTS. These are shown in Figure 9 where the major trajectory parameters are given and the mission is explained in detail. The variation of system weights with liftoff F/W is shown in Figure 10. The design point with the selected number (15) of liftoff engines is shown.

	Booster	Orbiter
BODY VOLUME (FT3)	236,000	89,000
BODY WETTED AREA (FT <sup>2</sup> )	26,610	14,900
BODY PLAN FORM AREA (FT <sup>2</sup>	8,170	4,910
ENTRY PLANFORM LOADING (LB-FT <sup>2</sup> )	69	59
BASE AREA (FT <sup>2</sup> )	1,200	632
WING AREA (FT <sup>2</sup> )	2,901	1,336
STABILIZER TRUE AREA (FT <sup>2</sup> )	2,283	1,397
BODY REFERENCE LENGTH (FT)	210.1	179.2
LAUNCH CONFIGURATION LENGTH (FT)	235.5	



The variation of gross liftoff weight with staging velocity and the current design point are shown in Figure 11.



Figure 10. FR-3 System Weights versus F/Wat Liftoff

#### 5.2 AEROTHERMODYNAMICS

The FR-3 orbiter and booster elements were analysed using the nominal entry trajectories. The orbiter entry trajectory was initiated at 400,000 feet and approximately 25,000 fps at an initial flight path angle of -1.0 degrees and an angle of attack of 37 degrees. The booster recovery trajectory was initiated at the staging conditions shown earlier.





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### Figure 9. Mission Profile and Velocity Requirements

Volume I



Figure 12. Orbiter Temperatures



Figure 13. Booster Temperatures

The orbiter peak radiation equilibrium temperatures were determined from temperature histories for the launch and entry trajectories. Peak entry temperatures for 800 n.mi. cross-range were used for an input for thermal protection system material selection.

Except for stagnation regions the surface temperatures were less than 2,000F as shown in Figure 12.

The booster radiation equilibrium temperatures shown in Figure 13 are considered conservative because the large nose radius will produce flow field properties that give lower heat transfer rates, and the relatively rapid transients in conjunction with the low structural temperatures associated with the propellants would reduce the surface temperatures significantly below the radiation equilibrium temperatures.

The minimum total TPS weight of 35,370 pounds was calculated for a 400 n.mi. crossrange which was 2,630 pounds less than the 37,900 required for the 800 n.mi. oncearound abort cross-range. Increasing the cross-range from 800 n.mi. to 1,500 n.mi. increased the total TPS weight from 37,900 to 41,830 pounds, an increase of 3,930 pounds.

#### 5.3 THERMOSTRUCTURAL DESIGN

5.3.1 INTRODUCTION. A preliminary design investigation was accomplished on the primary structure TPS. The objectives were to determine feasibility, verify the thermostructural weight and identify critical technologies.

5.3.2 CANDIDATE CONCEPTS. The most significant competing candidates include integral versus non-integral propellant tanks and hot versus insulated structure. A TPS is essential on the windward surfaces at least for the orbital element in view of the high surface temperatures (1800 -2000°F). In conformance with the requirement

for complete reusability, an ablative system was not considered applicable except possibly as a backup for early flights. Current temperature predictions indicate a peak of 800°F for the sides and upper surface aft of the forward 20 percent of the body and a hot structure or titanium is an attractive candidate for this area. The use of titanium, with its low product of modulus of elasticity and coefficient of expansion, would greatly mitigate thermal stress problems, and a tight-sealed structure would facilitate purging and eliminate hot gas and moisture ingress problems. The primary disadvantages with this concept are difficulty in access for inspection and repair of propellant tanks and technical risk due to current uncertainties in heating rates. An alternative to this concept features a fully thermally protected body structure into which circular cross-section propellant tanks are integrated. The major advantages of this arrangement are flexibility, and relatively easy access for inspection and repair. The flexibility of the design readily permits a change of surface materials to accommodate temperatures higher than predicted. In this respect a potential for vehicle growth in maneuvering capability is also available. On the basis of minimized technical risk the fully thermally protected concept was adopted for baseline thermostructural design, although the hot structure concept is still considered a prime candidate.

5.3.3 BASELINE DESIGN. Figure 14 depicts the baseline structure of the FR-3 orbital element. The body shell is a semi-monocoque with integral tanks of circular cross-section. The TPS which controls this structure to 200°F follows the bell shaped periphery shown in the cross-sections. The tank structural material is aluminum for compatibility with liquid oxygen. The use of higher temperature resistant materials is under consideration for increased efficiency and provision of fail-safe for local heat shield failure. Titanium and composite materials are used outside the tank area in the baseline design. The wing utilizes a torsion/bending box with stringer stiffened skins. Titanium is the structural material. The wing pivot uses a plain bearing adapted from the F-111. Wing moments are transmitted across the body by a truss beam, an integral part of a bulkhead which distributes the wing shear into the body shell. Wing torque is reacted by a beam attached to the pivot and extending forward to a reaction bulkhead.



Figure 15. Thermal Protection System

The TPS shown in Figure 15 consists of metallic panels over fibrous insulation. Large panels are used to minimize complexity and to reduce leakage through slip joints. The windward surface panels are supported from the primary structure by standoff posts. Degrees of freedom at the attachment to the post accommodate thermal expansion.

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Figure 14. Orbiter Structure Arrangement

Volume

A metallic barrier across the inner face of the insulation inhibits the flow of hot gas due to the pressure differential between the windward and leeward surfaces of the vehicle. The leeward panels are attached to the frames via standoff fittings which also incorporate slip features to permit expansion. Cover panel materials for the orbiter consist of HS-188 cobalt alloy for the windward and titanium for the leeward surfaces with some use of coated refractories, TD-NiC, and Hastelloy X in critical areas. The boost vehicle used HS-188 for the entire windward surface and titanium for 90 percent of the leeward surface.

5.3.4 ANALYSIS. The mission profile was investigated to identify the design cases for the primary structure and TPS. For the body shell analysis was required for ground winds, maximum thrust, booster burnout, maximum  $\alpha q$  and  $\beta q$ , subsonic gust, and landing. Other portions of the thermostructural system are affected by pressure loads, flutter, sonic tatigue, and high temperature considerations. In addition to the loads and other data provided by analysis, some discrete design criteria were established: safety factors, dynamic amplification factors, and material temperature limitations. Major components such as wing and body were subjected to shell analyses to facilitate material selection, to establish load intensities and to size the structure for design and weight estimates.

#### 5.4 PROPULSION SYSTEM DESIGN

5.4.1 <u>MAIN PROPULSION</u>. Main propulsion for the FR-3 is provided by high chamber pressure hydrogen-oxygen bell nozzle engines. The main engines operate in a pump fed mode during boost and on-orbit to provide orbit maneuvers. Characteristics of the engines are summarized in Figure 16.

	BOOSTER	ORBITER
NUMBER OF ENGINES	15	3
SEA LEVEL THRUST (LB)	400,000	-
VACUUM THRUST (LB)	462,000	472,000
SPECIFIC IMPULSE, SFA LEVEL (SEC)	389	-
VACUUM (SEC)	449, 5	459
AREA RATIO, SEA LEVEL	35	35
VACUUM	80	160
O/F RATIO	6, 5	6.5
LH2-NPSH 1007F (FT)	60	60
LO <sub>2</sub> -NPSH 100%F (FT)	16	16
ENGINE WEIGHT (LB)	4400	4600
TYPE NOZZLE	BASE	BASE

#### Figure 16. FR-3 Engine Performance

Booster engine area ratios are 35/80, operating retracted at an area ratio of 35 to an altitude of 30,000 feet. The orbiter engine has an area ratio of 160. The nominal engine mixture ratio is 6.5, based on a tradeoff considering the lower structural weight of higher mixture ratios and the higher specific impulse and therefore lower propellant weight at lower mixture ratios.

A 15 engine configuration was selected on the basis of minimum structure weight,

which is a good indicator of vehicle cost. During the initial phase of ascent, the 15 booster engines operate at maximum nominal thrust providing a liftoff F/W of 1.387. The engines are throttled to limit maximum F/W to 3g, reaching 60 percent of nominal thrust at staging. The orbiter is also throttled to maintain 3g, reaching 70 percent of nominal thrust at injection.

For orbit maneuvers, one orbiter engine is used in a pump fed mode at 10 percent of nominal. Propellants for the orbit maneuvers are provided to the engine from man-

euver tanks through lines insulated to minimize boiloff. Propellants are settled by use of the ACS and engines are cooled for start by overboard dump. Use of a pressurc-fed idle mode for settling and cooldown would substantially improve performance, but at increased development risk.

5.4.2 ATTITUDE CONTROL. The attitude control subsystem uses 48 attitude control thrusters with nominal thrusts of 2500 pounds each. The thrusters are supplied with  $GO_2$  and  $GH_2$  from high pressure accumulators, sized to provide the propellant required for entry control. These accumulators are charged with main engine bleed gas during ascent and recharged in orbit from compressors using residual gases and liquids from the main tanks. The engines may be operated at low pressure directly from the main tanks during certain orbital phases where high thrusts are not required and low impulse bits are desired.

5.4.3 FLYBACK AND LANDING ENGINES. The final FR-3 vehicle has airbreathing flyback engines in the booster element and landing engines in the orbiter element. Booster engines will be deployed at an altitude of 25,000 feet or higher and windmill-started during glide to a normal cruise altitude of about 15,000 feet. Orbiter engines will be deployed at 15,000 feet or higher and windmill-started during glide toward the landing site. The engines will then operate at idle to about 1500 feet when they will be brought to the thrust required for a powered approach and landing.

Four Rolls Royce RB 211-56 engines were selected for the final FR-3 booster. Each engine has a maximum sea level static thrust rating of 52,500 pounds. Installed performance during flyback conditions is:

Cruise Altitude (ft)	15,000
Cruise Velocity (knots)	269
Required Cruise Thrust (lb)	78,386
Available Thrust at Max Cruise Rating (lb)	79,625
Available Thrust at Emergency Max Cont. (lb)	85, 586

If one engine fails, the flyback altitude is reduced to 7,900-ft and the three remaining engines are operated at the maximum continuous emergency rating.

Three Pratt and Whitney TF33-P-7 turbofan engines were selected for the orbiter vehicle to provide for powered landing with go-around capability. This older engine has a bare thrust-to-weight ratio of about 4.5 and so weighs more than an advanced turbofan engine of the same thrust rating but there are no advanced turbofan engines in this thrust range under development. Installed performance for the selected configuration under landing conditions is:

Velocity (knots)	178
Required Thrust (lb)	37, 135
Max Available Thrust (lb)	47, 520
Climb Capability: Rate (ft-min)	645
Angle (deg)	2.1

#### 5.5 SUBSYSTEMS

The major features of the selected subsystems are presented in Figure 17. In general, the subsystem designs incorporate redundancy and backup modes to fail operational after the first failure and to fail safe after a second failure. The electronic subsystem is designed for full operation after two failures and to fail safe after a third failure.

SUBSYSTEM		CHAR/	REMARKS	
		BOOSTER	ORBITER	1
۸.	ELECTRICAL POWER 1. ASCENT/ORMT/RFFNTRY 2. POWERED FLIGHT	NI-Cd BATTERY FUEL CELL ENGINE DRIVEN GENERATOR		CONCEPTS FOR AL SUBSYSTEMS ARE COMMON FOR BOT
B. AERODYNAMIC CONTROL 1. PRIMAMY 2. SECONLARY		ELEVONS/RUDD WING TRAIL	FR-3 & FR-4 BUT SIZING MAY BE DIFFFRENT	
c.	ENVIRONMENTAL CONTROL & LIFE SUPPORT 1. SOLIDS & ODOR CONTROL	NON E	PARTICULATE FULTER & ACTIVATED CHARCOAL	
	2. CO <sub>2</sub> CONTROL 3. HUMIDITY CONTROL 4. THERMAL CONTROL	NONE CONDENSATION RAM AIR	LIOH CONDENSATION RADIATOR/SUBLIM- ATOR/JP-4	
	5. ATMOSPHERE	8, 1., N2-02	7, 3 PSIA N2 - 2, 7 PSIA	02
D,	HYDRAULIC POWER GENERATION H2-02 DRIVEN TURBOPUMP		VEN TURBOPUMP	
E.	INTEGRATED ELECTRONICS 1. SYSTEM MONITORING & CHECKOUT 2. INFORMATION DISPLAY 3. GUIDANCE & NAVIGATION 4. AUTOMATIC LANDING	DATA MULTIPLEXING CATHODF-RAY TUBE GIMBALED IMU'S & STRAPDOWN UNITS CURRET 'AIRLINE 1L8		

Figure 17. Subsystems Summary

5.5.1 ELECTRICAL **POWER GENERATION** AND DISTRIBUTION. These systems are similar to the booster and to the orbiter. The booster uses rechargeable Ni-Cd batteries during ascent and descent. During cruise and landing, power is supplied by engine driven alternators with conversion and conditioning to provide 28 vdc and 115/200 vac, The orbiter 400 Hz. uses fuel cells for

ascent, orbital, and reentry operation. All using equipment accepts 28 vdc power and provides conditioning as necessary at individual subsystems. Two 4.5 kw fuel cells, each capable of supplying full vehicle load, are operated continuously in parallel. If a fuel cell system fails a remotely activated Ag-Zn battery is provided which will assure a safe return if the second fuel cell system should also fail. The  $H_2$ - $O_2$  reactants are stored supercritically with full redundancy in dual tankage. Electrical energy is provided by engine-driven alternaters during powered flight on the return phase as described for the booster.

5.5.2 <u>AERODYNAMIC CONTROL</u>. The aerodynamic control subsystem for both orbiters and boosters uses primary and secondary control systems. The primary system includes flaps, ruddervators and wing spoilers. Secondary flight control is supplied by wing trailing-edge flaps.

Three independent hydraulic systems supply power to the primary system. Three hydraulic actuators controlled by a triplex hydraulic valve are used to position each control surface. This is a fly-by-wire system and command signals include triple redundancy with monitoring to detect failures. Secondary controls are provided by two of the above independent hydraulic systems.

5.5.3 <u>ENVIRONMENTAL CONTROL AND LIFE SUPPORT (EC/LSS)</u>. These subsystems are common for both the two-and three-element systems and were designed for a baseline

seven-day mission with excursions up to 30 days. Because the basic booster flight mission is lass than 1.5 hours, the EC/LSS can be extremely simple. The proposed system requires no active control of O<sub>2</sub> or CO<sub>2</sub>. The cabin compartment, designed for low leakage, will be secured with sea level pressure and gas composition. During launch the cabin pressure regulator will allow cabin pressure to decay to 10 psia and to repressurize to atmospheric pressure on dement. Oxygen enrichment and/or use of oxygen masks will be required unless cruise is at an altitude less then 12,000 feet. Thermal control is by thermal inertia during ascent and descent phases nominally less than 9 minutes. Thermal control during cruise-back is by use of a ram air heat exchanger interfacing with the internal water coolant loop.

The orbiter subsystem provides a shirtsleeve environment at 10 psia with an  $O_2$  partial pressure of 2.7 psia. Pressurization and compositional control is provided from supercritically stored  $N_2$  and  $O_2$  and a two-gas sensing and control unit. Atmosphere purification is accomplished through use of particulate and activated charcoal filters for solids and odor control. LiOH beds are used for  $CO_2$  control. A dehumidifying heat exchanger with centrifugal water separation provides humidity control. Thermal control is achieved by use of space radiators with supplementing sublimators for peak or abnormal thermal loads.

Water management (without reclamation), food and waste management, and personal hygiene functions are provided with modular add-on features to accommodate missions in excess of the seven-day baseline mission.

5.5.4 <u>HYDRAULIC POWER GENERATION</u>. Because of the wide disparity between aerodynamic control surface loads and strictly electrical power loads, an  $H_2$ - $O_2$  powered turbopump system (APU) will provide hydraulic system power. An independent pneumatic system is provided for emergency actuation of some hydraulic system loads. The APU fuel, obtained from the main propulsion system residuals existing at the end of the orbital mission.is supplied at 100 psi from the reaction control supply system.

5.5.5 <u>INTEGRATED ELECTRONICS</u>. The integrated electronics subsystem is configured with a goal of lowering operating costs. The autonomous operation minimizes support required from extensive ground operations and lowers cost, but creates new requirements in the vehicle electronics subsystem (including multi-purpose displays, computers and data transfer).

Computer-driven CRT displays enable the crew to control and check out the vehicle. Conventional switches are replaced by fewer multifunction pushbuttons with computer control. Checkout, display generation, mission management, and autonomous navigation create computer and software requirements far exceed those of Apollo. Multiprocessors, because of low weight and power, appear to achieve the objectives best.

To minimize wire bundle and connector complexities, a digital multiplexed data bus is used for most data transfer. Data bus concepts include uniform interfaces for all subsystems and provide flexibility, low weight, and reliability. Subsystems used for guidance, navigation, and control are conventional, with the choice between sensors made on the basis of development status, accuracy, and probable reliability. Multiple-gimballed inertial measurement units with strapdown instruments for starfield mapping, horizon tracking, and rendezvous laser radar are used.

On-board automatic checkout was examined with particular emphasis on electrical power generation and the EC/LSS subsystems. It was found that once a vehicle is structured with an integrated electronics subsystem containing large digital computers, flexible displays, and a multiplexed data system, the inclusion of on-board checkout will not burden the vehicle with a large number of additional transducers, wire bundles, or special switching networks.

#### 5.6 AERODYNAMICS

Aerodynamic analyses and wind tunnel testing included boost flight drag and air loads estimates as well as the hypersonic, transonic, and subsonic stability and control characteristics of the orbiter and booster elements. A typical example is shown in Figure 18 which presents the FR-3 orbiter hypersonic characteristics. Booster subsonic characteristics are shown in Figure 19. The booster balance situation is more



Figure 18. FR-3 Orbiter Hypersonic Characteristics



Figure 19. FR-3 Booster Subsonic Characteristics

critical than the orbiter due to the aft cg incurred by the 15 engine installation. Since the booster hypersonic flight is all at a high angle of attack to increase drag and to reduce the downrange distance, the aft cg is acceptable. Continued wind tunnel testing, cg surveillance, improved body shaping and empennage optimizing for longitudinal and directional stability across the Mach range must be pursued as the design continues.

The stowable wing was selected for the Phase A final vehicles based on the results of wind tunnel testing of the stowed wing configuration hypersonically and transonically and of the extended wing configuration subsonically. Currently, fixed wing configurations are also being examined at Convair, and these will require further tunnel testing.

#### 5.7 FLIGHT MECHANICS

The configurations lift off vertically with a pitchover into a gravity turn. The launch configurations have a forward cg and are aerodynamically stable. Engine gimbaling

has been selected as the best control method. Ninety-nine percentile Marshall synthetic winds were used for the maximum  $\alpha q$  condition. The cg offset of the FR-3 and engineout conditions were examined. A gimbal angular rate of 10 deg/sec and an angular acceleration of 10 radians/sec<sup>2</sup> are the minimum requirements. The asymmetric arrangement of the FR-3 is countered by the built-in 6.5 degree cant angle in the booster engine array (Figure 7). An excursion of 5 degrees either side of this neutral position is required.

#### 5.8 WEIGHT ANALYSIS

The weight summaries for FR-3 and FR-4 are compared in Figure 20. The mass fraction of the elements, defined as the useful propellant weight divided by the stage gross weight less payload, is also shown.

	FR-3		FR-4	
	BOOSTER	ORBITER	BOOSTER (EA)	ORBITER
AERODYNAMIC SURFACES	82,798	41.084	52,988	46,608
BODY STRUCTURE	191,443	69,156	114,428	82,709
THERMAL PROTECTION	47,388	43,052	36,092	55,361
LAUNCH, RECOVERY. & DOCKING	30, 269	14,020	15,835	15,830
MAIN PROPULSION	141,261	47,539	88,464	49,717
ORIENTATION & SEPARATION	15,740	11,836	10,603	13,304
POWER	5,481	4.270	3,564	4,591
AVIONICS	847	1,100	847	1,100
CREW & CREW SYSTEMS	2,104	2,948	2,104	2,948
CARGO		50,000		50,000
RESIDUALS, LOSSES	26,190	9,253	14,497	10,252
JET FUEL	46,916	2,868	30,711	3,225
MAIN PROPELLANTS	2, 811, 879	629,558	1,508,915	825,351
ELEMENT GROSS WEIGHT (LB)	3, 402, 316	926,684	1,879,048	1,160,996
WEIGHT IN ORBIT		339,578		383,560
ENTRY WEIGHT	564,537	289,655	355,367	327,176
LANDING WEIGHT	517,462	286,787	324,789	322.475
ELEMENT MASS FRACTION	0.825	0.718	0.803	0.743
LAUNCH GROSS WEIGHT (LB)	4, 329, 000		4,919,092	

Figure 20. FR-3 and FR-4 Weight Summaries

#### 5.9 MISSION/PAYLOAD ANALYSIS

Mission requirements and typical space shuttle vehicle mission capability are shown in Figure 1. The Convair space shuttle system is compatible with the missions identified in the NASA space shuttle task group report. A modular payload approach was found desirable and feasible. The study results also showed that:

- a. Four mission-related personnel are required on 94 percent of the missions. A six-man in lieu of a two-man orbiter flight deck would eliminate the need for a small personnel module.
- b. For the logistics mission a 12-man personnel module convertible to a cargo module is desirable. The module should incorporate a universal docking mechanism.
- c. An  $O_2$ -H<sub>2</sub> propulsive stage can be used to deliver payloads to synchronous orbit altitudes, with useful payload weight a function of the mode of operation. Figure 21 shows the deployment of a propulsive stage.



Figure 21. Propulsive Stage Deployment

For propellant delivery, transfer of propellant tanks rather than propellant is preferred. This will minimize losses associated with inorbit fluid transfer and will reduce the number of space shuttle flights required.

The recommended baseline for in-orbit maintenance is a separate pressurized work module. Retrieval of inoperative satellites presents unique problems due to random tumbling. A stabilization and maneuvering system is proposed as a solution.

#### 5.10 MISSION ABORT

Safety and cost require that the crew, passengers, payload, and vehicle be returned intact to the launch site following abort. These requirements are satisfied by:

- a. Booster. The ability to continue to the staging point separate from the orbiter and return to the launch site under these following conditions:
  - 1. Two engines failed at any point in the boost trajectory.
  - 2. Three engines failed in the latter portion of the boost phase.
- b. Orbiter. The ability to separate from the booster and to achieve a once-around the earth trajectory under these conditions:
  - 1. Two engines failed in the booster at any point in the trajectory or up to three failed engines in the latter phase of the boost trajectory.
  - 2. One engine failed in any segment of the orbiter ascent phase or two failed engines in the latter portion of ascent.
- c. Booster and Orbiter. Provisions to suppress potential fire or explosion by isolating fuel and oxidizer with sealed bulkheads and diaphrams and by purging initial compartments with an inert gas to limit oxygen concentration and prevent formation of combustible mixtures.

To satisfy the once around abort philosophy, the orbiter was designed to have sufficient thrust-to-weight at staging with one engine out so that, by burning up the available onorbit maneuver propellants, the orbiter can overcome the additional misalignment losses incurred by the reduced thrust and make a single orbit, enter and return to the launch site. To make one orbit and return to the launch site requires a cross-range capability of 800 n.mi. for the nominal 55-degree orbit.

Current probability goals of 0.999 on intact vehicle return and 0.970 on successful mission completion are established.

#### 5.11 GROUND TURNAROUND OPERATIONS AND FACILITIES

A joint reusable space vehicle ground turnaround analysis was made by Pan American World Airways and Convair to determine the tasks, elapsed time, manhours, and facilities required to process the vehicle from landing to relaunch. Turnaround phases were established to allow for functional categorization of the analysis input/output. The analysis was made for the FR-1 (three-element with crossfeed) vehicle but the results were found to be parametrically applicable to the FR-4 and the FR-3 vehicles. Figure 22 depicts the nine functional phases required for vehicle turnaround and their



location relative to a hypothetical launch complex.

The turnaround times and manhour requirements presented in the following listing represent a summary of reusable space vehicle turnaround requirements based on the detailed analysis. They include all turnaround activities from landing to launch (including routine and non-routine maintenance). The orbiter element, requiring the longest ground time for

Figure 22. Turnaround Phases and Location

maintenance and servicing, is used as the driving factor for elapsed time required for turnaround. Manhours shown are a composite of element/vehicle requirements.

	Elapsed Time		
	(work-hours)	Manhours	
FR-1/FR-4 Vehicles	144.9	9104.7	
FR-3 Vehicle	137.8	6203.7	

Figure 23 is a turnaround flow diagram depicting the times the element/vehicle spends at various functional stop points of the turnaround cycle. The operational facility analysis was also conducted to determine operational requirements of a hypothetical new facility. Cape Kennedy launch facilities were examined, and the potential use of Complex 39 was investigated in some detail. The conclusion was reached that it is technically feasible and economically desirable to use Complex 39 for the space shuttle vehicle. Order of magnitude facility costs are:

- a. New facility for FR-4 configuration, 177 million dollars.
- b. New facility for FR-3 configuration, 155 million dollars.



- c. Modify Complex 39 for FR-4 configuration, 55.6 million dollars.
- d. Modify Complex 39 for FR-3 configuration, 29 million dollars.

Future scheduled launches of other programs using Complex 39 were not considered. If use of existing facilities is considered for the reusable space vehicle program, a com-

Figure 23. Turnaround Cycle Elapsed Time

prehensive study of the effects of joint occupancy and conduct of simultaneous launch programs should be made.

#### 5.12 DEVELOPMENT PROGRAM

The development program for the space shuttle is shown in Figure 24. The combined



Figure 24, Program Summary Schedule

C/D phase is assumed to begin in the second quarter of 1971 and continue 66 months to the first operational fl ght. The figure shows the key development milestones. This development program reflects the critical timing necessary to support a mid-1976 target operational date. The initial operational date is considered as the earliest likely date and with some development risk involved.

The flight test program, shown in Figure 25, uses six flight test elements (three orbiters,

three boosters). Two orbiters and two boosters are carried over into the operational program, leaving a complete launch configuration vehicle for continued R&D tests or operational backup testing. The flight test program consists of two basic flight test phases: horizontal or aircraft-type tests, and vertical or launch-vehicle-type tests. Single-element vertical launches precede the all-up vehicle launch configuration flights and serve to explore, in progressive increments, the limits of the velocity/altitude envelope attainable by a single element. The four multi-element flights serve to demonstrate stage separation, boost phase abort procedures, vehicle entry, and attainment of horizontal flight configuration and on-orbit maneuvers and operations.



Figure 25. Flight Test Program Approach

facility such as Edwards AFB could be used. Total program costs for the FR-3 and FR-4 vehicles, based on 50 launches per year for 10 years, are:

The test facility approach is to take maximum advantage of existing facilities within the government/industry complex. Most of the major ground test facilities required are available at MSFC or MSC, with some modifications required. The vertical launch flight test facility requirements are satisfied by the initial operational launch site and supporting facilities. However, if a suitable runway and service facility are not available in time for the horizontal flight tests, a

	<u>FR-3</u>	$\underline{\mathbf{FR}}$ -4
Development	\$5.23	\$4.88B
Investment	0.49	0.69
Operations	1.15	1.39
	\$6.87	\$6.96B

These costs are based on 1969 dollars and Convair cost estimating relationships.

The development program costs are shown in Figure 26. The high costs associated with airframe ground test and flight test are due primarily to hardware costs.

	COST	5 (\$M)
	FR-3	FR-4
DEVELOPMENT		
AIRFRAME	984	942
PROPULSION	557	527
AVIONICS	79	79
AGE	254	243
GROUND TEST	1,267	1,098
FLIGHT TEST	1,384	1,331
FACILITIES	224	248
SE &I	452	385
TOTAL	5,201	4,853

#### Figure 26. Development Program Costs

#### 5.13 MAJOR SYSTEM SENSITIVITIES

Some of the FR-3 systems sensitivities, linearized about the design point, are summarized in Figure 27. The effect of designing a 22-foot-diameter payload bay envelope over 30 feet of the existing 15-foot diameter by 60-foot long payload bay were examined. The resulting system has a gross liftoff weight of 4.63 million pounds, or 300K pounds heavier than the 15-foot diameter payload baseline. The total system inert weight increases by 46,000 pounds.

5.14 CONCLUSIONS		MAJOR DEPENDENT PARAMETER					
•••		VARIED PARAMETER	∆GLOW	ATOTAL DRY WT	A PAYLOAD .	<b>∆ COST</b>	
The Pha	major conclusions of the se A study are:	ΔI <sub>SP</sub> ORBITER ΔI <sub>SP</sub> BOOSTER ΔINERT WT ORBITER ΔINERT WT BOOSTER	18000 LB/SEC 16750 LB/SEC 28.5 LB/LB 5.23 LB/LB	2070 LB/SEC 2040 LB/SEC 3.31 LB/LB 1.597 LB/LB	979 LB/SEC** 1184 LB/SEC** -1.0 LB/LB -0.188 LB/LB	12.2M\$/SEC 11.1M\$/SEC 20K\$/LB 3K\$/LB	
a.	Fully reusable space shuttle concepts that will achieve an order	ΔCONTINGENCY ORBITER ΔCONTINGENCY BOOSTER Δ ΔV MANEUVER FLYBACK L/D	55,350 LB/% 23,060 LB/% 553 LB/FPS 35,600 LB/UNIT L/D	6530 LB/4 2622 LB/4 63.3 LB/FP8 4510 LB/UNIT L/D	 	 375K\$/FP8 	
of magnitude reduction in total logistics sys- tem recurring costs		4.0 g BOOST LIMIT +7360 LB •FIXED GLOW ••CONSTANT FLOW RATE					

Figure 27. FR-3 System Sensitivities

- b. A high degree of safety can be achieved through redundancy and system design methods comparable to commercial aircraft.
- c. The FR-2 concept maximizes payload potential with current system requirements.
- d. Many technology areas must be pursued to ensure realization of performance, cost and schedule.

#### 6 STUDY LIMITATIONS

are feasible.

The study limitations were primarily those imposed by the study guidelines, which defined many of the vehicle characteristics (Section 4). Depth of the study was limited by the funds available. For example, tests to verify design assumptions and aerodynamic, aerothermodynamic, and dynamics characteristics of the final vehicles were beyond the scope of the study. These limitations, however did not inhibit meeting the study objectives or reflect on the validity of the study results.

#### 7 IMPLICATIONS FOR RESEARCH

The space shuttle system described in this report encompasses the functions of a booster, a spacecraft, and an airplane. As such, all vehicle concepts considered face technological problems in the areas of aerodynamics, aerothermodynamics,

flight dynamics, structure and propulsion. None of the major areas requiring research require 'breakthroughs', but are logical extensions of technologies already in hand. The following identifies research requirements in several important areas.

- a. <u>Aerodynamic Configuration Definition</u>. A systematic aerodynamic analysis and experimental program must be conducted to define the vehicle configuration to be developed in the hardware phase. Work started in the pre-development phase will continue into the hardware development phase.
- b. <u>Aerodynamic Heating Analysis and Wind Tunnel Tests</u>. These tests will better define the aerothermodynamic environment and the resulting heat transfer rates on the space transportation system during launch and entry, for selection of the TPS materials, TPS sizing, and configuration development.
- c. <u>Radiative Thermal Protection Systems</u>. Experiments and studies must be conducted to develop a nose cap capable of sustaining temperatures up to 4000°F and vehicle leading edge systems capable of temperatures up to 3100°F. Concepts and materials for investigation include actively cooled structures, transpiration cooled structures, and passive structures of materials such as diborides, hypereutectic carbides, graphites, and refractories. The performance and reusability of the lower surface radiative TPS must be demonstrated and fabrication techniques must be defined.
- d. <u>Ablative TPS</u>. Although the objective for the space transportation system vehicles is radiative reusable TPS, ablatives could provide a backup because the materials and techniques for ablative systems are well developed.
- e. <u>Main Vehicle Structure</u>. Since the main vehicle structure (including the propellant tanks) is the major vehicle weight item, this area must receive concerted attention to establish the validity of predicted structural design approaches and the predicted structural mass fractions. This can be accomplished by designing and fabricating with the proposed fabrication techniques a number of full size primary structure components to be tested under their simulated thermal, mechanical, and functional conditions to verify the structural sizing procedures and estimates of non-optimum factors used to estimate vehicle weights.
- f. <u>Composite Material Applications</u>. The use of composite materials using boron fibers in epoxy and metal matrix offers a potential for significant weight reductions in high stress areas and should be investigated.
- g.  $\underline{LO_2/LH_2}$  Engine Development. The  $\underline{LO_2/LH_2}$  rocket engine analyses and design studies must be accelerated, and component and assembly testing of engines carried out at the earliest possible date.
- h. <u>Attitude Control Rocket Subsystem</u>. Attitude control requirements are beyond the capability of existing control engines. This requires technology advancement to develop long life, reusable engines in the 1200 to 1500-pound thrust range.

- i. <u>Flight Control Subsystem</u>. The technology program for the flight control subsystem does not require a flight-test phase, but all control characteristics should be verified through evaluation techniques such as six-degree-of-freedom simulations based on wind tunnel aerodynamic data of the vehicles.
- j. <u>All-Weather Automatic Landing Subsystem</u>. A fully automatic landing subsystem is desirable to permit fully automatic approach and landings under all conditions and may reduce or eliminate the need for a go-around capability if the reliability is sufficiently high. The development schedule for commercial applications appears compatible with the development of the space shuttle.
- k. Integrated Electronics Subsystem. The many functions to be performed by the space transportation system and the relatively complex equipment on this vehicle make it desirable, if not mandatory, to use a completely integrated electronics system.

#### 8 SUGGESTED ADDITIONAL EFFORT

Many technical problems encountered during the study were beyond the scope of the effort. Solutions require a more detailed definition of design criteria, additional and more detailed design studies, additional technical analyses, and/or research. Many of the problem areas, such as staging dynamics, are concept oriented, and real solutions can be provided only after concept definition is relatively firm. Other areas, such as design criteria for structure, and TPS are fundamental to all concepts and must be resolved early in any future program. The selection of these design criteria will greatly influence the ultimate inert weight, safety, and reliability and the subsequent development and operational costs of the space shuttle system.

Areas that required additional effort beyond the scope of the study were identified for NASA as the study progressed. Currently, within NASA, plans are being formulated for a Phase B definition study and for implementing the research requirements. These plans, if pursued, will provide the additional effort required to resolve the identified problem areas.

#### REFERENCES

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