AD 700 436

THE HUMMINGBIRD XV-4B VTOL RESEARCH AIRCRAFT PROGRAM

Lockheed Georgia Company
Marietta, Georgia

October 1969

AFFDL-TR-69-75

THE HUMMINGBIRD XV-4B VTOL RESEARCH AIRCRAFT PROGRAM

Lockheed-Georgia Company

A Division of Lockheed Aircraft Corporation
Marietta, Georgia

TECHNICAL REPORT AFFDL-TR-69-75

OCTOBER 1969

This document has been approved for public release and sale; its distribution is unlimited.

AIR FORCE FLIGHT DYNAMICS LABORATORY
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

37

NOTICE

When Government drawings, specifications, or other data are used for any purpose other than in connection with a definitely related Government procurement operation, the United States Government thereby incurs no responsibility nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data, is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission to manufacture, use, or sell any patented invention that may in any way be related thereto.

Copies of this report should not be returned unless return is required by security considerations, contractual obligations, or notice on a specific document.

200 - December 1969 - CO455 - 103-2262

THE HUMMINGBIRD XV-4B VTOL RESEARCH AIRCRAFT PROGRAM

FOREWORD

On 21 September 1966, the Lockheed-Georgia Company, Marietta, Georgia, entered into a contract with the Air Force to provide the necessary engineering, manufacturing and ground test to modify a government-bailed XV-4A into an XV-4B direct jet lift VTOL aircraft, and to conduct a flight test program to demonstrate the capability of the configuration to meet its specified performance. Well into the flight test program on 14 March 1969, the aircraft was lost on a routine test mission near Dobbins Air Force Base. This report recounts the significant developments, results and conclusions of the program spanning this interval of time, and is submitted to fulfill the requirements of Contract F33615-67-C-1035, Data Item B037.

The program was administered under the direction of the V/STOL Technology Division (FDV) of the Air Force Flight Dynamics Laboratory. Mr. Randall B. Lowry was the Program Manager.

Management of the program at Lockheed-Georgia was provided by Mr. John G. McReynolds, Project Manager. While a list of names of those individuals who made significant contributions would be too extensive to present here, acknowledgement is made to all those who participated in the program for their contribution to the many technological achievements.

Manuscript of this report was submitted by Lockheed-Georgia Company on 1 July 1969.

This technical report has been reviewed and is approved.

Jay D. Pinson
Lt. Col. USAF
Chief, V/STOL Technology Division
Air Force Flight Dynamics Laboratory

ABSTRACT

The XV-4B VTOL aircraft was developed to fulfill the vehicle requirements of Part IV of the VTOL Integrated Flight Control System (VIFCS) program conducted by the Air Force Flight Dynamics Laboratory. The requirements developed for the XV-4B in this role basically called for a direct jet lift aircraft with the capability of vertical take-off and landing, hovering, conventional flight and transition between these flight modes, with a flight control system compatible with a Variable Stability System (VSS) to be developed in Part I of the VIFCS program. The performance capabilities included guarantees in the areas of hover time, control powers, and thrust-to-weight (T/W) ratios in order to assure the aircraft would be capable of performing the basic mission of handling qualities and control system criteria investigations.

The XV-4B is a modification of an earlier research aircraft, the XV-4A, developed by Lockheed for the U. S. Army. Only part of the aft fuselage and empennage of the XV-4A were finally salvaged for the XV-4B; the remainder of the aircraft was designed and manufactured anew. The XV-4A propulsion system was replaced with 6 turbo-jet engines arranged to provide direct lift thrust for VTOL operations, with two of the engines nacelle mounted and capable of being diverted to a cruise-thrusting mode.

Programs carried out in support of the aircraft development program included:

- Wind Tunnel Tests

  A 16% scale-model of the XV-4B was built with vertical thrust producers to simulate engines. This model was used in two series of wind tunnel tests at Langley to investigate VTOL and transition characteristics, and in the University of Maryland wind tunnel to investigate "deep-stall" characteristics.

- Inlet Development Tests

  A full-scale model of the lift engine section of the XV-4B was equipped with YJ-85 engines and tested to develop the lift engine inlet configurations.

- o Cyclic Tests

  A test rig with the XV-4B propulsion and reaction control system installed was used to develop propulsion system hardware components in a 35 hour test program.

- o Flight Test Program

  A 50-hour flight test program to verify VTOL, transition and conventional flight envelopes was planned. At the time of loss of the aircraft, 23 flights of a planned total of 95 had been made, exploring all of the 4 phases of flight for the aircraft.

- o Inverted Telescope and Balance System

  In order to check VTOL operations in the safety of a captive flight device, an Inverted Telescope was developed with the capability of raising the aircraft to various ground plane heights up to about 15 feet, with freedom to make attitude changes while operating the engines and aircraft systems in simulated flight. A force balance system was also developed for this device to measure lift and control forces and moments. This device was used for measuring guaranteed T/W and control power performance.

- o Escape System Tests

  A series of static and dynamic ejection seat firings were conducted for the purpose of qualifying a zero-zero escape capability for the XV-4B.

Testing completed at the time of loss of the aircraft indicated that the XV-4B would have met its design specification requirements. Control Powers and T/W ratios had exceeded the guarantee values. Development problems in the areas of tire and lower fuselage skin temperatures, noise environment, and engine reingestion stalls had been encountered in ground test preparatory to VTO operations; however, fixes had been incorporated or planned that would have surmounted these problems. Satisfactory stability and handling qualities were demonstrated throughout the envelope tested.

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Cont'd)

LIST OF ILLUSTRATIONS (Cont'd)

LIST OF TABLES

SECTION I

INTRODUCTION

This report presents the results of a program undertaken by Lockheed-Georgia Company to develop a direct jet lift VTOL aircraft, designated the XV-4B. Development of this vehicle was a part of the VTOL Integrated Flight Control Systems program conducted by the Air Force Flight Dynamics Laboratory (AFFDL) to establish handling qualities and flight control systems design criteria for VTOL aircraft. Part IV of this program called for development of a vehicle with VTOL capabilities, including extended hover time, compatible with a Variable Stability System (VSS) to be developed under another part of the VIFCS program.

After working with AFFDL technical planning personnel at Wright-Patterson Air Force Base late in 1965 and through the early months of 1966 the basic definition and scope of the XV-4B program were established. These efforts resulted in the issuance of USAF Request for Proposal RTD 266766-KNA, dated 7 June 1966. In response to this request, the XV-4B Program Plan and associated Statement of Work were submitted in Lockheed-Georgia Company Report ETP 693, dated 5 July 1966.

On 21 September 1966 Letter of Contract F33615-67-C-1035 was entered into by the Government and Lockheed-Georgia Company for development of the XV-4B, including the following tasks:

- o Engineering and manufacturing required to modify one (1) government-bailed XV-4A aircraft into the XV-4B configuration.
- o An Inlet Development Test Program to establish and verify the design of the lift engine air inlets.
- o A Cyclic Test Program to qualify VTOL propulsive and reaction control systems.
- o Pneumatic tests to permit evaluation of the performance of the reaction control system as installed in the airplane.
- o Escape System Tests to demonstrate compatibility between the aircraft and the Escapac 1D-1 escape system.
- o Ground and Flight Tests to investigate handling characteristics and demonstrate airworthiness of the vehicle to perform its intended mission.

- Design and manufacture of an Inverted Telescope to be used as a captive flight device throughout the test program.

- Design and development of a Balance System for the Inverted Telescope to be used in determining six-component aircraft data at heights from static ground position to a height of fifteen feet.

- A small scale Wind Tunnel Test Program.

- Demonstration of performance guarantees (hover time, thrust-to-weight ratios and control powers).

As development progressed, the contract was modified to incorporate several program changes, most significant of which were addition of provisions for procurement and fabrication of parts for one additional XV-4B, and additional development and testing of the Escape System.

The initial program spanned over a 21 month period from September 1966 to June 1968. Numerous development problems resulted in several contractual changes to increase the program span. At the time of the loss of the aircraft on 14 March 1969, the program was scheduled for completion in June of 1969, one year later than originally anticipated. Descriptions of these development problems are included along with the descriptions of the aircraft and its systems and the associated test programs in the appropriate sections of the report that follows. Conclusions derived from this design, development and test experience are presented in the final section to complete the report.

SECTION II

AIRFRAME DESIGN

1. INTRODUCTION

This section presents a general description of the aircraft with particular definition of the airframe and structure. Section II in combination with Section III, Aircraft Systems, comprises a definitive description of the complete aircraft. In addition, the Appendix provides a complete history of the structural fatigue damage experienced during the test program and relates this to established inspection requirements.

2. GENERAL DESCRIPTION

The XV-4B is a two-place mid-wing monoplane designed with the capability of vertical take-off and landing in conjunction with conventional flight operations. Propulsive power for conventional flight is provided by two nacelle mounted General Electric YJ85-19 turbojet engines. In the vertical take-off and landing (VTOL) mode, the thrust of the two conventional flight propulsion engines can be diverted downward to combine with the thrust of four additional vertically mounted General Electric YJ85-19 turbojet engines to provide the lift required for flight operations at speeds below conventional wing stall speeds down to free hover. The general arrangement of the XV-4B configuration is shown in Figure 1. Close resemblance to the XV-4A external configuration was retained in order to minimize the unknown configuration effects on conventional flight characteristics as well as to permit maximum utilization of XV-4A structural components. Distinctive external features of the XV-4B are the four exposed lift engine inlets in the top of the center fuselage, retractable engine exhaust doors in the lower center fuselage, relocation of the cruise engine nacelles to a position approximately 20 inches further forward on the fuselage than those of the XV-4A, larger wing tip pods to contain larger reaction control valves and similarly, local refairing at the forward and aft extremities of the fuselage to accommodate the changes in reaction controls at these positions. Prominent features retained from the XV-4A configuration include the dual side-by-side crew position arrangement, the "T" tail arrangement, and the retractable tricycle landing gear arrangement characterized by a slightly offset nose gear position.

a. Interior Arrangement

The interior arrangement of the XV-4B is shown on the inboard profile drawing appearing in Figure 2. Extending forward of the nose of the aircraft is the boom on

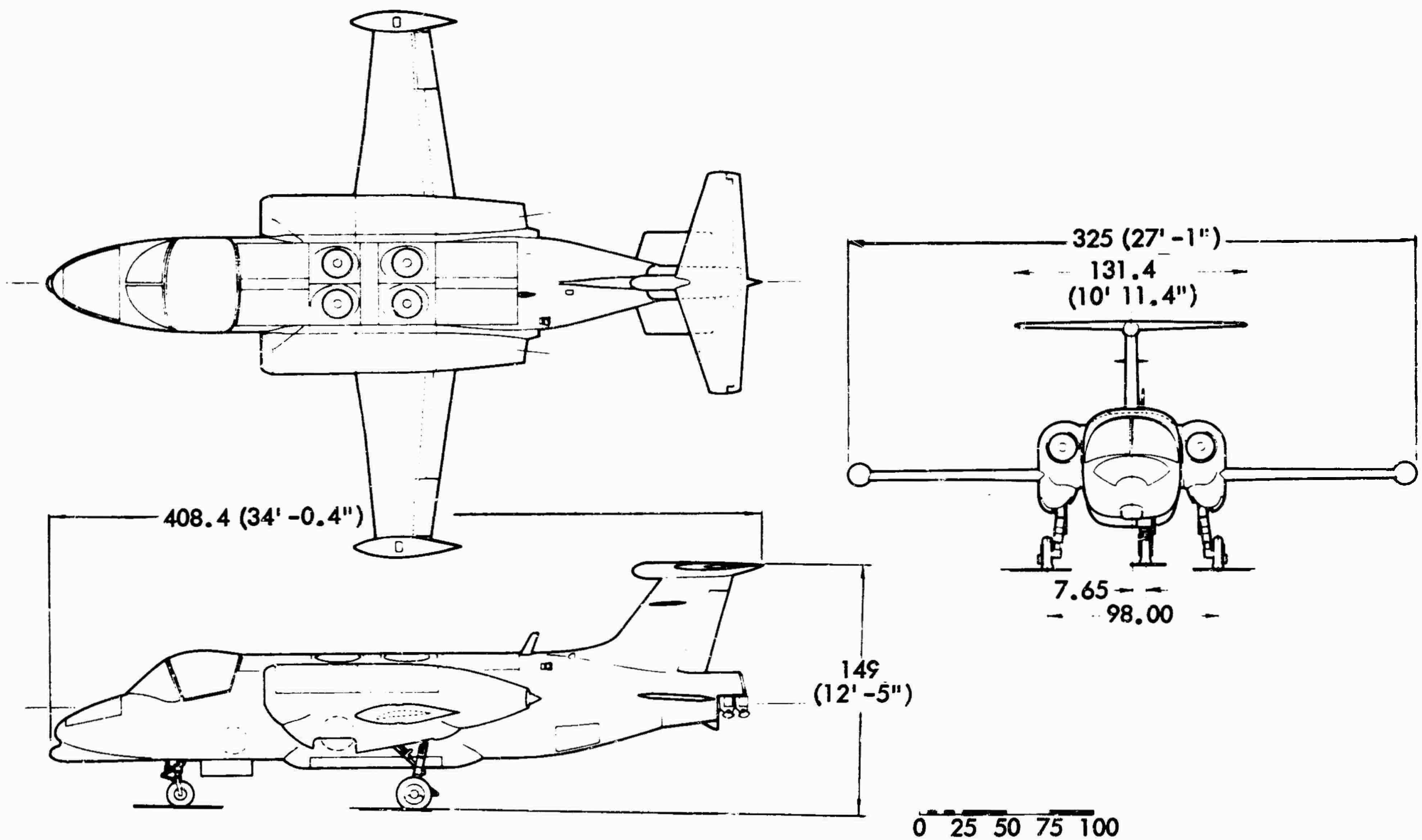


FIGURE 1 - GENERAL ARRANGEMENT

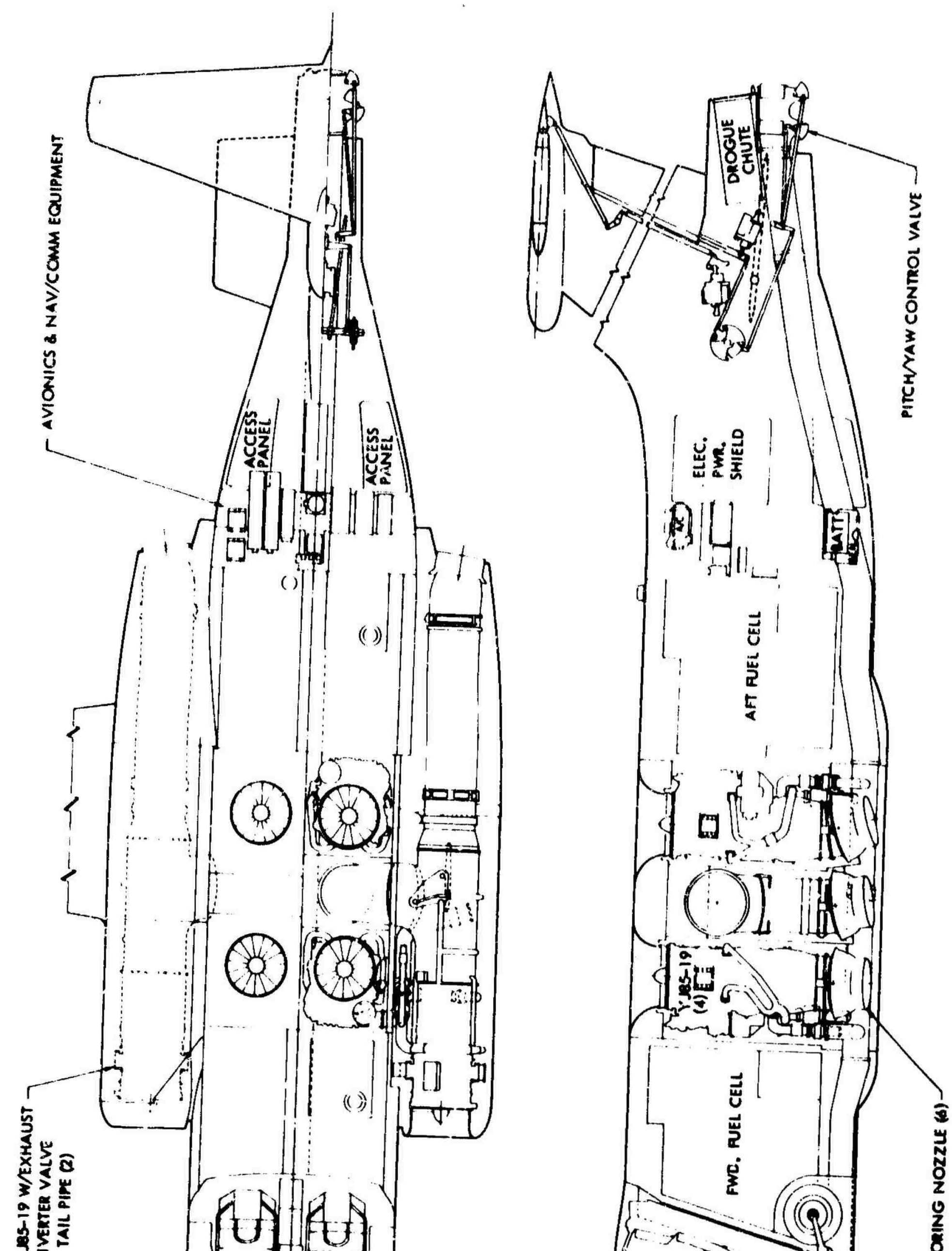


FIGURE 2 - INBOARD PROFILE

which airspeed, angle of attack, and angle of sideslip sensors are mounted. Below the boom in the forwardmost part of the aircraft is the forward pitch reaction control valve, with the ground air supply connection, used primarily for ground starting of the engines, located immediately aft of the pitch valve. Above the pitch valve, in the forward fuselage equipment compartment much of the flight control system input components such as feel springs, trim actuators, gradient change actuators, and the hydro-mechanical engage/disengage clutches for the manual system are contained. Relay panels are also located in this compartment, which is accessible through a large removable panel (see Figure 1). Behind this area is the flight station with side-by-side crew positions featuring a dual, sequenced ejection system for the crew. The right-hand position is designated the primary, or safety pilot position; the left-hand position is for the alternate, or evaluation pilot. Each pilot is provided with identical controls with normal stick and adjustable rudder pedals, and a throttle console containing separate throttle levers for each cruise engine and a collective lever for the four lift engines located conveniently for left-hand operation by each pilot. In addition, a lift lever is located aft of the throttle quadrant for the left-hand pilot as an alternate throttle control for the four lift engines. This lift lever was installed for pilot evaluation purposes and can be engaged or stowed very simply at the discretion of the pilot seated in the left-hand position. The area below the cockpit floor is occupied primarily by the nose landing gear and the air supply duct for the forward pitch reaction control valve.

Aft of the cockpit, separated by a double walled bulkhead, is the forward fuel tank. An aft fuel tank is provided at a point aft of the aircraft center of gravity to balance the fuel load. The two approximately equal-volume tanks provide a total fuel capacity of 740 U.S. gallons. Each tank is of integral construction with a filler point and a maintenance access door at the top and a surge box in the lower aft corner. The area above each fuel tank contains much of the subsystem equipment such as air conditioning supply ducts, hydraulic reservoirs, accumulators, selector valves, oxygen equipment, and related plumbing lines. The area between the two fuel tanks houses primarily the vertical lift propulsion system; engine exhaust elbow ducts extend inward and down from the cruise engine diverter valves, with the vertically mounted lift engines fore and aft of the lift elbow ducts. The compartment above the lift elbow ducts contains the basic fire extinguishing system and the engine pressure ratio transmitters. At the bottom of this lift system compartment are the exhaust vectoring nozzles and the main center manifold of the reaction control bleed air duct system.

Immediately behind the aft fuel tank is the aft equipment compartment, accessible by large door panels located on each side of the lower aft fuselage. This compartment contains most of the large sub-system equipment items such as primary flight control system rate gyros and computers, the air conditioning unit, inverters, battery and remotely located radio equipment. The compartment also provides the space for most of the aircraft payload such as flight test instrumentation and recording equipment. The extreme aft portion of the aircraft fuselage contains the drogue chute system including the container and release/jettison mechanism, and the aft pitch/yaw reaction control valve.

b. Dimensional Data

Basic airplane overall dimensions are given on the general arrangement shown in Figure 1. Additional data of general interest are presented here; a more complete presentation of general aircraft dimensional data is included in Reference 1.

(1) Wing (Theoretical)

Area	104.17 sq. ft.
Span	25.0 ft.
Root Chord	72.0 in.
Tip Chord	28.0 in.
Mean Aerodynamic Chord (MAC)	53.23 in.
Aspect Ratio	6.0
Incidence:	
Root	1.5°
Tip	-1.5°
Airfoil Section	
Root	NACA $64_{A}012$
Tip	NACA $64_{A}212$
Leading Edge Sweepback Angle	$8^{\circ}20'$

(2) Horizontal Stabilizer (Theoretical)

Area	26.44 sq. ft.
Span	10.67 ft.

Root Chord	42.5 in.
Tip Chord	17.0 in.
Mean Aerodynamic Chord	31.57 in.
Aspect Ratio	4.30
Incidence	0°
Airfoil Section	
Root	NACA 0010-2.00-40/1.575
Tip	NACA 0010-2.00-40/1.575
Leading Edge Sweepback Angle	17° 41'
Arm	192.63 in.
(Distance from Wing MAC 10% Chord to Horiz. Stab. MAC 25% Chord)	

(3) Vertical Stabilizer (Theoretical)

Area	27.5 sq. ft.
Span	6.08 ft.
Root Chord	71.5 in.
Tip Chord	36.85 in.
Mean Aerodynamic Chord	56.02 in.
Aspect Ratio	1.3485
Airfoil Section	
Root	NACA 64_A012
Tip	NACA 64_A012
Leading Edge Sweepback Angle	36° 45'
Arm	156.86 in.
(Distance from Wing MAC 10% Chord to Vert. Stab. MAC 25% Chord)	

c. Weight and Balance

The following table shows a comparison of the XV-4B Group Weight Statement for the actual airplane with that of the specification airplane. The actual weights by group represent either a calculated weight, results of component weighing, or in the most common case, a combination of both. The Predicted Weight Empty is based on a simple summation of these group weights. The Actual Weight Empty is determined from actual weighing of the total aircraft as it existed in March 1969. The difference between the Predicted and Actual Weight Empty is then assumed to be unaccounted for manufacturing variations.

During the course of the aircraft configuration development a number of design changes occurred that had the effect of increasing the aircraft weight empty. A change in the contract was negotiated to increase the VTOL Gross Weight by 220 pounds to help offset these increases. Although no change in the specification weight empty was made, this increase ultimately has to appear in the weight empty. Therefore, for the purpose of this report, this negotiated weight increase is shown added to the specification weight empty.

Group Weight Statement

Group	Specification Weight, lbs.	Actual Weight, lbs.
Wing	395	417
Tail	167	198
Body	1,274	1,204
Landing Gear	389	443
Surface Controls	655	764
Nacelle	333	188
Propulsion	3,096	3,232
Instruments & Navigational Equipment	133	167
Hydraulic & Pneumatic	116	186
Electrical	394	469
Electronics	35	20
Furnishings & Equipment	391	380
Air Conditioning Equipment	58	53
Auxiliary Gear	27	25
Predicted Weight Empty	7,463	7,746
Manufacturing Variations		43
Specification Revision	220	
Actual Weight Empty	7,683	7,789

Figure 3 presents an overview of the aircraft Weight History from design inception through development test. The target weight line shown was based on reaching the original specification weight at a point midway into the flight test program as initially scheduled, with an allowance for growth during the design and development phases. As indicated estimated weight was substantially over target weight in the initial design phase, however, actual weight was reduced well within target by 90% design release. A gradual increase as anticipated was experienced throughout development test. The actual aircraft weight empty in March 1969 exceeded the increased specification weight by 106 pounds.

The aircraft center of gravity for the Actual Weight Empty of 7,789 pounds is at F.S. 280.0, or 9.1% MAC. The aerodynamic center-of-gravity limits of 4% MAC forward to 11% aft shall not be exceeded for any aircraft flight condition. Ballast tables for various crew weights and fuel weight/C.G. tables are included with the detail weights data presented in Reference 2.

The following table presents a comparison of the useful load combinations with the Actual Weight Empty for the specification airplane and the aircraft in the test configuration as it was operated in the flight test program at Lockheed-Georgia Company. The test configuration was normally flown with one pilot, with payload being in the form of flight test instrumentation.

VTOL Design Gross Weight

Summary

		Specification Configuration		Flight Test Configuration
Weight Empty		7,683		7,789
Operating Equipment		1,122		860
Crew & Seat Pack	430		215	
Payload (Test Equipment)	600		(515)	
Oil	62		64	
Unusable Fuel	30		66	
Oxygen			3	
Usable Fuel		3,995		4,151
Design VTOL Gross Weight		12,800		12,800

WEIGHT HISTORY MODEL XV-4B

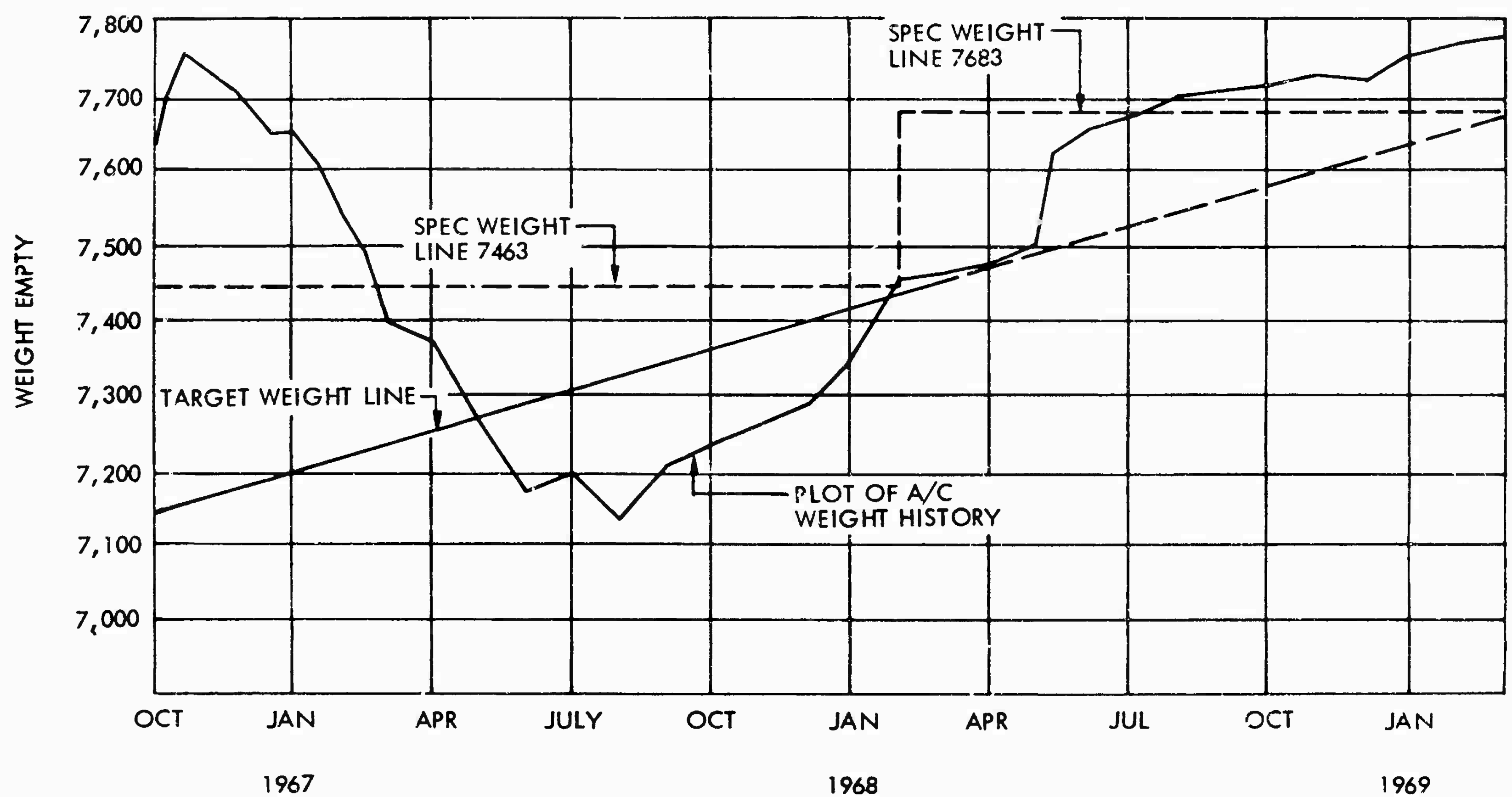


FIGURE 3 - XV-4B WEIGHT HISTORY

Complete group weight statements for Specification design weights can be found in Reference 1. Group weight statements for the actual aircraft are presented in Reference 2.

3. AIRFRAME STRUCTURAL DESCRIPTION

The airframe structure consists of the wing with its associated control surfaces, the empennage with associated control surfaces, the fuselage, and nacelles. The empennage and part of the aft fuselage are used directly from the XV-4A aircraft bailed to Lockheed-Georgia Company for modification under this contract. Both of these assemblies required modification to be compatible with sub-system changes in the XV-4B configuration. All other airframe structure was completely re-configured, essentially within the basic external contours retained from the XV-4A, and manufactured anew to meet the XV-4B configuration requirements. Dimensional information related to these airframe structure assemblies is included in Section 2, Volume I of Reference 3.

a. Wing

Each outer wing is fully cantilevered and attached to the body structure at the outboard face of the nacelle by means of a bolted shear splice. The wing is of all metal construction consisting of a basic box beam structure, leading edges, flaps, ailerons and streamlined pods containing lateral reaction controls at the wing tips. No fuel is carried in the wings.

The wing box is an aluminum structure consisting of a front spar at 16% chord and a rear spar at 60% chord; upper and lower covers of chem-milled tapered skins stiffened by similarly tapered hat-section stringers; ribs appropriately placed to react flap and aileron hinge loads and maintain contour in the skin panels, with cutout provisions for accepting the reaction control bleed air supply ducting; and the inboard attachment fittings which accept the shear-type attachment bolts along the surfaces and spars for connecting the wing to the aircraft.

The leading edge consists of an inboard and outboard section, with separation between the two sections occurring approximately midway along the exposed wing. The outboard leading edge assembly is of conventional aluminum skin and rib construction, permanently assembled to the 16% spar of the wing box. Cutouts are provided in the ribs for installing reaction control bleed air supply ducting. The inboard leading edge assembly is of similar construction, but made readily removable by means of screw attachments at the 16% spar and at the ends of the assembly to provide ready access for reaction control bleed air supply ducting inspection, removal and replacement. The ribs for the inboard leading edge assembly are made of 17-7 PH stainless steel because of higher strength requirements brought on by provisions for ready removal.

The trailing edge flaps are the single-slotted type, mounted on external hinges below the lower surface, using anti-friction bearings. The flap is of single spar, aluminum alloy construction. The ailerons are of similar construction and are statically and dynamically balanced. A pressure-resistant fabric seal is installed between the aileron leading edge and the wing structure for maximum effectiveness at low speeds. Connected directly to the outboard end of the aileron is the drive linkage for the roll reaction control valves, which are contained in streamlined aluminum tip pod fairings. The tip pod fairings are fixed structure permanently attached to the outer rib of the wing box, with removable access panels provided for servicing of the reaction control valve and linkage.

b. Empennage

The empennage of the XV-4B is a "T" arrangement, with the horizontal surface mounted atop the vertical. The empennage assembly is attached to the aft fuselage structure by means of a continuous pattern of tension bolt connections around the base of the vertical stabilizer. The empennage is of all metal construction consisting of a vertical and horizontal stabilizer structure joined together at a screw-connected, but not readily separable, joint where the surfaces come together; interconnected elevator surfaces which are hinged at the 80% horizontal stabilizer chord line, and which contain a separately hinged and actuated manual trim tab; a rudder which is hinged about the 75% chord of the vertical stabilizer; and a bullet-type fairing at the juncture of the horizontal and vertical stabilizer.

The vertical stabilizer is an all aluminum structure with a rear spar at the 62.5% chord, a structural leading edge acting as a front spar, hat-section stiffened skin panels, and formed ribs appropriately placed to react rudder hinge loads and maintain contours. The horizontal stabilizer is basically a two-spar structure with front spar at 20% chord, aft spar at 68% chord, hat-section stringer stiffened skin panels, and ribs spaced as required. Since the horizontal stabilizer structure is retained from the XV-4A, it includes the provisions for boundary layer flow control which is not a requirement for the XV-4B.

The elevator and rudder are of aluminum single spar construction, with rubbing contact seals for improving control effectiveness. The elevator has a separate trim tab, of aluminum single spar construction, driven by an electrical input actuator. The rudder has a similar trim tab which was a requirement for the XV-4A; since this tab was not required on the XV-4B, the tab actuator was replaced with a fixed link.

The vertical-horizontal stabilizer joint is covered with a streamlined bullet fairing of aluminum construction, made removable to gain access to the elevator control linkage.

c. <u>Fuselage</u>

The fuselage assembly consists of three distinct but structurally integral sections: forward, center and aft sections.

The forward fuselage section is considered to be that portion of the fuselage extending forward of the canted bulkhead to which the crew seat tracks are affixed. This all-aluminum structure is basically the same as that of the XV-4A, built around cockpit provisions for a crew of two. The nose wheel well and the forward pitch reaction control bleed air supply duct compartment below the cockpit form the basic beam structure around which the forward fuselage is designed. The opening for the cockpit canopy and windshield, and the large access panel forward of this area render the upper portion of the forward fuselage essentially non-structural. The cockpit floor is extended the full structural length of the forward fuselage section to provide lateral strength and stiffness.

Crew compartment closure is completed by a large transparent plexi-glass windshield and laterally hinged one-piece canopy.

The center fuselage section is considered to be that portion of the fuselage between the canted bulkhead, which becomes the forward boundary of the forward fuel tank, and the aft bulkhead of the aft fuel tank. The integral fuel tanks are of double-walled construction; the inner walls of the tanks are made of integrally-stiffened skin panels chem-milled to precise thickness requirements, and the outer walls become the external skins of the aircraft, with closely spaced formers between the two walls. The lift engine compartment contained between the two fuel tanks, separated by double walls, consists of six separated compartments for the lift engines and lift tailpipes from the cruise engines. This all-titanium-and-steel structure includes the center wing box which carries the wing loads across the fuselage through the area of the lift engine cavities. The area above these engine compartments contain the lift engine inlets. The areas above the fuel tanks are covered by removable panels to provide access to the large number of functional components installed in these areas.

The all-aluminum aft fuselage structure, retained essentially from the XV-4A, is of conventional longitudinally stiffened skin construction with supporting frames spaced as required. A large, removable structural access door is provided in the lower quarter on either side to gain access to the large amount of equipment installed in this area.

d. Nacelle

The nacelles, located above the wings and adjacent to the fuselage, house the cruise engines, accessories, diverter valves, horizontal thrust nozzles and the main landing gear. They comprise the necessary mounting structure and the engine cowling, including access panels. The cowling is of conventional, stiffened skin, aluminum alloy construction. The nacelle compartment is completely isolated from the fuel tank areas by titanium firewalls. The lower portion of the nacelle forward of the wing carry-through structure is separated from the engine compartment by a titanium structure to form the main landing gear wheel well. This area is left open externally to simplify the main landing gear system.

4. STRUCTURAL DESIGN

This section summarizes the design criteria used for the XV-4B structures and results of the substantiation analyses used to verify that the design satisfied the required criteria. References 4, 5, 6, and 7 contain the detail information upon which these summaries are based.

a. Structural Design Criteria

The basic XV-4B structural design crteria are derived on the basis of the MIL-A-8860 (ASG) Series Military Specifications, with additional criteria as necessary to provide for operational areas not specifically covered by the military specifications. In some instances, where specific MIL-A-8860 (ASG) Series requirements were considered to be too severe, imposing redesign and weight penalties, and to modify it did not compromise the basic XV-4B mission, exceptions were imposed.

The criteria selected were chosen to satisfy four primary objectives:

- o To provide adequate structural integrity.
- o To maintain as much of the existing XV-4A structure as possible.
- o To provide capability for performance of the desired mission.
- o To permit simplified load analyses which produce adequate strength requirements.

In general, the structural strength resulting from application of these criteria are equivalent to that resulting from an across-the-board application of the Military Specifications, but the load analysis requirements are considerably less.

Fatigue-resistant design principles were incorporated to provide structure resistant to sonic fatigue, or to load-induced structural fatigue, for a service life goal of 500 hours.

All criteria unless specifically noted otherwise resulted in limit loads which were multiplied by 1.5 to obtain ultimate loads. Specific exceptions were design landing loads and crash load factors.

(1) Weight and Center of Gravity Limits--Design weights and center-of-gravity (C.G.) limits used as design criteria for developing aircraft loads data are the same as those presented in paragraph 2. For convenience these are repeated below in summary form.

Minimum Operating Weight	8,185 pounds
Design Gross Weight	12,000 pounds
Design VTO Weight	12,800 pounds
Design Ramp Weight	13,100 pounds
Forward C.G. Structural Limit	3.5% Mean Aerodynamic Chord
Aft C.G. Structural Limit	12.5% Mean Aerodynamic Chord

(2) Structural Design Airspeeds--The XV-4B is designed, in the clean configuration, for all attainable airspeeds within the envelope bounded by 20,000 foot altitude, Mach Number (M) = .68, and 410 Knots Equivalent Airspeed (KEAS). Pertinent structural design airspeeds are defined in the following summary.

V_H/M_H	- Maximum Structural Design Level Flight Speed	350 KEAS as limited by M = 0.53
V_L/M_L	- Maximum Structural Design Limit Airspeed	410 KEAS as limited by M = 0.68
V_{L_F}	- Maximum Flap Design Airspeed Maximum Landing Gear Operating Speed Maximum Lift Engine Operating Speed*	240 KEAS as limited by M = 0.53
V_{Demo}	- Maximum Demonstration Speed	260 KEAS as limited by M = 0.53

*For airspeeds up to 240 KEAS with lift engines operating, the effect of reaction control forces shall be combined with the aerodynamic design loads.

(3) Limit Flight Load Factors--The XV-4B is designed for gust load factors that result from a 66 fps gust at speeds up to V_H/M_H and 50 fps gust at speeds between V_H/M_H and V_L/M_L. Maneuver load factors used in design are as follows:

	Clean	Flaps Down
Symmetrical Flight Positive	3.0	2.0
Symmetrical Flight Negative	1.5	0.0
Rolling Maneuver Positive	2.4	1.6

(4) Flutter Criteria--The aircraft structural components are designed to be free from divergence, flutter, and other aeroelastic instability at all speeds up to 115% of the structural design limit airspeed (V_L) for the design ranges of altitudes, maneuvers, and loading conditions.

(5) Control Surface/System--The rudder, aileron, elevator and reaction control system are designed to withstand the loads imposed by the pilot(s) and the Stability Augmentation System during flight. The systems will withstand the following loads, applied by the pilot(s) at the top of the stick grip or at the point of foot contact with rudder pedals:

	Elevator	Rudder	Aileron
Each of Two Pilots (aiding or opposing)	150 lbs	225 lbs	75 lbs
Single Pilot	200 lbs	300 lbs	100 lbs

These loads are considered to be reacted:

- o by the control system stops only
- o by components specifically supplied for reacting pilot-applied loads
- o by all applicable portions of the specific control system under consideration, assuming that any part of the system is jammed and that the particular power system is inoperative, or operative.

Design loads are based on control movements and corresponding control surface movements as follows:

Rudder	$\pm$ 20 degrees
Rudder Pedal Travel	$\pm$ 3.25 inches
Elevator - Up to 260 KEAS	$\pm 30^{\circ}$
Elevator - Above 260 KEAS	$\pm 20^{\circ}$ *

(*For operations at speeds greater than 260 KEAS, provisions must be made to limit the control deflections).

Elevator Control travel	$\pm$ 3.75 in.
Ailerons	$\pm 20^{\circ}$
Aileron Control travel	$\pm$ 3.75 in.
Wing Flap	40 degrees total

The maximum no-load surface travel rates in the VTOL flight mode, as limited by actuator capability are:

Elevator	221 Deg/sec
Rudder	147 Deg/sec
Aileron	147 Deg/sec

Secondary control systems, including cranks, wheels, levers, are designed to withstand the loads shown below:

Control	Limit Applied Load
Push-pull cranks, wheels or levers	50 pounds
Twist Operated wheel or knob	153 inch-pounds
Push-pull knob	100 pounds

The primary flight control surfaces are designed to the hinge moments resulting from the deflections attainable considering the maximum output of the powered control system. The control surface tabs are conservatively designed to full deflection at limit speed. Control surfaces are designed for inertia loads, acting parallel to the hinge line, of:

24 x Weight of Surface for Vertical Surfaces
12 x Weight of Surface for Horizontal Surfaces

The wing flaps are designed to the loads resulting from full flap deflection of 40° at the maximum flap operating speed of 240 KEAS. The load factors are, alternately, 0.0 g and 2.0 g.

(6) Landing Gear Criteria--The landing gear loads developed during landings are considered to be design landing loads. The cumulative effects of elastic, permanent, and thermal deformations resulting from application of these loads will not interfere with the mechanical operation of the landing gears or adversely affect the aerodynamic characteristics of the airplane. The reactions to the gear loads furnished by the gear back-up structure are considered to be limit loads. The gear loads developed during all phases of operation except landing are considered to be limit loads. A summary of the landing and ground handling design parameters is shown in the following table.

LANDING AND GROUND HANDLING PARAMETERS

CONDITIONS	GROSS WT. RANGE POUNDS	MAXIMUM SINKING SPEEDS Ft/Sec	FWD SPEED RANGE KNOTS	LIFT/WEIGHT
VTOL LANDINGS	8,185 12,800	13	-	0.83
STOL AND CONVENTIONAL	8,185 12,000	10	UP TO 200 KEAS	1.00
GROUND HANDLING	8,185 13,100	0	-	-

The airplane is considered to land at any of the above design sinking speeds at aircraft attitudes listed below:

LANDING ATTITUDES

ATTITUDE	DESCRIPTION
THREE POINT	BOTH MAIN AND NOSE WHEELS IN CONTRACT WITH GROUND - WINGS LEVEL
TWO POINT LEVEL	BOTH MAIN WHEELS IN CONTACT WITH GROUND, WITH NOSE WHEEL JUST CLEAR AND NOT CARRYING LOAD THROUGHOUT THE LANDING.
TAIL DOWN	BOTH MAIN WHEELS IN CONTACT WITH GROUND WITH THE AIRCRAFT AT AN ANGLE OF PITCH OF 12.5° (STALL ANGLE FLAPS DOWN).

Spin-up and spring-back loads shall be those developed when the airplane lands on surfaces that develop a sliding friction coefficient of 0.55 between the tire and surface and any lesser values of sliding friction coefficient that are critical.

For ground handling design conditions refer to Reference 4. The following arbitrary landing conditions apply.

(a) Main Gear Conditions--The airplane shall be in the level attitude with only the main gear wheels on the ground. The vertical reaction shall be equal to the maximum vertical gear reaction obtained in the two-point symmetrical landing at a sink speed of 10 feet/second. A load of one-half of the vertical reaction shall be concurrently applied at the ground in any direction in the x-y plane. These loads shall be reacted by aircraft inertia.

(b) Nose Gear Conditions--The airplane shall be in the three-point landing attitude with all landing gears contacting the ground simultaneously.

- The vertical load resulting from a conventional landing at a sink speed of 10 feet/second shall be combined with a load applied at the ground in any direction in the x-y plane equal to one-half of the vertical reaction. The loads shall be reacted by aircraft inertia.
- The vertical load resulting from a VTOL landing at a sink speed of 13 feet/second shall be combined with a load applied at the ground in any direction in the x-y plane of one-fourth of the vertical load. The loads shall be reacted by aircraft inertia.

(7) Drogue Chute--The drogue chute system and fuselage attachment structure is designed for loads occurring at airspeeds up to 260 KEAS. A shear connection limits loads build-up beyond 1.15 x limit load. Drogue chute load is considered to be applied in a cone of 60° included angle symmetrical about the x-axis.

(8) Jacking Loads--Jacking loads are specified in the following table. The vertical loads are considered to act singly and in combination with the longitudinal loads, the lateral loads, and both longitudinal and lateral loads The horizontal loads at the jack points are reacted by inertia forces so as to cause ι ..ıange in the vertical loads at the jack points.

JACKING LOADS

COMPONENT	LANDING GEAR 3-POINT ATTITUDE	OTHER JACK POINTS LEVEL ATTITUDE
VERTICAL	1.35 F	2.0 F
LONGITUDINAL	0.4 F	0.5 F
LATERAL	0.4 F	0.5 F

F IS THE STATIC VERTICAL REACTION AT THE JACK POINT

The applicable weights for jacking are as follows:

Jack Combinations	Maximum Jacking Weight
Fwd Fuselage and Main Landing Gear	Design Ramp Weight
Fwd Fuselage and Wing	Minimum Operating Weight

(9) Crash Loads--The loads and loading conditions specified herein are ultimate and are applicable to the design of crew seats, mechanisms for holding doors in their open position, attachments of equipment items, ballast, payload, engines, fuel tanks, and their carry-through structure. Fuel tanks are considered to contain one-half of their fuel capacity. The following ultimate load factors, acting separately, apply for crash design conditions.

Longitudinal	9.0 g Forward; 1.5 g aft
Lateral	1.5 g to right and to left
Vertical	4.5 g Down; 1.0 g Up.

b. Basic Loads

A detailed derivation and analysis of the XV-4B external loads are presented in Reference 5 and are summarized in this section. The loads are based on the application of the Structural Design Criteria summarized previously and presented in detail in Reference 4.

In general, the basic loads calculations are made using airplane aerodynamic data based on direct force and moment measurements of a 16% scale model of the XV-4B. These data were measured in wind tunnel tests conducted at NASA Langley in early 1967 and 1968 and at the University of Maryland in early 1968. All of these data are "power-off"; no adjustments are made to the loads analyses to account for the effects of thrust.

Because of the unavailability of wind tunnel pressure data for the wing, fuselage, and nacelles, the component aerodynamic force and moment data were derived from theoretical pressure distributions. Theoretical pressure distributions were also used for the derivation of the unit airloads for the wing, fuselage, nacelles, and horizontal tail in the same manner as was done in the XV-4A loads' analysis. Although the amount of pressure data available for the vertical tail and the aft fuselage strakes was limited, these data were used for determining the unit airloads for these particular components.

For all load conditions the airplane was placed in equilibrium by the application of the appropriate inertia forces. A quasi-static maneuver analysis was used for all flight conditions, and a dynamic analysis was used for all landing conditions. A dynamic analysis was made for gust and maneuver conditions, but the resulting loads were not ctitical for design. The following degrees of freedom of airplane motion are considered for the maneuver analyses:

(1) Pitching Maneuvers - two degrees of freedom (pitch and vertical translation).

(2) Sideslip Maneuvers - two degrees of freedom (yaw and lateral translation).

(3) Roll Maneuvers - one degree of freedom.

Considerations of the control system characteristics were made in all of the maneuver analyses.

The net design loads were determined as the algebraic sum of the critical aerodynamic and inertia loads resulting from investigation of flight, landing, and ground handling conditions. Loads were generated for the following airplane components:

- Wing
- Flaps
- Horizontal Tail
- Vertical Tail
- Fuselage
- Nacelles
- Landing Gear
- Engines
- Control Surfaces
- Control System
- Canopy
- Wing Tip Pods
- Landing Gear Doors
- Elevator Tabs
- Fuselage Strakes

The critical design conditions which are found to design the aircraft are contained in Table I.

c Structural Analysis and Substantiation

The stress analysis used as substantiation for the structural integrity of the XV-4B is contained in Reference 7. It is intended to present basic information for the major structure; such as control systems, wing boxes, fuselage stringer-panels, empennage control surfaces, and engine mounts. Also included are some analyses of detail fittings

TABLE I

CRITICAL CONDITIONS

COND.	DESCRIPTION	GROSS WEIGHT	SPEED	LOAD FACTOR	C.G. LOCATION	SINK RATE
1	Balanced Maneuver - Clean	12000	410K	3.0	3.5%	
5	Accelerated Roll - Flaps Down	12000	240K	1.6	3.5%	
16	Balanced Maneuver - Clean	12000	300K	3.0	3.5%	
32	Accelerated Roll - Clean	12000	350K	2.4	3.5%	
34	Accelerated Roll - Clean	12000	410K	2.4	3.5%	
10.18	Dynamic Landing - 12 1/2° T.D.	8185	200K	3.4	4.6%	10 fps
RCHK1-P	Rudder Kick - Parachute	12000	260K	1.0	12.5%	
NPC1-P	Negative Check Pitch - Parachute	12000	300K	-1.8	3.5%	
4002	Max. Vert. Conv. - Fwd. Drag	12000	200K	3.3	12.5%	10 fps
4003	Max. Vert. Conv. - Side Load	12000	200K	3.3	12.5%	10 fps
122	Hover - Max.	8185	0	1.0	All	
236	Dynamic Landing	8185	0	5.5	7.3%	13 fps
252	Dynamic Landing	8185	200K	4.2	7.3%	
Deep Stall						
Max. Output	Diverter Valve Act. @ 3000 psi					
Jam	Control Servos @ 3000 psi					
9G Crash	Forward Crash Load					
Max. Pres.	Maximum Engine Exhaust					
PMC9-2	Positive Checked Pitch	12000	288K	3.0	12.5%	

and localized special structure as a sample of the type stress analysis contained in work sheets which are available as back-up information should a specific detail item require further investigation. Stress analyses for both landing gears are contained in References 8 and 9 "Main and Nose Gear Stress Analysis, XV-4B" Reports which were prepared by Loud Co. (now Howmet Corporation), Pomona, California.

Similarity between the XV-4A and the XV-4B structure plus certain flight restrictions provide the justification for flight testing without major static structural testing, a decision which was made during the contract negotiation phase.

Operational tests on the bleed air duct system, reaction control valves, diverter valve system, and control system were performed as proof of their adequacy and made in addition to the stress analysis. A control system proof test was also made on the aileron, rudder, elevator and flap controls. Other tests which were carried out include landing gear drop tests, tire tests, bleed air duct leak and burst tests, and miscellaneous vendor item static and burst tests.

Allowables are calculated using standard approved methods, most of which are found in the "Military Handbook -5A", Lockheed's Engineering Stress Memo Manual; Roark - "Formulas for Stress and Strain"; Bruhn - "Analysis and Design of Flight Vehicle Structures"; and TN 2661, "A Summary of Diagonal Tension". An ultimate of 1.5 is used, and "B" probability value material allowables are used where they are available; otherwise, "A" values or specification values are used. Internal loads and panel shear flows for the wing and fuselage are calculated using an IBM RAX 360 system computer.

The XV-4A parts including the horizontal and vertical stabilizers, the aft fuselage between fuselage stations 387 and 448, the elevator and rudder surfaces used in the XV-4B were modified to improve their strength in the critical areas where load increases dictates a change. The forward fuselage and outer wing maintained the basic design concepts of the XV-4A but required new assemblies because the changes were of such a significant magnitude to preclude use of the XV-4A hardware. Modifications were made to the canopy as a result of Escape System tests; the windshield and its support structure are identical to the XV-4A.

Areas where the design is new include the center wing, center fuselage, landing gears, power plant installations, nacelle, hydraulic system, equipment installations, and the control system.

A summary of the margins of safety are included in Table II. It covers the control mechanisms, outer wing, center wing, forward fuselage, aft fuselage, propulsion and empennage. The critical load conditions for the structure are defined in Table I.

5. OPERATIONAL EXPERIENCE/FATIGUE DAMAGE AND INSPECTION

Several areas of the XV-4B structure are particularly susceptible to dynamic loads such as acoustic and vibratory loading. Structural and equipment components sensitive to these types of loadings were identified by extensive testing during the cyclic test and aircraft ground test programs. A summary of the structural fatigue damage experience and Inspection Plans derived during these test programs is included in the Appendix.

TABLE II

SUMMARY OF MARGINS OF SAFETY (M.S.)

DESCRIPTION	COND.	M.S.	DESCRIPTION	COND.	M.S.
Control Mech. - Aileron					
Member 88 Bellcrank	Jam	.16	W.S. 77.0 Elem 4	1	.05
Member 46 Bellcrank	Jam	.14	W.S. 90.0 Elem 11	1	.00
Member 61 Bellcrank	Jam	.03	W.S. 103.0 Elem 11	1	.08
Member 89 Rod End	Jam	.00	W.S. 108.0 Elem 11	1	.06
Member 7 Rod	Jam	.00	W.S. 117.35 Elem 11	34	.00
			W.S. 128.0 Elem 11	34	.05
Control Mech. - Rudder	Jam		W.S. 138.0 Elem 11	34	.06
Member 16 Bellcrank	Jam	.00	Rear Spar 90.0/103.0	5/32	.20
Member 41 Rod End	Jam	.16	Front Spar 90.0/103.0	1	.25
Member 10 Rod	Jam	.46	Spar Cap to Surface Attach.	16	.13
			W.S. 83.5 Front Spar Ftg.	1	.11
Control Mech - Elevator	Jam		W.S. 52 Front Spar Ftg.	Telescope	.05
Member 15 Bellcrank	Jam	.09	W.S. 65 Flap Hinge Ftg.	Flap	.66
Member 7 Rod End	Jam	.03	W.S. 123 L.E. Rib	1	.34
Member 48 Rod	Jam	.12	Center Wing		
Control Mech - Flap	Jam		Rear Spar Upper Cap	4003	.03
Member 22 Bellcrank	Jam	.08	MLG Drag Brace Ftg.	4002	.06
Member 2 Rod End	Jam	.11	MLG Outboard Trunnion Ftg.	4003	.39
Member 4 Rod	Jam	.04	W.S. 24 Elem 8	1	.18
			W.S. 28.3 Elem 8	1	.05
Outer Wing			W.S. 36.9 Elem 8	1	.07
W.S. 65.0 Elem 6	34	-.01	W.S. 46.4 Elem 9	1	-.07
			W.S. 51.6 Elem 10	1	.12

TABLE II (Cont'd)

DESCRIPTION	COND.	M.S.	DESCRIPTION	COND.	M.S.
Forward Fuselage			Empennage - Vert. Stab.		
Fuel Tank Wall	9G Crash	.02	VSS 68.0 Elem 9	PMC9-2	-.03
Under Tank Beam Cap	9G Crash	.05			
Under Tank Beam Attach.	9G Crash	.28			
Aft Fuselage					
F.S. 366.65 Elem 17	10.18	.04			
F.S. 377.35 Elem 14	RCHK1-P	.60			
F.S. 387.35 Elem 18	RCHK1-P	.05			
F.S. 399.50 Elem 9	NPC1-P	.15			
F.S. 411.75 Elem 16	RCHK1-P	.10			
F.S. 424.0 Elem 16	RCHK1-P	.21			
F.S. 436.25 Elem 20	RCHK1-P	.09			
Strake					
BL 16.7 L.E. Rib	Deep Stall	.03			
Front Spar	Deep Stall	.06			
Front Spar Splice	Deep Stall	.10			
Propulsion - Diverter Valve					
Lower Mount	252/236/122	.30			
Actuator Mech. Brg.	Max. Output	.02			
Actuator Mech. Bolt	Max. Output	.08			
Bellcrank Shear Pin	Max. Output	.12			
Bellcrank	Max. Output	.07			
Door Shaft	Max. Pres.	.00			

SECTION III

AIRCRAFT SYSTEMS

1. INTRODUCTION

This section presents the major systems as installed in the XV-4B. Systems discussed include: Flight Controls, Propulsion and related subsystems, Electrical, COM/NAV, Flight Station, Hydraulic, Landing Gear and Escape System. This section combined with Section II, Airframe Design, provides a complete description of the XV-4B aircraft. More detailed descriptions of the aircraft systems are contained in Reference 3.

2. FLIGHT CONTROL SYSTEM

a. Description

The XV-4B Primary Flight Control System (PFCS) is a hybrid fly-by-wire arrangement that includes an integral Stability Augmentation System (SAS) to provide augmented rate damping about all three airplane axes throughout the flight envelope. The system is designed to be compatible with the installation and interface requirements of a Variable Stability System (VSS). The "hybrid fly-by-wire" terminology is used to describe the system since a conventional mechanical system is provided as a back-up for the normal fly-by-wire mode of operation.

The airplane moment producing elements of the system are conventional aerodynamic control surfaces and engine compressor bleed air powered reaction control valves that operate on a demand basis. The control surfaces and their respective reaction control valves are mechanically interconnected and mechanically functional at all times. In VTOL flight the reaction control system is pressurized and the control moments are the sum of the contributions of the reaction controls and aerodynamic surfaces. In conventional flight the bleed air system is depressurized and control is accomplished through the conventional aerodynamic surfaces only.

Dual reaction control valves are provided on each axis to provide a fail operative capability. The design is such that the failure of a single valve on a given axis will not affect the mechanical operation of the remaining valves and the pilot retains approximately one-half normal control in one direction and full normal control in the opposite direction. Differential valving of bleed air through upward and downward pointing nozzles at each wing tip provides lateral reaction control. Directional reaction control is provided by differential opening and closing of sideward pointing nozzles at the aft end of the fuselage and longitudinal control is effected through opening and closing downward pointing nozzles located at the fore and aft ends of the fuselage.

In the normal fly-by-wire mode of operation the aerodynamic surfaces and reaction control valves are positioned by electro-hydraulic power actuators that respond to electrical pilot command and SAS signals. The pilot force commands are reacted by feel springs and sensed with force transducers located in the control column stick grips and in the rudder pedal linkage. Pilot trim commands are accomplished through trim circuitry that operates trim actuators to relieve the force generated by the feel springs and at the same time provide an electrical signal equal to that removed from the force sensor command path by reducing the control forces to zero. The SAS signals are generated by rate gyros that provide electrical outputs proportional to angular rates about the airplane body axes. The pilot command and SAS signals are summed in the computer and these signals are then the input to the electro-hydraulic servo-valves of the hydraulic power actuators.

In the normal mode of operation the pilot's cockpit controls are mechanically disengaged from those portions of the system driven by the power actuators so that the response of the system to SAS damping signals will not be fed back to the pilot's controls. The isolation of the pilot from the SAS inputs is accomplished through hydro-mechanical clutching devices located in the forward fuselage area. On disengaging the fly-by-wire system these clutches automatically engage to provide a direct mechanical path from the pilot's controls to the airplane's aerodynamic control surfaces and reaction control valves. In addition these clutches automatically engage in the event of total failure of either the electrical power or hydraulic power supply systems but remain disengaged in the event of a single failure of either power source or a multiple failure of one electrical and one hydraulic system.

The system includes gradient change actuators to provide for a change in pilot command force gradients between the VTOL and conventional flight regimes. The actuator is controlled by a flight mode selector switch that energizes the actuator to change the moment arm between the pilot's input force and the feel spring thereby changing the feel gradient. The flight mode selector switch also changes the gains in the electronics portions of the system so that full surface and valve displacement corresponds to full pilot control displacement for both the VTOL and conventional flight values of force gradient. In addition it changes SAS gains from high values for VTOL flight to lower values for conventional flight.

The PFCS utilizes, except for certain portions of the trim circuitry, triply redundant electronic components and sensors, and employs a voting stage to guard against faulty signals in a branch of the triple redundant channels. The voter is an intermediate signal select and/or gate which accepts the three branch signals as its input and selects the median signal as its output. This voting process guards against faulty signals being sent to the servo-valves after a single fault upstream of the voter. If a faulty signal does exist the error detection logic of the system triggers a switch in the select monitor circuit which illuminates a fault light for the channel. This fault information is presented to the pilot while the system continues to function with undegraded performance.

Each of the electro-hydraulic power actuators have three servo valves that receive the median signal selected by the upstream voter. Two of these valves actually drive the main ram utilizing the hydraulic pressure provided by dual independent hydraulic pressure sources. The third valve serves as a model. The model has the hydraulic logic to detect a faulty signal from a servo valve and when an error is detected the model trips the error indicating mechanism which illuminates a fault light for the channel. The model also trips the faulty bypass and cutoff valve which shuts down hydraulic pressure to the faulty valve and bypasses the main ram piston supplied by this pressure. Thus the actuator is protected against any single fault between the upstream electronic voter and the main ram of the actuator itself.

As noted above two independent hydraulic pressure sources provide the hydraulic power for the actuators. In addition, two independent electrical sources provide electrical power for the electronics. These dual sources of power coupled with the redundant characteristics of the electronics, sensors, and actuators provide a fail operative system that continues to function with undegraded performance following any single failure. If a failure does occur the pilot is made aware of the condition through the fault monitoring features of the system.

Figure 4 is a schematic diagram that depicts the major components of one axis of the flight control system. More detailed diagrams and a more detailed description of the system are presented in Reference 3.

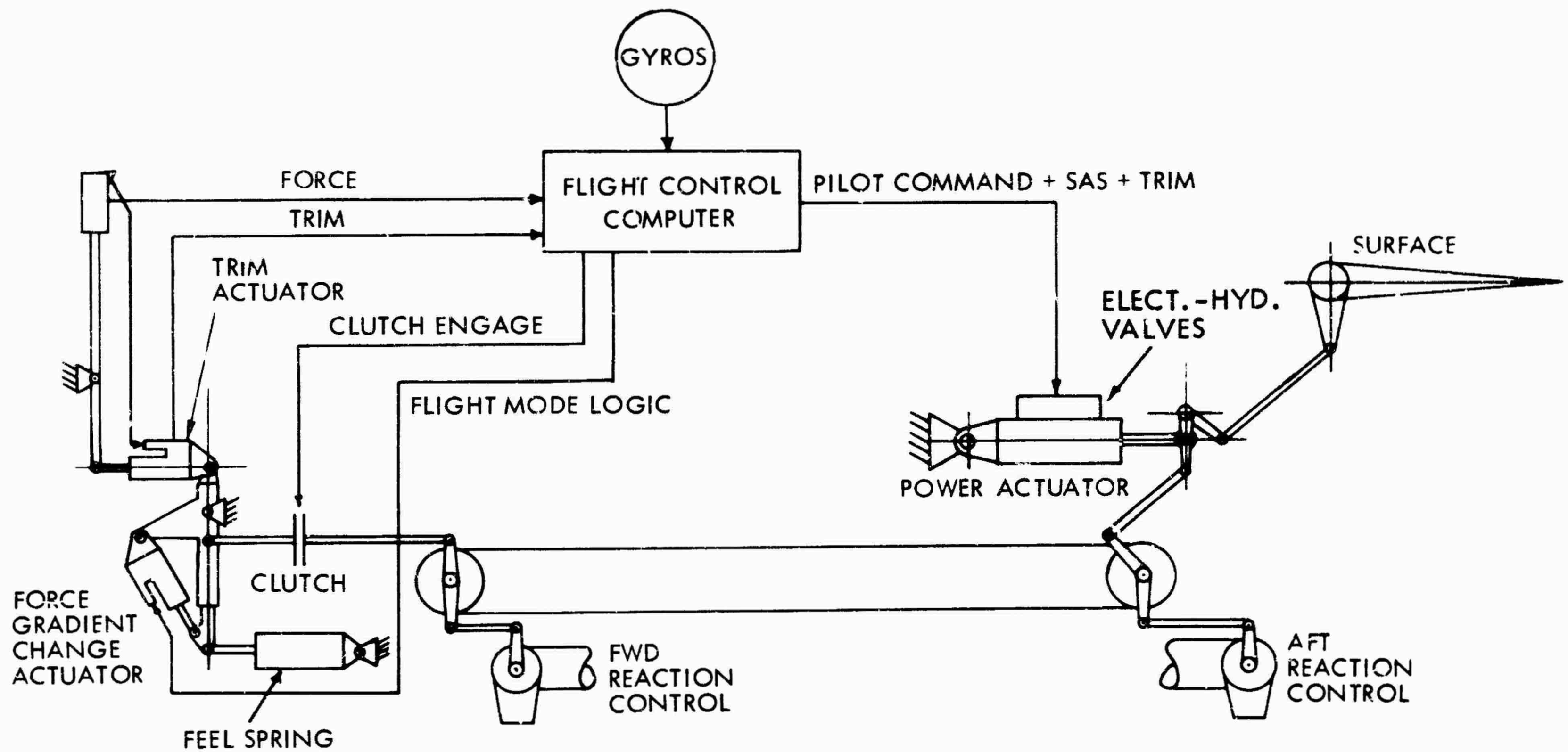


FIGURE 4 - PRIMARY FLIGHT CONTROL SYSTEM SCHEMATIC

b. System Development

(1) Design Evolution--The PFCS briefly described above is the same as that originally conceived and specified in Reference 1 with the following major exceptions:

- The system has dual reaction control valves as opposed to the original single valves.
- The system incorporates hydromechanical clutches and feel springs to isolate the pilot from SAS activity and to provide pilot feel forces, respectively. The original concept utilized a series servo and electronic circuitry to provide isolation and pilot feel.
- The rudder system includes a centering spring over and above the feel spring mentioned above. The original concept depended on the series servo and electronic circuitry to provide centering.
- The roll and pitch axes have electromechanical trim actuators to relieve the feel spring forces. The original concept did not require these actuators since the series servo and power actuator were simultaneously positioned by trim signals.
- The yaw axis trim system does not have a trim actuator and the rudder pedals do not reflect the trim deflection of the rudder and yaw valves. The original concept provided for pilot control displacement proportional to commanded trim.
- The system incorporates first order lag filters to attenuate power actuator amplitudes at the higher frequencies. The original concept did not include these filters although the potential requirement for them was recognized.

The above mentioned items are discussed in more detail below.

Early in the program the decision was made to incorporate dual reaction control valves on each axis. This decision was based on the recognition of the potentially catastrophic results of losing or jamming a valve and was in keeping with the redundant design philosophy generally applied to the remainder of the system. As noted previously the dual valves provide a fail operative capability in that with the failure of a single valve control in one direction is still one-half the normal value and full control is retained in the opposite direction.

The change in the control system to one employing hydromechanical clutches and feel springs from one utilizing series servos and electronic circuitry to provide the same functions occurred as the detailed design of the mechanical portions of the control system developed. As noted earlier the series servo was designed to isolate the pilot's commands from the power actuator activity and provide artificial feel. To accomplish this the input signals to the servo included:

- Rate gyro signal - provided to mask stability augmentation activity from the pilot's controls.
- Force command signal - provided for control force versus control deflection characteristics.
- Servo position follow-up signal - provided for servo stability.
- Trim command signal - provided for control deflection versus control surface position characteristics.

Although this concept was evaluated in a simulation environment and found to function well the following servo problems led to the decision to modify the system:

- The isolation servos were found to be structurally inadequate to withstand the design loads for the mechanical linkages of the system.
- It was determined analytically that the force supplied by the center-latching spring was insufficient and too easily overridden by pilot commands.
- The servo was supplied by one hydraulic system and commanded by a single electronic signal path, thus, a single failure could cause the servo to fail and centerlatch. This action would allow the power actuator activity due to the SAS to be reflected to the pilot's controls. Degradation of the PFCS would result and this was contrary to the basic fail-operational philosophy applied throughout the rest of the system.

A new servo would have been required to correct the above deficiencies. Since such a servo was not an off-the-shelf item and because the lead time to procure one was prohibitive, the clutch/feel spring arrangement was chosen to replace the servo system. This arrangement was selected because it involved no technology voids, was dually redundant electrically, hydraulically, and mechanically (dual feel springs), and because, upon failure or disengagement of the fly-by-wire system, it would positively engage and remain engaged with the mechanical backup system.

As a consequence of modifying the system to the clutch/feel spring arrangement it was also necessary to modify the trim system to provide a means of relieving the feel spring force resulting from control displacement. This was accomplished by adding electro-mechanical trim actuators to reposition the feel spring neutral reference as a function of trim command inputs. Initially this was implemented identically on all axes but in the subsequent modification of the rudder system described below the rudder trim system was also modified.

The modification of the rudder system occurred after the airplane had been constructed and tests of the system revealed erratic trim and poor centering characteristics. Both of these deficiencies were attributable to high friction in the mechanical portions of the rudder system between the pilot's pedals and the force transducer. To function properly the original trim system was dependent on friction levels between the force transducer and pedals being no greater than the force equivalent to the electrical breakout of the force transducer and the preload of the feel spring. Under these circumstances the only signal generated to position the rudder to the commanded trim position would be that from the sensor measuring the trim command and the pilot's controls and the rudder would both be displaced an amount equivalent to the trim input. With friction levels greater than the equivalent breakout of the force transducer the trim system functioned improperly since the friction constrained the movement of the linkage containing the transducer and as a consequence trim commands caused a force transducer output as well as a trim sensor output. These outputs tended to cancel one another since the sense of the transducer output was opposite to that and summed with the trim sensor output. This caused erratic behavior of the trim system since the friction forces were not constant. It also caused the pilot controls not to reflect the surface position under all conditions of trim. The obvious solution of redesigning this portion of the system to reduce friction was considered but discarded because of program schedule and cost constraints. In lieu of this the trim actuator was removed and replaced with a fixed link and the trim sensors that originally measured the actuator extension were moved to the cockpit and positioned so that the pilot could manually generate a trim position command from the sensors. This arrangement proved satisfactory although the pilot's controls no longer reflected the surface position resulting from trim inputs.

The rudder centering problem was resolved by installing a centering spring between the pilot's rudder linkage and the force transducer. The spring breakout force was selected to just overcome the friction forces in the system. Although the introduction of the spring increased the breakout forces somewhat, and also increased the force gradient, it did correct the centering problem and the increased forces were acceptable to the pilot.

Prior to first flight, structural vibration ground tests showed that the frequencies of certain structural vibrational modes were such that the possibility existed that these modes might be excited by high amplitude control inputs at the higher control input frequencies associated with the high gain SAS. To preclude the possibility of control system/structural coupling first order lag filters were installed in the electronic circuitry of the rate gyro feedback loops of all axes of the PFCS. Subsequent flight testing showed these filters had no adverse effect on the pitch and yaw control systems but that the phase lag caused by the filter on the roll axis could have been a contributing factor to the low amplitude roll oscillation encountered at high speeds at high SAS gain settings.

(2) Simulation Program--Digital computation, analog computation, and pilot simulation techniques were utilized to confirm the PFCS concept and to choose system parameters. The digital analysis used three-degree-of-freedom linearized small perturbation equations to analyze the dynamic aircraft characteristics and determine stability augmentation requirements. The analog analysis used simplified six-degree-of-freedom equations together with control system mechanization to analyze aircraft and system responses to basic disturbances. Finally, piloted simulation utilized a fixed-base cockpit mockup together with the analog aircraft and control system model to evaluate the impact of system parameters and system failures on real-time handling qualities.

Because of limitations in the simulation mechanization and the absence of visual and motion cues, no attempt was made to analyze the overall handling qualities of the aircraft quantitatively. Instead, emphasis was placed primarily on evaluating the adequacy of the PFCS and the establishment of trends of performance for changes in design parameters. This qualitative assessment established the PFCS parameters to be used in flight test and confirmed that the VTOL transition maneuver could be performed in a reasonable manner.

The results of the piloted simulation validated the fly-by-wire and rate-only stability augmentation system concept and revealed that the isolation servo was unacceptable. The handling qualities resulting from a failure of the isolation servo were deemed unacceptable, and the servo was replaced by the redundant clutch/feel spring system which disconnects the fly-by-wire system from the mechanical backup system in the normal mode of operation. Details of the PFCS analysis can be found in References 3 and 10.

In addition to the simulation effort described above a joint Air Force/Lockheed non-contractual simulation of the XV-4B was implemented at the AFFDL simulation facilities at WPAFB. This six degree of freedom simulation provided motion and visual cues and permitted realistic handling qualities investigations. Lockheed pilots flew the simulator many times, prior to and after flight testing had begun, and considered the simulation quite valuable in preparing them for actual flight experience.

(3) Operational Experience--The PFCS performed quite satisfactorily during the flight test program. The following observations regarding system performance and problems encountered are, however, considered pertinent:

- After the PFCS was first interfaced into the XV-4B many minor faults were discovered which could be attributed in almost every instance to incorrect wiring (wires connected to incorrect connector pins) or improper wiring procedures (cold solder joints, intermittent short circuits, etc.). All of these faults were discovered and corrected during the functional tests of the PFCS. The Line Test Unit (LTU), supplied by the controls subcontractor, was the most useful tool in troubleshooting the system to pinpoint faults. The fault light panel on the engaging controller gives only an indication of which axis has faulted. The LTU has inputs from test points throughout the PFCS and can determine in most instances the faulted cube or incorrect wire. The Line Test Unit was used prior to each flight by flight test engineers to preflight the PFCS and again after engine start in conjunction with the pilot's checkout of the system.

- Only a small number of faults were encountered after release of the airplane for flight. This can be attributed to two things. First, the complete exercise of the PFCS prior to flight; and secondly, the selection of tolerance bands to minimize nuisance faults. For these reasons, no fault lights resulted while the XV-4B was actually in the air. There was a recorded roll light on one landing rollout, a yaw light on taxi out, and a pitch light that caused an aborted takeoff. These faults were spread over the entire flight testing period of the XV-4B and did not occur on the same day. The landing on which the roll axis light illuminated was accompanied by a collapsed right strut which required the pilot to hold in full left control column deflection for a relatively long period of time. After troubleshooting the PFCS, it was determined that spikes appeared on the pulse signals (see Reference 1 for detail system description) when the actuator was commanded to full deflection. Further, if this deflection is commanded for a long length of time, the fast-charge/slow-discharge characteristics of the fault detection circuitry would integrate these spikes causing the fault light to illuminate. The yaw axis light was non-recurrent, resettable and was attributed to a PFCS transient. The pitch axis light was due to loss of signal from the branch A force transducer. This was corrected by straightening a pin in the connector to the engaging controller. This type of elusive fault was not common but when it did happen was disconcerting because of the delays spent troubleshooting. During the latter portion of the flight test program the number of faults, especially nuisance faults, diminished greatly. Of these faults there was a preponderance of 12 volt power supply cube failures. These were attributed to quality control of the cubes. The cube was manufactured with an incorrect resistor.

- The only fault traced to the power actuators was due to an O-ring failure in the pitch axis actuator model sensor.

- The PFCS has three parallel branches of electronics on each of the pitch, roll, and yaw axes. The majority of all problems which were experienced with this type of system can be placed into the following categories: system nulls, component tolerances, and aircraft wiring. It seems worthy of note that actual component failures were an insignificant part of the total maintenance requirements of the system. The performance of the components necessary for the surface deflections, i.e., amplifiers, rate gyros, lvdt's, power actuators, and force sensors, trim and gradient-change actuators, was quite reliable.

- In the triplex system configuration used on the XV-4B there were only two stages of mid-value logic, one stage at the end of the electronics chain and another stage in the electrohydraulic power actuators. This mechanization reduces the number of system components, thus upgrading system reliability. However, this demands that the individual components possess very tight tolerances so tolerance buildups in the system will not prejudice the fault detection circuitry or require inordinately large detection thresholds. The component tolerances in the XV-4B PFCS were large, however, since this system was not part of a large production program. The large tolerances could be married to produce close tolerance outputs on a long signal chain. A component which was of nominal value in the high direction could be married to a component which was lower than nominal in value producing an output signal near nominal value. This practice is not recommended for systems in which maintenance is a problem.

- Null signal stability was also encountered in the XV-4B PFCS. The triplex signal paths and fault detection logic null imbalances, especially in high gain stages, were detrimental to the fault detection and indication processes. These imbalances were a problem only in the force transducer assemblies, where the high gain of these signals greatly amplified the effects of a null imbalance. Because of the null drift characteristics of the sensors null balancing was required on a daily basis. To facilitate balancing, special trim potentiometers were installed in a readily accessible area.

- Because of spring cartridge friction which varied as a function of temperature and humidity the spring cartridge breakout forces were not constant. As a consequence a perfect match could not be main tained between the force sensors and the mechanical forces.

- The mechanical backup system is considered an alternate for conventional flight only since it is questionable whether or not the airplane can be controlled at the lower VTOL speeds with an ideal unaugmented mechanical system and because the XV-4B system is far from ideal. The high friction levels and large hysteresis bands exhibited by the system are attributed to the following:

  1) The force levels necessary to drag the power actuators after they have been bypassed.

  2) The breakout and force gradients of the feel system spring cartridges.

  3) Cable and pushrod friction.

  4) The friction and high actuating forces of the reaction control valves when pressurized.

  In conventional flight the last item is relatively small since the bleed air system is depressurized. Even for this case, however, control forces are still quite high and control centering is poor with a consequent degradation in handling qualities. In spite of these detracting characteristics, from a flight evaluation of the system, it was concluded the system was satisfactory for emergency conventional flight operations.

3. PROPULSION SYSTEM

This section presents a description and related test experience of the XV-4B propulsion system as installed in the aircraft. The propulsion subsystems covered in this section include: Engines, Fuel, Start, Bleed, Exhaust, Cooling, Power Control, Engine Instruments and Protective Sub-systems.

a. Engines

As shown in Figure 2, the XV-4B is powered by 6 General Electric YJ85-GE-19 engines. Each of the cruise engine installations contained in wing-root nacelles include a propulsion engine, diverter valve, longitudinal ducting to the horizontal thrust nozzles and ducting downward and inward to the lift nozzles in the fuselage. Each of the four mid-fuselage vertically mounted direct lift engine installations include a propulsion engine, tailpipe, and lift nozzle.

The lift/cruise engines provide forward thrust when exhausting through the nacelle tailpipe and nozzle assembly and lift when diverted into the vertical lift nozzles by the actuation of a diverter valve in the nacelle tailpipe. The mid-fuselage mounted lift engines provide vertical thrust and are shut down once the vehicle has reached wing borne flight. All six lift tailpipes are equipped with swivel nozzles to permit the thrust vector to be varied $\pm$ 10° from the nominal position in the fore and aft plane while in the VTOL mode of operation. In addition all six engines supply compressor bleed air for VTOL reaction control power. The compressor bleed extracted from each engine is ducted through a central manifold to the aircraft's extremities providing thrust on a demand basis.

The sea level static standard day uninstalled rating for all six engines is 3015 pounds thrust each at maximum (five-minute limit) power setting. The maximum continuous thrust rating of 2950 pounds at sea level static standard day conditions permits extended hover time operations as the gross weight diminishes with fuel burn-off. The engines were qualified to run in the vertical position within a 3 1/2 hour Preliminary Flight Rating Test. The uninstalled weight of both lift and lift/cruise engines is 387 pounds.

The basic engine as received was modified physically to meet the requirements of the installation in the XV-4B. These modifications, approved by the General Electric Co., were to the external configuration and consisted of the following:

- The T_2/P_3 aspirator line was remocked to accommodate a new P_3 pick up point, and the fuel flow transmitter was relocated.
- A new rerouted high pressure fuel line to the overspeed governor was installed in order to accommodate the rotated fuel flowmeter.
- A high pressure fuel tap was added to the overspeed governor line to provide motive fuel flow for the airvehicle fuel boost system.
- The ignition exciter unit was removed from the engines assigned to the lift positions and remotely mounted.
- The trunnion mount pads were removed from all engines assigned to the lift positions and from the outboard side of the engines assigned to cruise positions.
- The tail cones were removed from the engines assigned to the cruise positions and replaced with a Lockheed fabricated diverter valve assembly.

(1) <u>Operational Experience</u>--The engines were initially assembled in the aircraft configuration and installed in the cyclic test rig. This rig was operated in a proof of operation test of the airplane hot hardware, exhaust systems and bleed systems, for a period of 3[illegible] test hours, and resulted in an actual engine running time of approximately 50 hours. The rig was installed on a VTOL test pad configured so that the engine exhaust gases were ducted away from the engine inlets. This effectively allowed the engines to be operated relatively free of ground effects, discounting the far field effects after prolonged operation.

In the course of engine running on the cyclic test rig numerous high speed engine stalls were encountered. Each stall was attributed to hot gas reingestion. The open, unshielded, nature of the rig and an improperly configured exhaust deflector in the VTOL pad exhaust system were determined to be the cause of the stall problem. After the exhaust deflector was modified, no further engine stalls were encountered during the cyclic testing. Upon completion of the cyclic testing the engines and hot hardware were transferred to the airplane.

The airplane was then operated on the inverted telescope (as described in Secrion VII) over the same VTOL pad used for the cyclic tests, both with the pad exhaust system active and inactive. During operation with the aircraft raised approximately 12 feet above the normal static ground position, an engine stall was encountered on the number 2 cruise engine. The cause was determined to be that the engine inlet was in the lee of the vehicle with regard to ambient winds. To evaluate this condition the airvehicle was rotated putting the #1 cruise engine in the lee and an engine stall was encountered on that engine. These occurrences were with the VTOL pad exhaust system deactivated by a steel cover over the grate at the ground plane.

In actual free flight and ground running, engine stall did not appear as a problem in the spectrum of operation covered, although the area of operation most likely to produce a stall inducing environment was not explored.

To reduce the probability of a stall occurrence in vehicle operation a study was conducted with the aid of the General Electric Co. to determine the most effective and least penalizing change to the engine to effect a more comfortable stall margin. It was estimated that an increase in stall margin of 6.6% with a corresponding thrust loss of 2.95% could be effected through a reduction of engine compression ratio by increasing the turbine entry nozzle area by 7.75%. Two sets of turbine nozzle assemblies were reworked at the engine manufacturer's facility to provide this increase in area. These reworked nozzle assemblies were then installed in the lift/cruise engines and tested, where it was determined that, although the actual increase in nozzle area amounted to only slightly more than 6%, the maximum change permissible within the physical size limits of the existing swivel nozzle assemblies had been accomplished. For the remainder of the program the lift-cruise engines were operated in this configuration. A planned layup to accomplish similar changes on the lift engines was not achieved prior to the termination of testing. No evaluation of stall margin improvement effected by this modification was made.

At the beginning of the program a hot-section inspection schedule was established for all engines at 10, 25, and 50 total cumulative hour periods, and again at each succeeding 50 hours of accumulated time. As a result of the Preliminary Flight Rating Test experience, the engine manufacturer revised this schedule to require hot-section inspections at each interval of 2.5 hours cumulative operation at maximum bleed conditions; further, that such inspections would be required on one representative lift engine and one representative lift-cruise engine. This inspection was accomplished during the cyclic test program on one lift-cruise engine upon accumulation of 2.5 hours maximum bleed time, which corresponded to a total engine time of 31 hours, and on one lift engine after the same interval of maximum bleed operation, which occurred at 19 hours engine time. These inspections revealed no engine degradation, permitting continued operations until the next prescribed inspection period. At the next inspection period, upon accumulation of 5 hours of maximum bleed time, the cruise engine had accumulated 66 hours and 43 minutes total operating time, and the lift engine had accumulated a total time of 48 hours and 16 minutes. Inspection results again revealed no engine degradation. As a result of this operating and inspection experience, the final hot-section inspection schedule jointly established by the engine manufacturer, the contractor and the Air Force, was 100 hours cumulative engine time, or 75 hours total (not maximum) bleed operating time, whichever should occur first.

b. Fuel System

The XV-4B fuel system consists of two integral tanks, boost pumps, ejectors and associated valving. The two approximately equal volume tanks, placed forward and aft of the aircraft C.G. for fuel balance have a total capacity of 740 gallons. Each fuel cavity includes a surge tank of sufficient capacity to ensure a positive head of fuel on the boost pumps. The primary boost pumps are the ejector type, using excess fuel flow from the engine driven feed pumps as motive power. Each engine powered ejector discharge is manifolded into a common line feeding one cruise and two lift engines to provide redundancy and fail safe operation. Proportioning is accomplished by the fuel consumption characteristics of the engines. An electrically driven boost pump was provided in each tank to supply fuel to the engines in the start cycle and to operate the scavenge ejector in the tank. The boost pump also powers the cross feed system between tanks and in conjunction with the Fuel Quantity System automatically effects a fuel balance condition. Relative fuel levels in the tank can also be controlled by manual operation of the balance control valve switches in the Flight Station. The two tanks are gravity filled by an individual filler on each tank located flush with the external upper fuselage skins. Each tank is vented through a one-inch diameter vent line discharging into a portion of the airvehicle slipstream that will remain at ambient pressure or slightly negative. A water drain is provided in each tank accessible to the ground crew so that any accumulated water may be drained from the tank on a preflight inspection. Defueling is accomplished by removing each of the cruise engine feed lines and pumping the tanks dry with the ship's boost pumps and draining the remaining residuals through the water drain valves.

Balance and gaging is effected by a single fuel probe located in each tank to indicate fuel weight in the tank. This intelligence is indicated to the operating crew in a dual conventional round dial instrument. Fuel management is effected either manually or automatically with a selector switch operated by the crew. In the automatic mode a system of differential gearing in the dual instrument controls the fuel pumping system by switching the fuel balance valves from forward to aft pumping such that a maximum unbalance of 100 lbs of fuel is maintained between the two reservoirs.

(1) Operational Experience--In operating the airplane it was found that due to the symmetrical usage of fuel by the engines it was possible to maintain adequate C.G. control by power demand modulation of the engines. The automatic system then became a fail safe C.G. control in the event of an engine out condition which was the basic design premise of the system. The manual mode was used quite extensively to effect a C.G. control in lieu of ballast. The fuel system in all respects conformed to the requirements of the Model Specification. (Reference 1)

c. <u>Engine Start System</u>

The YJ85-GE-19 engine has an integral air impingement starter and requires air delivered to it from a power source to motor the engine. The impingement starter is built into the turbine case and directs the air onto the second stage of the turbine rotor. The compressor bleed ducting is utilized as a start air supply manifold. As described in Reference 11, check valves in the ducting act to prevent reverse flow into the compressor sections of the engines during the engine starting cycle.

(1) <u>Operational Experience</u>

It was found that during the start cycle the leakage from the reaction controls, normally supplied by the compressor bleed ducting, was large enough to necessitate the use of two MA-1A ground air supply carts. With two carts providing air, problems were not encountered during the XV-4B test span. However, at higher ambient temperatures, the system could become marginal due to the reduction in air delivery from the ground carts.

The engine can also be started by cross bleeding from an operating engine or engines. Lift engine starting by cross bleeding from the cruise engines in flight was completely successful. The trim correction necessary in flight did not require sufficient reaction control power to reduce the air supply below the engine starting requirements when engine starts were accomplished. Windmill starting of the cruise engines within the engine starting envelope was demonstrated successfully.

All engines were modified to include an auto-ignition system to initiate the ignition cycle on a sharp decay in compressor discharge pressure, P_3. This system was incorporated to accomplish the intent of continuous ignition during adverse operating conditions. This system was required since the engine-supplied ignition exciters were of a limited duty cycle and could not be energized for the extended time periods required for prolonged VTOL operations.

d. <u>Compressor Bleed System</u>

The compressor bleed system uses compressor bleed air from the customer service ports of all six YJ-85-19 engines. As depicted in Figure 2 this air is collected and manifolded into a common duct subsystem, designed to convey the bleed air to the reaction control system, to provide air for the air conditioning subsystem, to provide a path for air from a ground service cart to the air impingement starter on each engine, and to provide a cross bleed starting capability between engines.

The compressor bleed system on the XV-4B is an integrated portion of the flight control system. This system provides the air to the reaction control valves used for maneuver control when the normal control surfaces are ineffective. In this usage the system necessarily had to be designed on a prime reliable basis. This design criteria necessitated the formulation and testing of ducting in a new area of operational requirements, which could only be established through operational testing in a VTOL environment. The cyclic test program described in Section VII was the means of accomplishing this task.

The following criteria established during the cyclic test program for prime reliable bleed ducting in a VTOL environment became the basis for the bleed system developed for the XV-4B:

- All significant forces both static and dynamic shall be taken into account in the ducting system. The maximum design pressures shall include peak pressures resulting from surges, stalls, sonic flow, etc. A factor of 1.5 x operating pressure for proof pressures and a factor of 2.5 x operating pressure for burst pressure shall be used. The above pressures shall be at the highest design operating temperature.
- All noise and vibration levels shall be taken into account in a fatigue analysis of the ducting system.
- Stress concentrations shall be held to a minimum by eliminating excessive mismatch, concentricities, sharp corners, etc. A condition which indicates a stress concentration factor equal to or greater than 4.0 shall not be permitted.
- The factor of safety shall be 2.0 on burst unless it can be shown by a rational fatigue analysis that a lesser factor provides adequate life.
- A.R.P. 699 - High Temperature Pneumatic Duct Systems shall be used for design parameters not expressed in the foregoing items.
- A complete verification test shall be run on the actual installation and results used to evaluate the system on a prime reliable basis.

The installation and components of the bleed subsystem are described in detail in Reference 3.

As a result of continued testing of the bleed subsystem during the guarantee demonstrations with the system installed in the airplane the recommendations for a 25 hour replacement of all ducts was revised in accordance with Table IX of the Appendix. The minimum life predicted as a result of data obtained during actual airplane operation was in excess of 2000 hours.

e. Exhaust System

The exhaust system is considered to be the components between the engines and the exhaust nozzles conveying the hot discharge gases from the engines to the nozzles and thus providing the propulsive power for the airplane. The cruise engine exhaust system includes the diverter valve, cruise tailpipe, diverted lift tailpipe, cruise nozzle and a vectorable lift nozzle. The lift engine system includes a tailpipe and a vectorable lift nozzle.

The material used for these components was Inconel 718 and they are of welded construction. The joints used at the juncture points are standard "V" band coupling assemblies. Differential movement due to the thermal expansion and airplane bending is compensated for with two ply bellows sections located at controlled points in the system. The cruise system is insulated with 1/2 in. thick high temperature insulation for the control of the airplar structural temperature.

(1) Operational Experience--These items were qualified for use in the cyclic test program. The following modifications were made as a result of experience with the actual installation on the airplane.

(a) The original diverter valve actuator was a single cylinder with a built-in shuttle valve to shift between #1 and #2 hydraulic systems. The shuttle was set to transfer systems when the active system was below 875 psig pressure. When the system was tested on the airplane, the time delay for the inactive system to bleed down to the shuttle operating pressures was too long and this time period would allow the diverter valve to assume a trail position. This trail position would allow both exhaust systems to be active and, due to the double size effective exhaust nozzle, result in unsatisfactory engine performance. This was considered to be a safety of flight item. The actuator was redesigned to a dual tandem unit so that both hydraulic systems were effective at all times. Due to the necessary time required to make this change, the first conventional flights and the early Phase III flights were made with the diverter valves mechanically locked in the cruise mode.

(b) The dual hydraulic system control of the diverter valve provided a dual redundant failure mode control of the diverter valve, but the results of the failure tests indicated that the valve would trail without hydraulic pressure. Based on this experience it was decided to provide a system of mechanical locks so that in the event of a dual hydraulic fault or failure, the diverter valve would be held in its commanded position mechanically. These locks were installed in conjunction with the aft nacelle cooling doors which are discussed in a later paragraph in this section of the report.

(c) The single unresolved component failure at the completion of the cyclic test, namely that of the tailpipe expansion control bellows rupture, was not a problem in the airplane installation. An improved dual laminated bellows was installed in the tailpipes and was exposed to operational environments throughout the guarantee and flight programs with no apparent problems. Therefore, the recommendation for a five hour replacement of the expansion bellows assemblies was revised in accordance with Table VI.

f. Engine Bay Cooling System

The engine bay cooling system is an ejector powered system. The main engine exhaust flow induces the cooling air flow through an annular ejector. The cooling air enters the cruise engine bay through an inlet in the chin of the nacelle and exhausts from the bay either through the cruise tailpipe ejector or the lift tailpipe ejector depending on the mode of operation. The lift engine bay cooling air enters a plenum above the engine firewalls through louvers located forward and aft of the lift engine inlets. The air then routes through the equipment bay located over the diverted tailpipes and into the engine bay via a passage in the vertical fire wall.

Cooling performance was demonstrated to be acceptable in all modes of operation of the airplane and temperature data were approved by the General Electric Co.

(1) Operational Experience--In the course of development, it was found that the internal diverter valve leakage through the inactive exhaust system, although only about 1% of the primary flow, was sufficient to cause excess heating of the inactive portion of the nacelle. This was attributed to the fact that the exhaust velocity was too low to eject the hot gases from the airplane. This problem was resolved in two steps.

(a) Cruise Engine in the Cruise Mode--The leakage gases exhausting from the lift exhaust system were allowed to egress from the airplane by the addition of a hole in the lower exit door with an effective seal to the cruise engine swivel nozzle ejector shroud. An extension was added to the hole with a suction cut so that in adverse airplane attitudes this hole would not be pressurized by ram flow. The upper fire wall on the lift elbow compartment was insulated and the equipment bay above the fire wall was cooled with the addition of an ejector pump, powered by bleed air, exhausting into the cruise engine nacelle. The most temperature sensitive equipment located in this bay was relocated to areas with more acceptable ambient temperatures.

(b) Cruise Engine in the Lift Mode--The leakage gases exhausting from the cruise exhaust system would back up into the cruise engine nacelle through the ejector shroud. It was found by experimentation that the cooling system would operate successfully by removing the shroud and allowing the gases to egress from the tailpipe by natural ventilation. This was implemented on the airplane with the addition of mechanically actuated cooling doors. These doors were located so that when open a path was available to the gases to leave the nacelle by convection and when closed the ejector shroud would be functional for pumping cooling air with the engine in the cruise mode.

The doors were articulated with the addition of a drive crank on the individual engines diverter valve drive assembly so that the diverter valve actuator would also drive the cooling doors. A hydraulically actuated pin was installed in this linkage so that with the absence of hydraulic power the pin would engage the linkage and effectively lock the system in the commanded position.

This leakage problem did not affect the cooling air flow to the basic engine assembly since it all occurred downstream of the engine.

g. Engine Thrust Control System

The engine thrust controls consist of three throttle levers at each pilot station arranged for pilot left hand operation, with the left curise engine throttle lever on the left, the collective lift engine throttle lever, which controls the four lift engines, in the center, and the right cruise engine throttle lever on the right in each group of levers. Each power lever has a positive lift action gate at the engine minimum bleed setting and the start idle position. The left hand pilot is also provided with a collective lift lever located just aft of the left hand group of levers. This lift lever physically operates the collective throttle lever from the 80% minimum bleed gate to 100% power setting only. The lift lever has a magnified travel compared with the collective throttle lever and may be mechanically engaged or locked out at the discretion of the pilot. In order to adjust power on the lift engine individually, a system of trim actuators is provided with an actuator in the control linkage to each engine which provides a 30% throttle position authority.

(1) Operational Experience--Due to manufacturing tolerances in fabrication and installation of the system the geometry of each engine control system was not identical, resulting in unsymmetrical programming of the power controls. This throttle misalignment condition did not exceed the allowable deviation; however, it did represent an undesirable condition which was not evaluated in the VTOL regime. Although never properly assessed, concern was often expressed during the test program that because of the inability to completely control each lift engine throttle separately the pilot would have no capability of recovering from a lift engine compressor stall. This remains as the most questionable area of throttle system performance since no experience was gained to evaluate this design aspect of the system.

h. Instrumentation System

The cockpit instrumentation in the XV-4B is sufficient to show engine health, approximate power delivered, and state of operation. The parameters presented are as follows:

(1) Engine pressure ratio

(2) Engine RPM

(3) Exhaust gas temperature

(4) Oil pressure

(5) Fuel quantity

(6) Diverter valve position

(7) Swivel nozzle angle

(8) Bleed system manifold pressure

(9) Fuel engine inlet pressure minimum

The only problem that occurred in this sub-system was erroneous indication of the vertical tape EGT system during UHF or VHF transmission. This was corrected by the addition of a RFI filter network in the electronics of the EGT system.

i. Protective System

The protective system consists of four elements.

(1) Engine bay fire and overheat detection sub-system.

(2) Duct overheat detection sub-system.

(3) Fire extinguisher sub-system.

(4) Fireproof isolation of bays.

Each engine compartment or bay can be completely isolated from the rest of the airplane. This is accomplished through the medium of titanium firewalls and shut off devices for all services both to and from the engine. Therefore, any hazardous situation that occurs in an individual engine bay can be controlled so that it will not migrate throughout the aircraft.

In the process of installing an ambient temperature sending system in the engine compartment, judgment had to be used to determine proper location of the element to do the sensing job required. After the airplane was functional, minor relocations were made to correct occurrences of false warnings. This process was accomplished during the initial runs on the Inverted Telescope.

One addition was made to the system in the cruise engine nacelles resulting from the hydraulic actuated diverter valve lock installation. Because this introduced additional combustibles in the compartment, an ambient temperature sensitive switch was installed in parallel with the continuous element unit in control of the compartment, located in close proximity to the diverter valve lock hydraulic installation in the tailpipe section of the nacelle.

4. ELECTRICAL SYSTEM

a. Description

XV-4B electrical power is supplied by two 300-amp engine-driven generators operating in parallel to furnish 9-volt dc power to the bussing and distribution system. Two 2.5 KVA inverters feeding from the main bus furnish 115-volt, single-phase, 400-cycle constant frequency AC power. A sintered-plate, nickel cadmium battery provides 3 ampere-hours of emergency dc power to those necessary aircraft functions powered by an essential bus. Fifteen distribution busses power the various electrically operated units and equipment. The overall power system is schematically illustrated in Figure 5. Other than the generators and control switches, all of the equipment shown is located in a power shield mounted in the aft fuselage equipment compartment. A circuit breaker panel is mounted adjacent to the power shield and contains all breakers except those serving the flight control system which are located in the right hand pilot's console.

DC power is extracted directly from the main bus. The generators feeding the bus are automatically controlled by electrical control panels which function to: regulate output voltage, equalize generator load sharing, protect against overvoltage, and allow a generator to be connected to the bus only when its voltage is within prescribed limits. Reverse current relays (RCR) operating in conjunction with the control panels perform the necessary switching functions. Should one generator attempt to feedback into the second unit while in parallel operation, the RCR will open and trip the generator off the bus. The essential bus powered by either the main bus or the battery allows critical functions to be powered without requiring the pilot to monitor their operation. The bus is connected to the main dc bus with a RCR identical to those protecting the generators.

Normal AC power is provided by the two inverters operating in a non-paralleled manner and each carrying a share of the total load. Failure of one inverter automatically transfers the load to the operating unit through voltage-sensing relays. For loads which operate from 26 volts ac, a stepdown transformer provides the necessary power from the No. 1 inverter.

The basic philosophy underlying the electrical power system is to have sufficient capacity in a single generator and inverter to carry the total system loads. Maximum connected dc loading is 265.8 amps which drops to 250.6 amps for 5 minute operation under taxi conditions. The connected ac load is 1193.5 va with a .8 power factor.

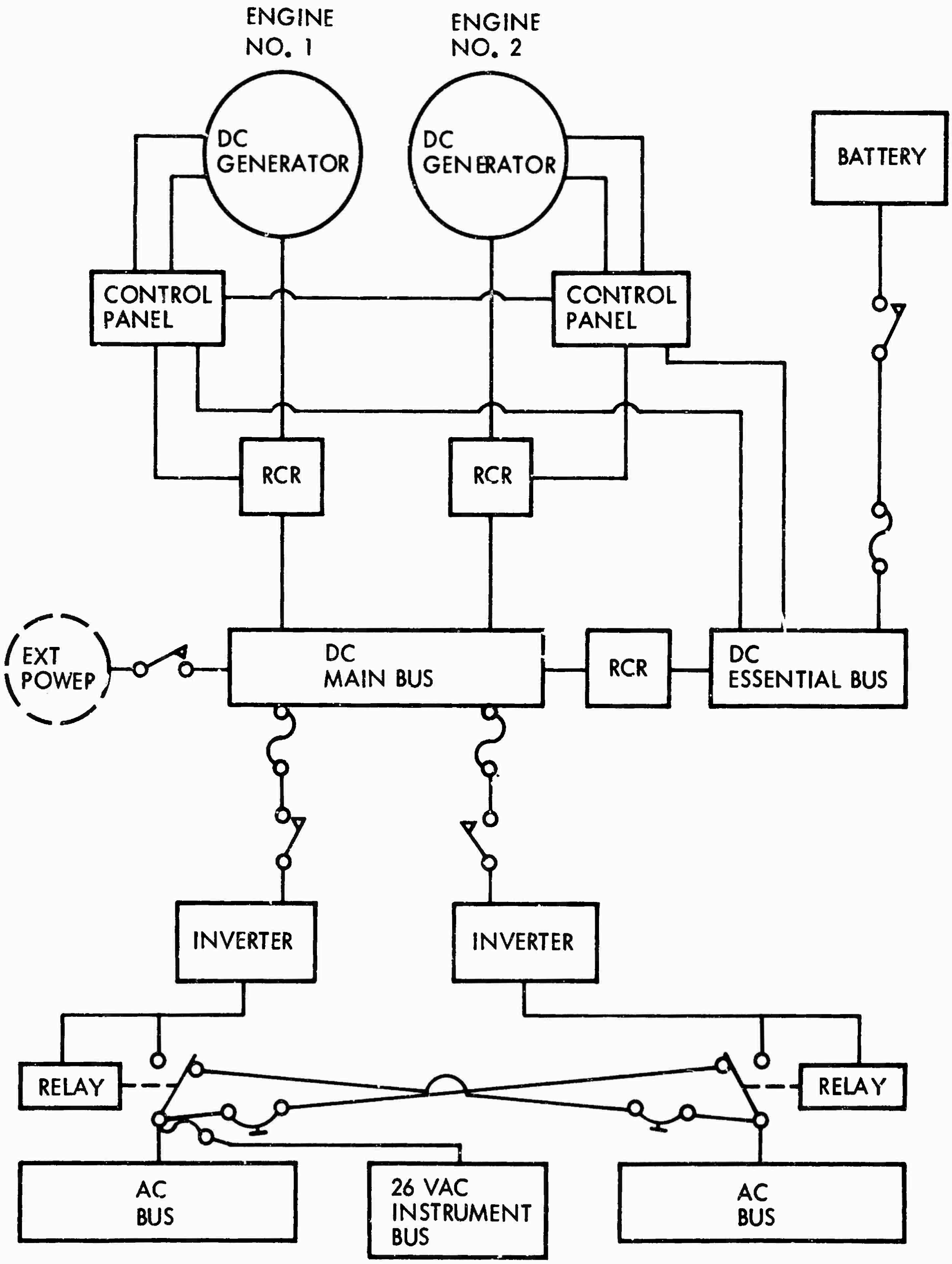


FIGURE 5 - ELECTRICAL POWER SYSTEM SCHEMATIC

b. Operational Experience

Experience with the system indicated satisfactory operation and reliability. On one occasion, however, while the aircraft was operating on the Inverted Telescope, shutdown of one engine resulted in complete loss of electrical power. Subsequent investigation and analysis disclosed an off-design point condition which when coupled with a low battery charge could result in the situation encountered. Several modifications were made and included: (1) rewiring the equalizing cut-out to delay it until the generator was disconnected from the bus; (2) adding diodes to the battery relay to enable it to be powered from either the main or essential dc busses; (3) generator trip switches were added for use in an electrical emergency; and (4) loadmeter shunts enabled generator load sharing to be monitored during engine ground running.

With the changes implemented, subsequent system performance and reliability were satisfactory.

5. COMMUNICATIONS/NAVIGATION SYSTEM

a. Description

Communication and navigation equipment in the aircraft consists of VHF and UHF transceivers, interphone and VOR/Localizer navigation systems. Power supplies and other major components are located in the aft equipment compartment close to the fuselage and vertical stabilizer antennas to minimize co-axial cable runs and radio frequency interference. All control and switching equipment is in the cockpit.

The VHF transceiver is amplitude modulated (AM) and operates in the frequency range of 118.00 to 135.95 MH_z with a capability of 360 crystal controlled channels separated by 50 KH_z spacing. Transmitter power output is a minimum of 6 watts; and power consumption is 3.2 watts during receive operations. Frequency selection, squelch, volume control and ON - OFF operation is housed in the control unit mounted in the center console. This same unit is also used to select navigation frequencies.

A solid state, crystal controlled UHF transceiver provides 5 channels of AM communication covering the 225 to 300 MH_z frequency band. Channels 1, 2 and 3 operate on a Lockheed engineering flight test frequency of 275.2 MH_z and the remaining channels were set to 275.8 MH_z for the Dobbins AFB tower frequency. The control unit located in the right hand console provides ON/OFF switching, channel selection, and volume and squelch control. Nominal power during transmittion is 3 watts.

Interphone and communication operation is controlled from the audio selector panel located in the center console adjacent to the VHF transceiver. Appropriate switch selection enables operation in the desired mode. The panel also provides the capability to interface three stations: pilot, co-pilot and external service at the right hand side of the forward fuselage. A priority system enables the pilot to override the co-pilot and both to have precedence over the service station. An emergency switch enables the interphone system to be by-passed and permit direct reception of transmission at the earphones. The interphone also functions as a pre-amplifier and impedance matching device for the various units in the communications systems. In addition, audible tone warning signals from the landing gear and stall warning subsystems are introduced through the pre-amplifier.

Navigational capability is provided by equipment which integrates the magnetic heading of the aircraft with VOR/Localizer signals and presents a continuous display of the aircraft's position. A flux detector sensing magnetic heading and a directional gyro for short-term heading stability coupled by a slaving accessory are the major system components.

The systems described were selected for the vehicle as providing the greatest degree of capability and flexibility at the lightest weight. Equipment is typical of that in commercial and business aircraft. Initial design considerations concluded that the VHF communications would be adequate for operations of a flight research vehicle. However, this presented some problems when operating with standard military chase aircraft such as the T-33. Either UHF was needed in the XV-4B or VHF was to be added to the chase plane. Unless this was done, aircraft-to-aircraft communications would be by ground relay and would present a somewhat unwieldy situation. As a result, the UHF minimal capability was added. Flight test operations were conducted utilizing the UHF as the primary mode and VHF for alternate or back-up operation.

b. Operational Experience

Aircraft operations with these systems were found to be satisfactory. UHF communications were good though the equipment was low on power output. Severe blanking was observed particularly with the chase aircraft operating to the side of the XV-4B and coincides with a null in the lobal antenna pattern. The navigation system was never used extensively and no large compass errors were observed. Interphone operation was adequate during ground running operations.

6. FLIGHT STATION

a. General

The flight station of the XV-4B is designed to be conventional in configuration and arrangement with two crew members seated in a side-by-side manner. A separate set of throttle controls is available on the left hand side of each station to enable aircraft operations to be conducted identically from either seat. Unlike conventional aircraft, the pilot normally flew the vehicle from the right as is the practice in helicopters. The pilots did not find this to be uncomfortable or awkward primarily because the relatively small size of the cockpit brought all switches and equipment in close proximity to either seat. The general arrangement is illustrated in Figure 6.

Access to the cockpit was from the right with the single-piece canopy hinged on the left. Entrance was somewhat awkward because the windshield frame required the pilot to crawl over the sill, stand on the seat and then assume a seated position.

The field of view was considered to be good both in the forward and side directions, Reference 12. However as the flight test program progressed, several switches were added above the center of the panel and somewhat reduced visibility. In this respect, the windshield frame is in a position to interfere with the forward visibility.

The cockpit was found to be roomy and generally well arranged. Overall the seat-stick-pedal arrangement was good for the 50th percentile pilot. However, legroom was considered to be limited laterally and the control stick contacted the thighs during large roll inputs. This may have been a result of either minimum rudder pedal spacing or an extremely narrow ejection seat shell with high side panels.

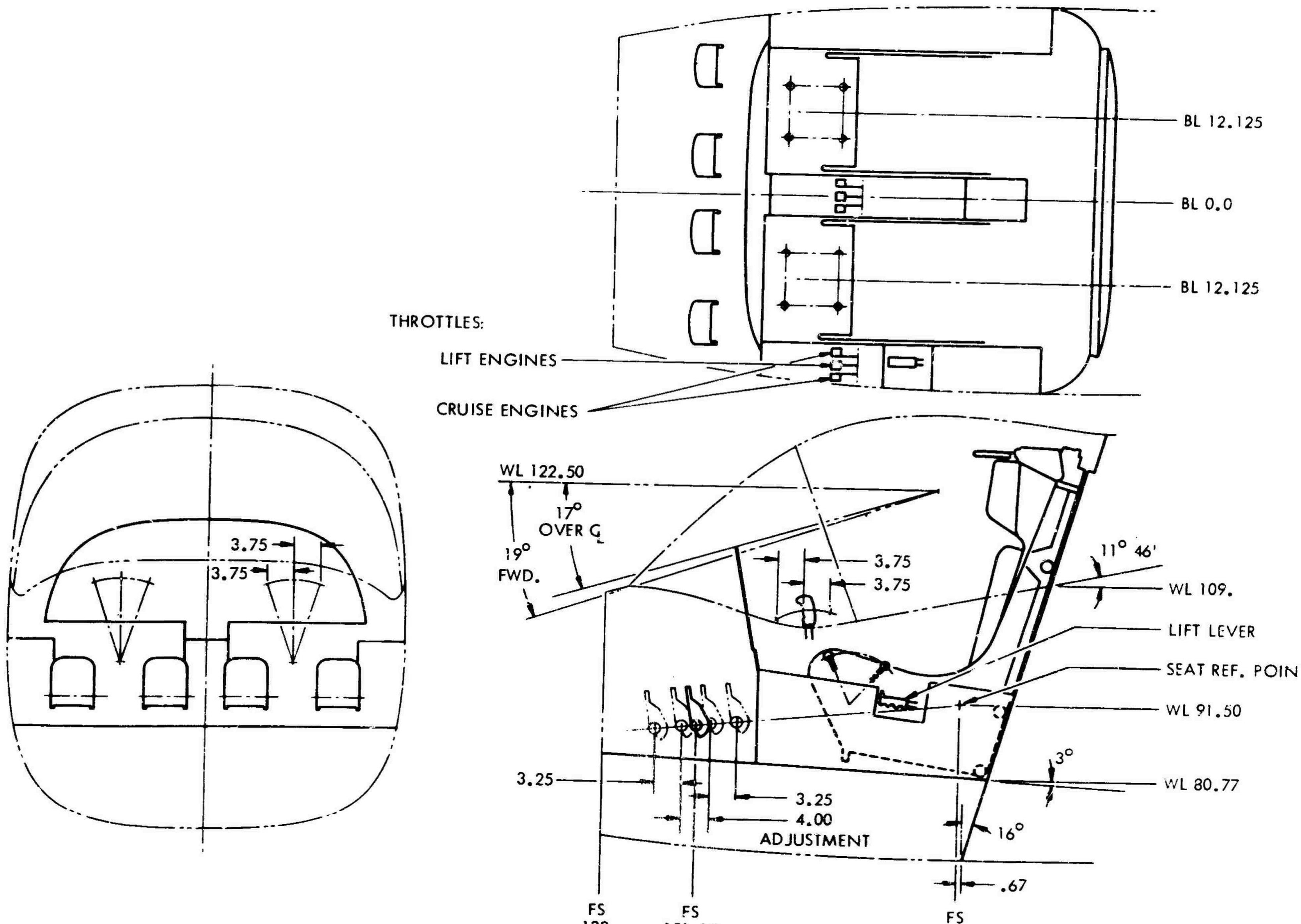


FIGURE 6 – FLIGHT STATION ARRANGEMENT

In conjunction with the rudder pedals, the angle was found to be too vertical and required excessive toe travel for application of the brakes.

b. Flight Instruments - Description and Operational Experience

Instrumentation was arranged as shown in Figures 7 and 8. Grouping of the basic flight instruments was conventional with the positions of the airspeed and vertical speed indicators interchanged. Within this grouping it was considered that the accelerometer could be moved to a less prime location and in its place substituted a large-scale, vector nozzle position indicator. Within the flight instrument group were:

(1) Attitude Director Indicator (ADI)--A conventional 5 inch diameter sphere provided basic attitude information for the pitch and roll axis. A blue field for the sky against a brown earth was found desirable; however, the circumferential 10° lines need to be extended. A combined pitch and roll attitude oftentimes rotated the sphere to a position whereby the indicated information was obscured from the pilot. The turn/slip pointer was found to be inadequate and should possibly be replaced with either a conventional turn/slip instrument or eliminated for an aircraft of this type.

Angle of attack (α) and sideslip (β) indications were provided by using the glideslope tape and vertical flight director pointers respectively. This was an effort to present all basic VTOL flight information on a single instrument. In operation, the ADI - α scale lagged airplane motions as a result of low tape drive speed and was found to be inadequate as a basic flight instrument. A 2 inch round dial indicator was installed directly above the left hand station master caution light on a test basis and proved satisfactory during the remainder of the testing. The pointer and dot β arrangement also proved to be too coarse an indication to be of much value in precision maneuvering.

During the early phases of the program, α and β warning lights were incorporated into the panel displays in anticipation of critical limitations on these two parameters. The light location chosen was a prime position on the center of the panel just below the upper edge so as to be directly in the line of vision. As the program evolved, additional data and experience indicated a possible revision to the lights. The β warning was of less significance as the results of wind tunnel

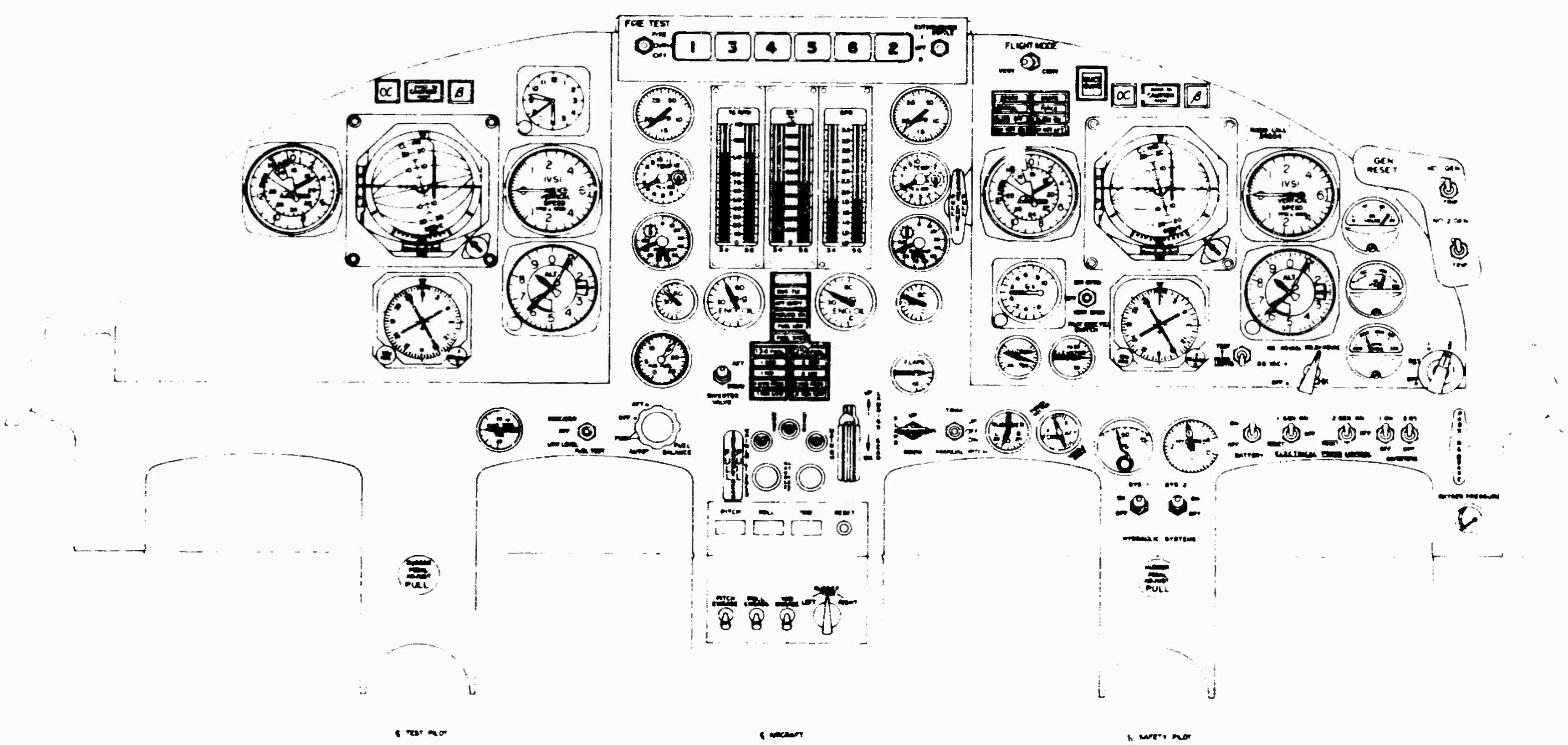


FIGURE 7 - INSTRUMENT PANEL ARRANGEMENT

CONTROL CONSOLE PANELS

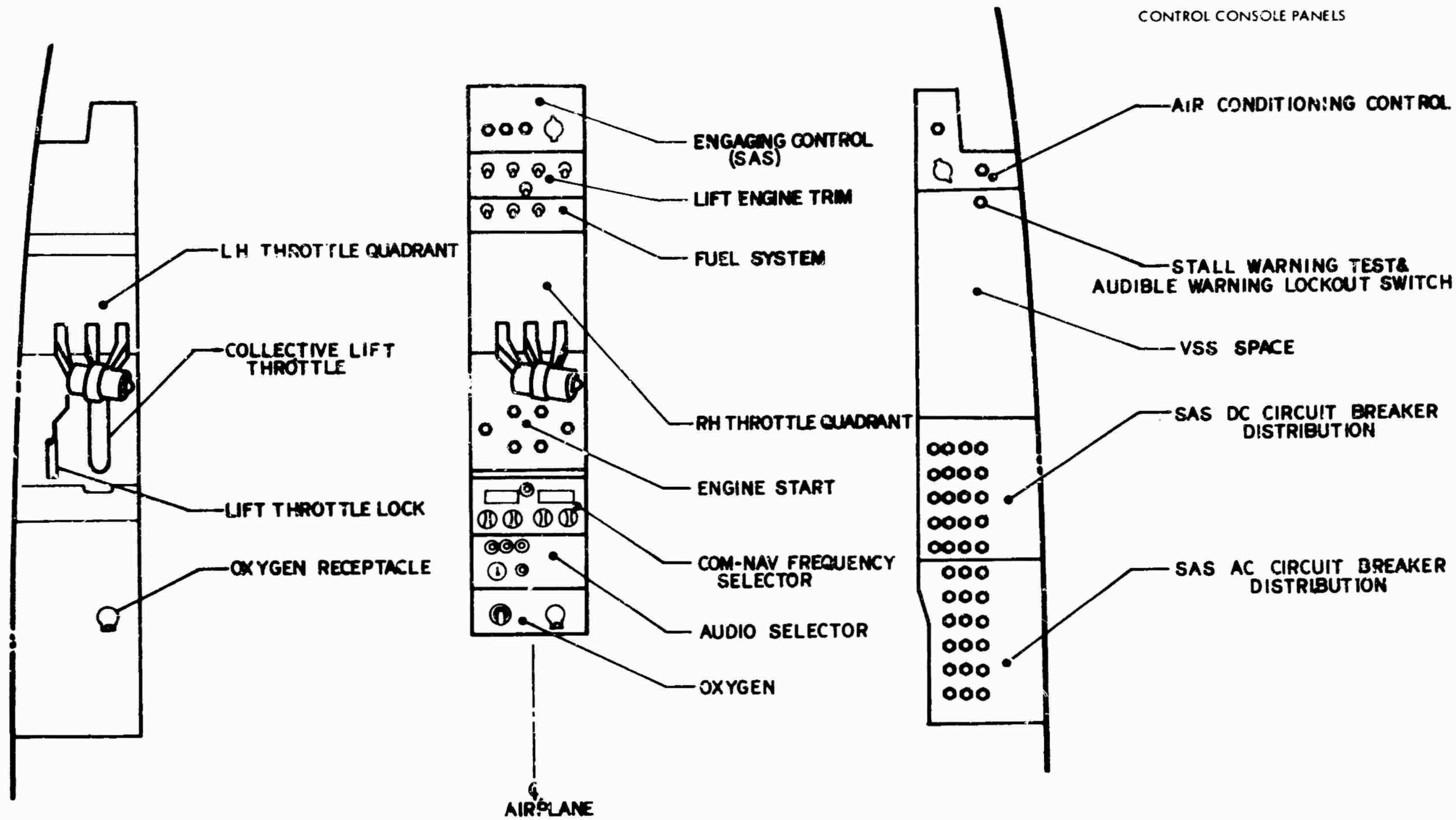


FIGURE 8 - CONSOLE ARRANGEMENT

testing showed the limits were greater than the initial analysis dictated. An audible warning tone added to the stall warning system and coupled with an improved indication superseded the function of the light. Assuming an improved β indicator and combined with the preceding modifications, the lights are of reduced value and could be either moved to a less prime location or deleted.

(2) Airspeed for both the CONV and VTOL speed ranges was provided on a single, dual pointer indicator. The outer dial covered the low speed range of 0 - 140 kts and the inner scale from 140 - 300 kts. At the crossover point, both needles were in view. Testing disclosed that five knot spacing was inadequate for precise speed control and more intermediate graduations are needed. In addition, at the changeover point, both needles indicated a different speed.

(3) An accelerometer was included in the basic flight grouping to insure the aircraft was kept within the + 3.0 and - 1.5 G limit load factors. The scaling on the instrument was not great enough to be of much value for these limits. Furthermore, maneuvering within these values posed no particular problems; consequently, the unit can be placed in a less prime position.

(4) Interchanging the altimeter and instantaneous vertical speed indicator (IVSI) was found to be a good choice. As the airplane flies into the Phase III mode, vertical speed and pitch attitude becomes a measure of α and the IVSI supplements the pilot's reference for this parameter. Though not verified on the aircraft at extremely low speeds, vertical speed provides better flight cues to the pilot than changes in altitude.

c. Propulsion Instruments - Description and Operational Experience

The propulsion instrumentation was arranged as illustrated in Figure 7 to provide a differentiation between the lift/cruise and lift engines (e.g. round dials as compared with vertical tapes). Slightly difficult to read during the early conventional flights when only two engines were functioning, the pilots became accustomed to the configuration and would scan the cruise instruments without really observing the vertical tapes in the grouping. However, revising the arrangement to place the cruise engine instrumentation adjacent to each other would be desirable. Another consideration would be to place all of the engine parameters on vertical scales instrumentation and distinguish between various engines by tape width and color coding.

With the XV-4B engines, rpm was the most used measure of engine state; however, the view of the No. 2 engine dial was partially obscured by the drogue chute handle. The vertical scale rpm indicators had expanded scaling for the 80 - 100 percent rpm range. This was found to be most valuable since most of the engine operations were conducted in this range.

Engine oil pressure readings were somewhat confusing on the instruments with dual pointers (lift engines). Since normal engine operation did not necessarily imply that the oil pressure was identical, there was always a question as to which engine had what oil pressure. The dual pointer presentation was only found to be satisfactory if the pointers were aligned for the normal operating condition. In the XV-4B this only applied to hydraulic pressure. Split needle operation was required for oil pressure and fuel quantity and a different presentation would be desirable.

The fuel quantity gage was difficult to read on the basis described above and its particular location on the panel resulted in a large measure of parallax distortion. In view of its significance the unit should be moved to a more prime location. The AUTO-FWD-OFF-AFT knob was awkward and difficult to manage.

Throttle lever location was good; however, the collective (center) throttle required too much force and too great a motion to lift it out of the detented position. It was learned that the throttle gates were somewhat awkward to use, particularly at the 80 percent position. A fingertip, pull-to-operate configuration such as in the T-38 would be more convenient. The throttle friction knobs were found to be too large and occupied a space that could possibly have been used for the throttle trim switches. These latter items need to be more readily accessible to the pilot.

d. Miscellaneous Functions - Description and Operational Experience

The remaining portions of the cockpit and panel arrangement were utilized to support a multiplicity of aircraft functions. Most were satisfactory as designed; however, modifications in the following would greatly enhance the utility of the cockpit.

(1) Nozzle Position Indicator--Perhaps this was the most controversial item on the panel. Conceptually the arrangement was to indicate thrust vector angle at the top and nozzle angle at the bottom of the gage. This philosophy was predicated

on the argument that the pilot desires to know what is happening to the aircraft and is more interested in the thrust than the nozzle position. The instrument face was selected to permit an evaluation of either arrangement. Due to an installation problem, the needle was reversed for a period of time and then corrected. Also a profile view of the airplane was added to the face with significant improvements in readability. Though only utilized for a short period of time, it was concluded that nozzle position information was most valuable. As a result of rapid airplane response to nozzle angle changes, a more prominent location should also be used.

(2) Engaging Controller--This unit is slightly inaccessible due to its position forward of the throttles. Though available to both pilots, it needs to be relocated.

(3) Master Caution Warning Lights and Panel--Though the "Tee" shape of the panel was unique, the overall arrangement was satisfactory. The intensity of the main caution light could be increased.

(4) Condition Lights--Conceptually these lights were to provide an illuminated, check-off list type of reminder to the pilot. As such, the wording was negative. For example, "DIVERTER NOT DN" as a reminder in VTOL flight meant the diverter position was aft. Consequently, some confusion became inevitable. Reducing the number of lights and changing it to a form of position indication would be beneficial.

(5) Flap Switch--Flap system design was predicated on a two-position, quick-acting system; therefore, a center-off, momentary-hold switch was selected. The time required for flap actuation was sufficiently long (8 seconds) to make this type of control unsatisfactory.

(6) Electrical Power System Gages and Controls--As a result of an overall airplane objective to protect the integrity of the flight control system and its power supply, dc and ac voltmeters and frequency meters had been incorporated into the panel design to enable electrical system operation to be monitored at any desired time. With the reliability of the basic system established, this equipment would be deleted and made a part of the flight control line test unit. The basic system switches should be of the lever-lok type.

(7) Oxygen Pressure Gage--The particular indicator is too small to be quickly read and should be increased in size.

7. HYDRAULIC SYSTEM

a. Description

The hydraulic system in Figure 9 consists of two essentially identical subsystems powered by an engine-driven variable displacement pump on each cruise engine. Differing only in that the No. 2 system has the utility or non-dualled functions of gear, flap, and exhaust door actuation, each system utilizes self-pressurizing fluid reservoirs located over the aft fuel tank. Capacity of each reservoir is 160 cubic inches. Each system operates normally under a pressure of 3000 psig, and relief valves open at 3500 psig (full-flow rating) to prevent damage to the lines and the equipment should the pump displacement compensator fail.

Both of the hydraulic pumps are of the constant pressure, variable displacement type whereby the flow of fluid (gallons per minute) through each system will vary with engine speed but not the pressure. As long as the engines are turning at idle rpm or greater, the operation of all hydraulically powered systems is invariant. The tandem powered flight control, throttle-nozzle and diverter actuators allow operation at reduced loads in the event of a single hydraulic system failure. Emergency flap and gear operation is provided by air-pressurized accumulators.

Operation of the system is automatic when the respective engines are started; however, the pilot has the option of de-pressurizing either hydraulic system through actuation of the switches on the instrument panel for check-out of system functions. No means are available for the pilot to correct hydraulic system failures, the dualled operation and accumulators providing sufficient power for all essential functions.

Because of the requirements placed on the primary flight control system and its proper operation, considerable emplasis was placed on minimizing and controlling fluid contamination. All system filters had a 2 micron nominal and 8 micron absolute rating. In addition to the pump pressure and suction line filters, in-line units were placed in the pressure line of each flight control actuator. These actuators also came with 40 micron strainers in each pressure port. As a design goal, system cleanliness was to be maintained at the Class III level recommended by SAE Committee A-6.

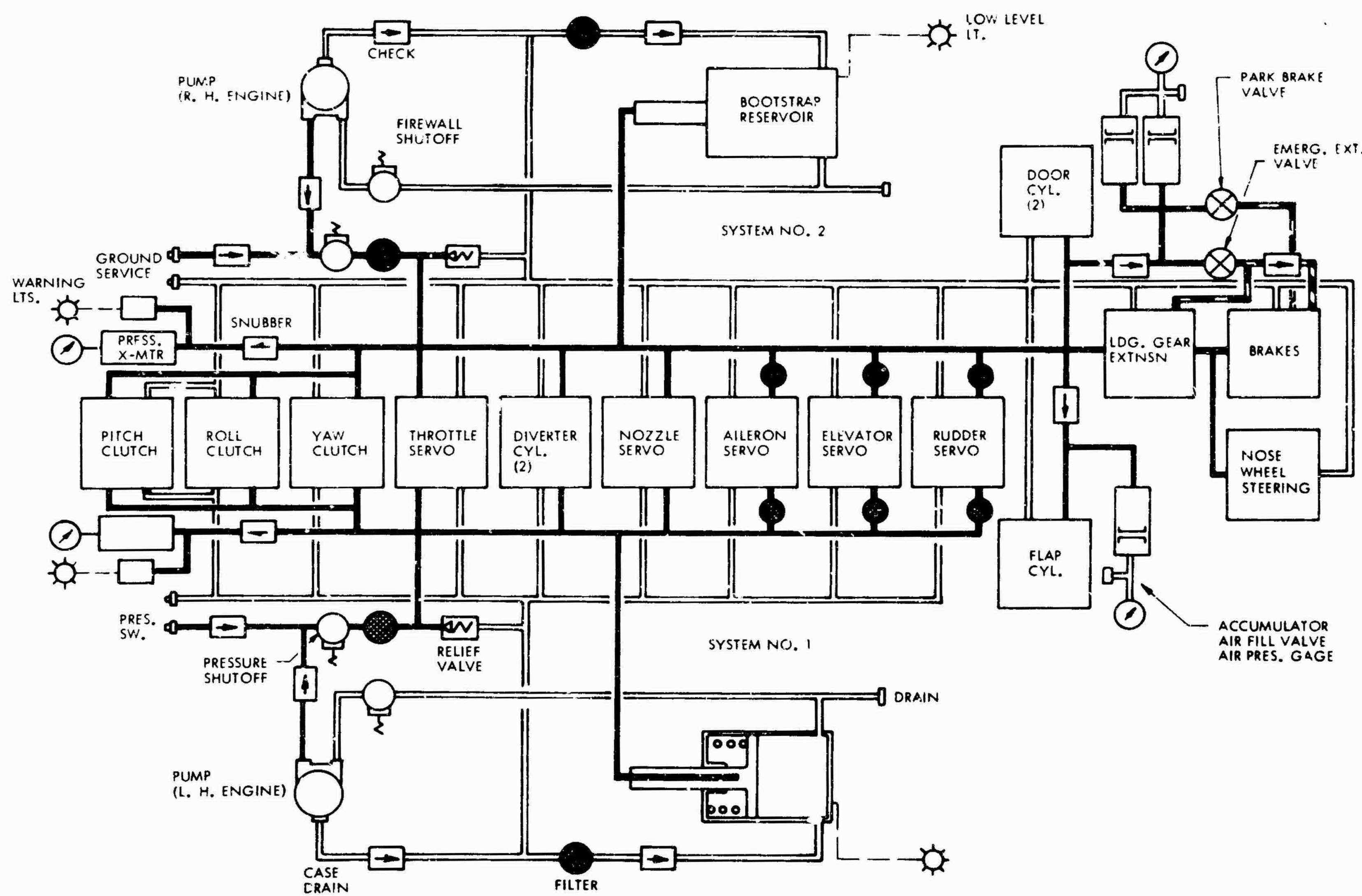


FIGURE 9 - HYDRAULIC SCHEMATIC

To minimize system heating effects with all engines operating in the VTOL mode, components and lines were concentrated over the forward and aft fuel tank. In addition a portion of the lines were routed through the fuel tank to effect heat transfer to the fuel and eliminate the necessity for coolers.

b. Operational Experience

Operational experience with the system disclosed a few minor developmental problems.

- During initial check-out of diverter valve actuation, a resonance or chatter was observed between the shuttle valves in the actuator and the hydraulic system proper. These valves had been installed to detect a pressurized No. 2 hydraulic system and select the active hydraulic system for powering the diverter valve. Relocation of the flow restrictors to a point downstream of the system shuttle valves eliminated the condition.

- Pressure fluctuations were observed on the cockpit instrument. The system was detuned by adjusting the pump compensators to a slightly greater pressure setting. An improvement was noted but the condition still persisted. Next, revisions in the hoses leading to the pressure transmitters were made with still further success but not completely eliminating the cause. It was concluded that there were vibrations of sufficient amplitude on the transmitter to cause the fluctuations even though the equipment was shock mounted. To eliminate the oscillation completely would necessitate moving the equipment or revising the mount.

- As originally planned, intermediate flap positions were secured by locking or trapping hydraulic fluid in the lines between the selector valve and the flap actuator. During the first flights, flap blow-back was quite pronounced and caused by fluid leaking back through the closed-center valve. The addition of an open-center selector valve and lock valve remedied the situation and the flaps would retain the selected position at all speeds within the flight envelope of the aircraft.

Perhaps the most recurrent problem arose with maintaining the desired level of cleanliness. During the preparations for first flight, the flushing and cleaning procedures brought the systems to the Class II level. However, continual maintenance action on the various sybsystems, as would be expected on a research type vehicle, brought this level up. System and servo filters were changed with some improvement in the level. As aircraft operating time increased, it became evident that the most substantial improvements in cleanliness came wher the fluid was circulated through the vehicle's system filters and by keeping the lines and components together. This latter technique was difficult to implement for the reasons noted earlier. Even though some progress was made in this area, consistency of cleanliness was never attained. Experience indicated that some significant improvements could be made through the addition of quick-disconnect couplings on those items of equipment removed frequently and a return line filter at the ground connections.

8. LANDING GEAR

a. Description

The general configuration of the landing gear is shown in Figure 1. Basically the XV-4B has a tricycle arrangement with a single wheel and tire attached to direct-acting, telescoping oleo-pneumatic shock absorbers. Each main gear retracts forward and has a single disc, high energy brake unit integrally attached to the wheel. The nose gear retracts to the rear and is fitted with a steering actuator and damper. All struts are hydraulically retracted through folding of the drag braces.

Main gear struts are trunnion mounted with the leg inclined 7° to the vertical in the front projection and inclined 10° forward in the side elevation. This configuration enables the trunnions to be mounted on the rear wing beam, allowing the gear to retract in a forward manner for the maximum free fall capability, and place the tire and wheel as far as practical from the exhaust cone of the lift engines. Based on this philosophy, the brakes mounted on an offset axle were installed on the outboard side of each wheel to reduce heating effects on the brake disc and hydraulic lines. Strut construction was conventional utilizing an aluminum forging for the strut housing and high strength steel for the piston and axle. Total strut deflection was 16.88 in. with an overall compression ratio of 8.6:1. Downlocks were incorporated into the drag brace assembly and the separate uplock was located in the lower surface of the wing box beam.

Similar in construction to that of the main gear, the nose strut was trunnion mounted in the forward fuselage with a lateral offset to the left of 7.65 in. Both up and downlocks were integral with the drag brace. Strut deflection was 8.50 in. with a compression ratio of 12.6:1.

Strut structural design was based on the cirteria described in Section II. Metering pin design was optimized for the complete load spectrum with a maximum air pressure of 655 and 2180 psig obtained on the nose and main struts respectively during the load test program. Shock absorber efficiency was 78 and 72 percent under these test conditions.

Each strut was fitted with a single, pneumatic tire of the characteristics noted.

	Nose Wheel	Main Wheel
Type	VII/Tubeless	VII/Tubeless
Size	16 x 4.4	20 x 4.4
Rim Diameter, in.	8.0	12.0
Ply Rating	8	12
Inflation Pressure, psig	125	215
Static load, lb.	2600	5200
Max. Speed, mph	200	200

The nose wheel tire was of conventional construction that had previously been qualified to a limit speed of 160 mph. To obtain the rating shown, a limited airworthiness test consisting of 20 taxi-takeoff-landing-taxi cycles was performed.

A variety of main wheel tires were used. During all of the taxi and flight testing, conventional black tires identical to those used on the T-38/F5 aircraft were used. To increase the speed margin, identically sized tires qualified to a speed of 250 mph for the SV-5P Lifting Body program were procured but never utilized. Lastly, a high-temperature tire constructed of materials developed for the B-70 and SR-71 aircraft was acquired and qualified for use on the aircraft.

This latter tire, hereinafter referred to as "silver", was fabricated from a proprietary compound of the B. F. Goodrich Co. and would permit it to withstand ambient temperatures of 375° without deleterious effects. This was approximately 100° F greater than the conventional black rubber tires. The silver tire passed 25 taxi-takeoff-landing-taxi cycles but only after the tread design had been modified. Initially the tread was made with square shoulders and excessive flexing was observed in this area. A more-rounded configuration successfully completed the testing without further difficulty. Both wheels were of cast magnesium construction consisting of two wheel halves, bolted together to permit assembly with the tubeless tires.

The brake assembly is of a single disc configuration with three sets of pads contacting the disc when the brakes are applied to generate the desired friction. Brake energy capability is:

Normal energy stop (20)	2,800,000 ft. lb.
Rejected take-off stop (1)	4,000,226 ft. lb.

These energy capabilities provided brakes that would decelerate the aircraft properly for a landing at anticipated weights and speeds. The rejected take-off (RTO) situation was a compromise between the requirements of a conventional aircraft and those of a VTOL flight research vehicle. For the latter case it was assumed that the brakes alone be capable of stopping a fully-loaded aircraft from a transition speed consistent with diverting engines between Phase I and II operations. An RTO stop from the conventional stall speed would be supplemented with the drogue chute. Pilot actuation of the brakes was through a two mode system: (1) the normal operation was boosted by hydraulic pressure from the No. 2 system, and (2) a manual brake cylinder provided emergency power. Parking brakes and back-up hydraulic pressure was provided by locking the brake pedal linkage and the emergency landing gear accumulators.

A nose wheel steering system is provided and derives its power from the No. 2 hydraulic system. Motion of the rudder pedals with the steering mode engaged ports hydraulic pressure in the direction until the strut follows-up the command input. The steering actuator also serves as a shimmy damper.

With only a single door on the nose gear, retraction and extension sequencing was very straightforward. Selection of the handle actuated a hydraulic solenoid-operated valve to the commanded position and hydraulic pressure performed the operation. In the UP position, up-lock switches de-energized the hydraulic system and removed all power from the landing gear and the mechanical uplocks prevent the gear from dropping or falling. In the DOWN position, pressure is maintained in addition to the downlocks as a back-up source. Emergency hydraulic power is provided to the landing gear system from two 75 cu. in. accumulators which provide the necessary pressure and flow to extend the gear at 200 kts.

b. Operational Experience

Operational experience with the landing gear disclosed several idiosyncrasies inherent in the arrangement that caused minor difficulties in servicing and maintenance. With the struts canted forward and outboard, lowering the vehicle from jacks had to be accomplished carefully or the struts would stabilize at different compressed lengths due to the friction at the lower bearing joint. Due to geometric and space limitations coupled with an unusually long piston stroke to obtain a low load factor, friction was greater than would be anticipated with a more conventional arrangement. Friction was again encountered during emergency nose gear actuation tests. The integral up and downlock links were determined to be extremely sensitive to the effects of friction. A 50 percent increase in the area of the nose gear actuator was required to achieve the desired extension speeds.

The brakes caused some operational problems particularly during initial taxi tests. To provide adequate braking energy, the design inherently had a high torque capability which made them extremely sensitive to pedal forces. The result was a jerking motion of the airplane when taxiing at low speeds making it difficult to maneuver without nose wheel steering. The brake disc size represents approximately the maximum amount of energy that can be absorbed in a single disc. Disc temperatures in excess of 1000°F were experienced and resulted in disc distortion and subsequent grabbing brakes unless care was taken to cool them properly. Thermal distortion grooves or slots were added to rectify this condition and evaluated in dynomometer testing but not on the airplane. However, after the initial learning period passed, brake operation was satisfactory. Landings were made at consistently higher energies than the design values without unusual problems. In addition a rejected take-off at approximately 12,000 lbs. gross weight and a brake application speed of 140 kts. was made successfully. The energy for this stop was 50 percent in excess of the design value.

The major brake problem was in the location of the unit itself. These had been mounted on the outboard side of the wheel necessitating removal each time a tire was changed. Reservicing and bleeding of the brake was then required. This was somewhat time consuming as well as opening the hydraulic system to possible contamination.

From the inception of the aircraft development program, the environment to be experienced by the tires was anticipated to be more severe than that of conventional aircraft. In particular, the close spacing of the tires to the lift engine exhaust would result in elevated temperatures in the tire. No data was available upon which a specific tire design could be based for the XV-4B. For this reason, two parallel courses of action were taken: (1) a high temperature tire was procured based on existing B-70/SR-71 technology; and (2) a test program was initiated to define tire temperatures on the actual aircraft.

During one of the lift engine run periods at the Inverted Telescope temperatures were measured by means of alumel-chromel thermocouples imbedded into the tire carcass at varying locations and depths. Coupled to a direct reading recorder, a continuous record of time vs temperature could be obtained and hot spots evaluated.

The initial run was made with the tire 7 feet from the ground plane. With all six engines operating at full power, a maximum stable tire temperature of 130 °F was reached. The aircraft was then moved to a position 5 inches from the ground.

During the first test period with the swivel nozzles in the full aft position, engines nos. 2, 1, 3, 4, 5 and 6 were started in a nine minute period. The nos. 1 and 2 engines were at 85 percent rpm (EGT = 900°F) to obtain sufficient bleed air pressure to start the remaining four engines. Temperatures noted prior to shutdown were:

No.		Temperature °F	Temperature Change Rate °F/min.
1.	Sidewall, surface	325	29
2.	Sidewall, 3/8 in. deep	295	34

The second test segment saw all engines started in five minutes and then brought to 80 percent rpm (EGT = 850°F) four minutes later. Temperatures of 460 and 415°F were recorded on these same thermocouples. During subsequent testing the thermocouples separated from the tire due to high exhaust gas velocity.

Sufficient information was acquired with which to reach some tentative conclusions:

- o Relatively high (450°F) temperatures were reached during periods of engine idle operation. At this time, EGT is approximately 1000°F and a minimum of secondary cooling air is induced to flow along the tire face.
- o Tire heating is relatively uniform throughout the carcass and concentrated in the area closest to the ground.
- o Tire rubber reversion occurred locally on the silver tire indicating excessive temperatures.
- o Shielding of the tire was required to operate the aircraft from a flat surface as opposed to any type of grid or grate structure.

Prior to the preceding tests a shield design had been conceived. It consisted of a metal pants which covered the lower portion of the main landing gear struts and extended to within 3 inches of the ground. An asbestos rubber curtain was used to close this gap. Between the shield and the tire, a protective rotating shield formed from a black tire carcass was attached to the wheel rim.

The next series of tests was performed with the airplane on the ground with encouraging results. With a single engine running (No. 1 at 90 percent rpm), the intermediate shield temperature was only 130°F after 12 minutes of engine running. Tire rib temperature was 184°F. With all six engines at idle, these temperatures rose to 303 and 395°F respectively.

With this data, an improved version of the shield was designed which utilized a brush rather than the asbestos rubber and a silicon rubber for the intermediate device. Modifications were made to the nacelle which permitted the complete shield to be retracted into the nacelle. This configuration had not been tested prior to the conclusion of the program.

Concurrent with the above testing, additional studies had been made to evaluate:

(1) the use of an inert gas such as helium or nitrogen to delay the increase in tire pressure with rising temperatures and (2) coating the tire with an insulating or ablative coating. The use of inert gas would not prove to be effective since the test data shows that heat transfer is nearly instantaneous through the carcass. Experiments with tire coatings disclosed only a minimal increase in the time that it takes for the tire to reach rubber reversion temperatures. In addition, the aging or curing process for the coatings results in a degradation of tire carcass strength.

It would appear that the problem of excessive tire temperatures requires a significant amount of development work. The tire companies have no ready answer. A promising development area is the heat shield and its refinement.

9. ESCAPE SYSTEM

a. Description

A Douglas Aircraft Company, Escapac ID-3, rocket-catapult ejection seat is provided for both crew members in the event of an emergency necessitating them leaving the cockpit quickly. A medium impulse rocket/catapult and ballistic opening parachute provides for safe egress under zero-speed and zero altitude conditions under a variety of attitude and sink rate combinations, Reference 13. Seat stabilization and seat-man separation is accomplished through a snubber/DART (Directional Automatic Realignment of Trajectory) mechanism which compensates for anthropomorphic variations in human center of gravity and inertias. A ballistic, seat-sequencing system enables either crewman to initiate the escape sequence for both aircraft occupants with the left hand man leaving the aircraft one-half second after the first. All seat functions are designed to minimize the time required to achieve full parachute inflation which was consistently demonstrated in the test program to be less than 4.5 seconds.

b. System Development and Operational Experience

Initial escape system selection for the XV-4B was based on the development of a light-weight, Escapac ID ejection seat for the U.S. Navy/Bell X-22A VTOL aircraft. To this basic system was added a basic DART stabilization device as a means for upgrading performance and insuring consistent trajectories. The first test firings demonstrated that somewhat marginal operation was attained with this system which relied on aerodynamic deployment and inflation of the NB-5, 26 foot conical parachute.

During the course of this program, escape system developments and technology which were undergoing final qualification testing indicated that significant improvements could be incorporated into the system with a minimal weight penalty and without altering the XV-4B seat or cockpit geometry. A contract change was negotiated to provide these improvements which included the incorporation of the Stencel/NB-11, 28 foot, flat circular ballistically spread parachute. With this change deployment and canopy inflation was positive and did not rely on air velocity for its operation. A reduction of nearly two seconds in the time to achieve full parachute inflation resulted. Further improvements in seat/man separation were added by snubbing seat motion with equipment added to the DART subsystem.

The remainder of the test program was conducted using this basic system and is described in Section VII.

Note: A live ejection took place on 14 March 1969, from the XV-4B. All aspects of the ejection and escape system operation were normal and satisfactory; however, the toe of the left boot was severely cut. Due to the violent maneuver the aircraft was in at the time of egress, it was concluded that the toe most probably hit the canopy frame which still had portions of the plexiglass rigidly attached.

SECTION IV

AERODYNAMIC CHARACTERISTICS

This section summarizes the basic aerodynamic characteristics of the XV-4B Hummingbird II. Reference 14 is a more detailed aerodynamic description in terms of basic aerodynamic data, aircraft performance, and aircraft stability and control with data provided for all modes of flight including hover, transition, and the conventional flight regime. Propulsion and reaction control characteristics are also contained in this reference primarily for the purpose of clarity.

1. GENERAL

The total aerodynamic characteristics are divided into components which may be combined to describe the aircraft in the various phases of flight. The basic aerodynamic characteristics generated by airspeed are applicable to all phases of flight from near hover to maximum speed in conventional flight. This is the total component in the conventional flight regime unless the direct engine net thrust terms are considered as a portion of the total aerodynamic characteristics which is contrary to normal procedures. For the hover and transition phases, the additional terms generated by the direct engine thrust must be added to the basic aerodynamic characteristics. These terms consist of the interference effects generated by the exhaust of the lifting jet engines, the interference effects of the engine inlet airflow which are small and are combined with the exit interference effects, the direct components of the engine inlet airflow, and the direct components of the lifting engine thrust which are not normally considered as aerodynamic components for the conventional aircraft but are the major components for a VTOL aircraft at or near hover.

a. Limits of Data

Since small maneuvering velocities about true hover induce a full spectrum of angles, the aerodynamic characteristics are evaluated for sideslip angles up to 90 degrees and angles of attack up to 48 degrees which is well beyond the stable deep stall. Although the XV-4B Hummingbird II configuration is characterized by pitch-up at the normal stall with a stable deep stall characteristic at high angles of attack, sufficient aerodynamic control is available to return to the normal unstalled angle of attack range. The speed range over which the aerodynamic characteristics are evaluated extends from small rearward and side velocities up to somewhat beyond the maximum operational speed of 260 knots or a Mach number of 0.53.

b. Data Sources

The basic aerodynamic data are primarily based on low speed wind tunnel test data. Both unpowered and powered test configurations using ejector propulsion units were utilized as a basis for deriving the basic, full-scale aircraft characteristics. Where necessary, the low-speed wind-tunnel data are augmented by data estimated in the conventional manner. Extreme angular ranges and power plant interference effects were emphasized during the wind tunnel programs. The results of these wind tunnel tests are summarized in Section VII and detailed in References 15, 16, and 17.

c. Definition of Flight Phases

The aircraft operates in one of the four conditions defined as Phases I through IV. Although the speed range for these phases have a large overlap under certain conditions, the normal range is defined as follows.

Phase I - The four lift engines are operating and the two cruise engines are operating in the diverted or "lifting" mode. The speed range is normally the low speed portion of transition from zero to a nominal forward speed. The Flight Mode switch is in the "VERT" position and the Diverter Valve switch is in the "DOWN" position.

Phase II - This phase is identical to Phase I except the Diverter Valve switch is in the "AFT" position causing the two cruise engines to be in the thrusting or cruise mode. The speed range is normally the high speed portion of transition.

Phase III - This is the region where the lift engines are shut down but are windmilling due to ram air because the exit doors are open. This phase occurs immediately prior to conversion in the transition or take-off flight region and immediately after reconversion in the retransition or landing phase.

Phase IV - This is the conventional flight mode; therefore, the Flight Mode switch is placed in the "CONV" position. The four lift engines are off, and the exit doors are closed.

2. BASIC AERODYNAMIC DATA

The basic aerodynamic data for the XV-4B which are applicable to all flight phases are discussed under this heading. Additional terms which must be included to describe the total aerodynamic characteristics in each of the VTOL flight phases are discussed elsewhere.

a. Lift Characteristics

The aerodynamic lift characteristics of the XV-4B are as would be expected for any typical aircraft configuration with an aspect ratio six wing and with a large percentage of the theoretical wing area buried in the fuselage and in the nacelles mounted on the fuselage sides. Figure 10 shows typical lift curves with neutral elevator for flaps both extended 40 degrees and retracted. The curves extend into the deep stall range up to an angle of attack of 48 degrees. With the center of gravity at a typical position of 10 percent mean aerodynamic chord, the maximum trimmed lift coefficients are 1.24 for the flaps retracted and 1.74 for the flaps fully extended to 40 degrees.

b. Longitudinal Aerodynamics

For the nominal center of gravity location of 10 percent of the mean aerodynamic chord, Figure 11 presents, as typical, the tail-on pitching moment characteristics throughout the normal operating range of angle of attack on into the stall and deep stall regimes. The curves shown for the two configurations, flaps retracted and extended 40 degrees, are for the elevator neutral. Although the T-tail configuration coupled with the large fuselage and nacelles causes pitch up into a stable deep stall situation, the elevator power coupled with the increasing effectiveness of the fuselage mounted strakes provides an adequate margin to recover from any inadvertent entry into the deep stall regime. In spite of the pitch up characteristic, the T-tail is utilized due to its high effectiveness in the normal angle of attack range and its relative freedom from induced effects in the hover and transition regimes which are particularly critical near the ground. For the least stable Phase IV configuration (in ground effect), a generous tail-on static margin of 13 percent exists about the 10 percent mean aerodynamic chord point.

c. Drag Characteristics

Figure 12 presents the trimmed drag polar for the Phase IV cruise configuration upon which the conventional flight performance data are based and from which the aerodynamic drag is obtained for all phases of flight. The drag data are based upon the low

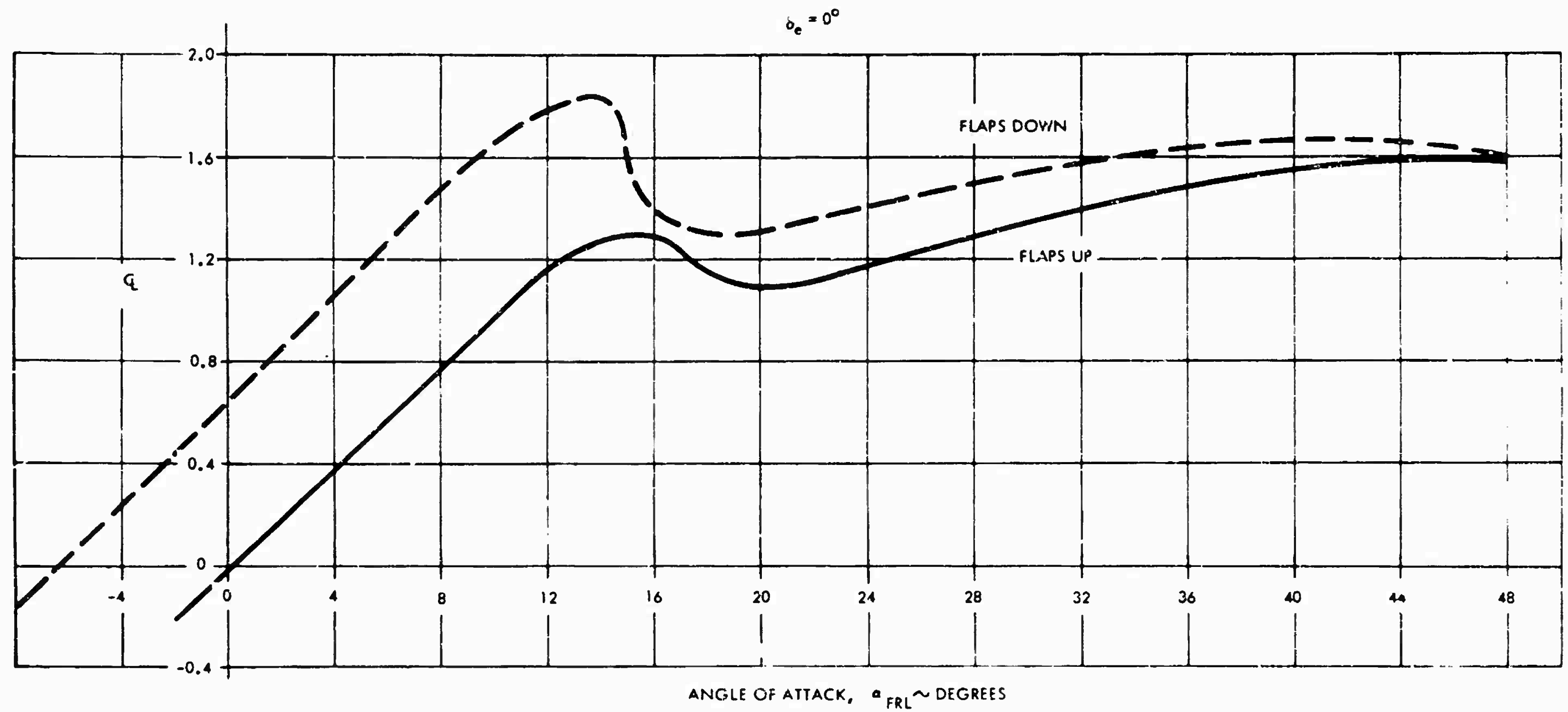


FIGURE 10 - LIFT CHARACTERISTICS INCLUDING DEEP STALL

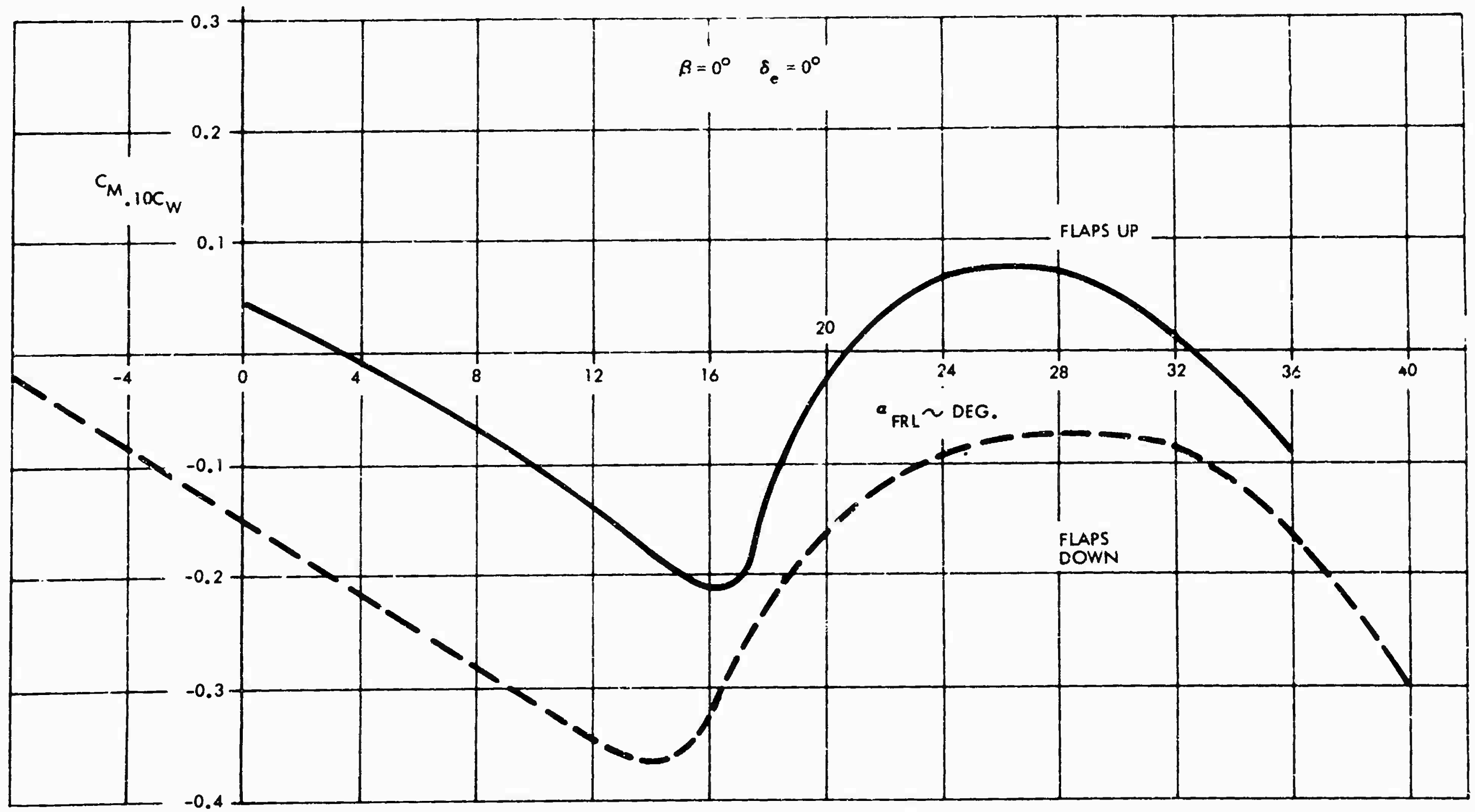


FIGURE 11 - AERODYNAMIC PITCHING MOMENT COEFFICIENT INCLUDING DEEP STALL

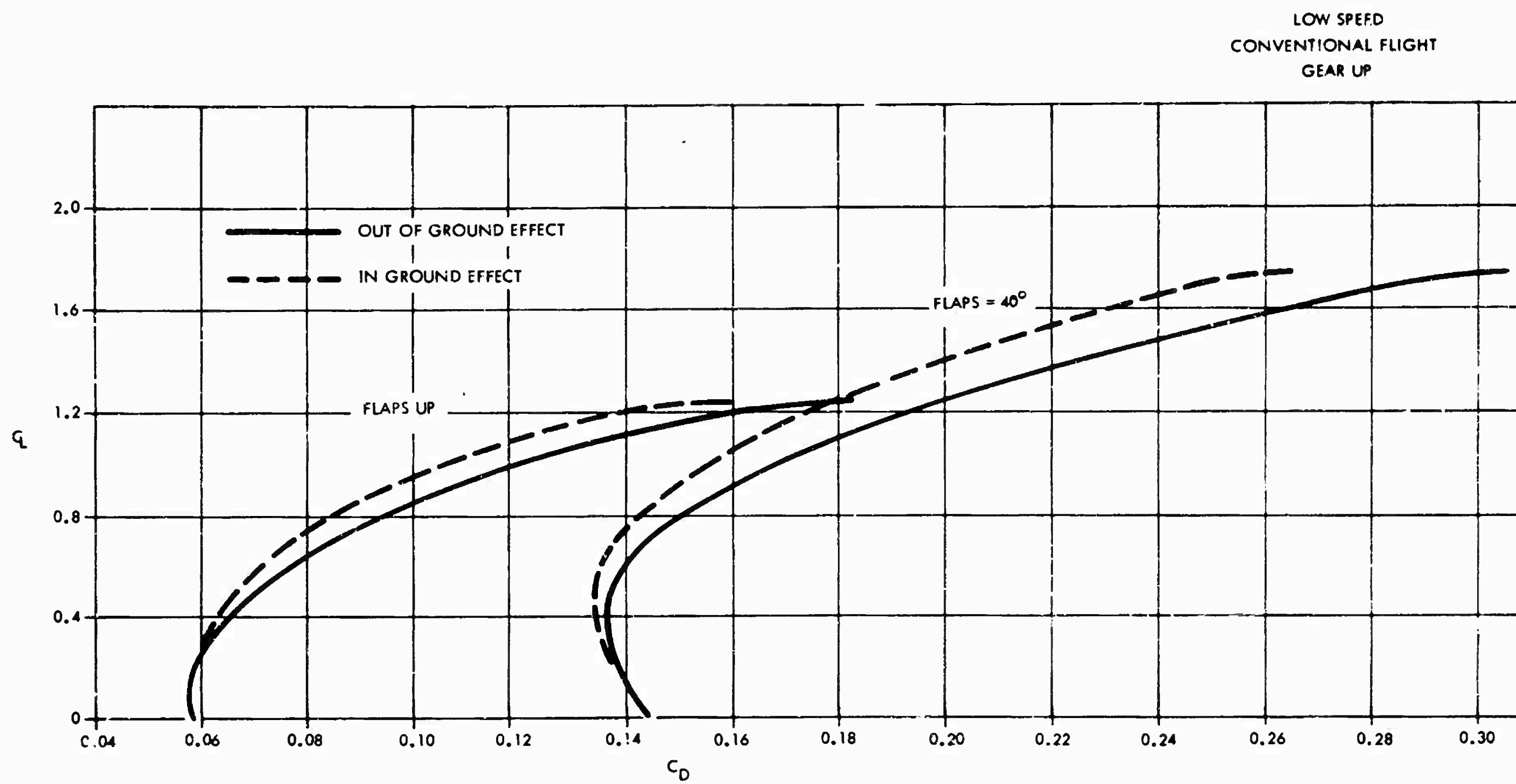


FIGURE 12 - DRAG POLARS

speed wind tunnel tests data as corrected by a normal Reynolds number drag relief on skin friction drag level, plus the drag associated with the configurational differences existing between the tested model and the full-scale airplane, and a drag increment associated with the anticipated roughness of the flight article.

d. Lateral-Directional Aerodynamics

Low speed lateral-directional characteristics are typically presented in Figure 13 in the form of directional stability (C_n vs. β) and dihedral effect (C_ℓ vs β) for sideslip angles (β's) up to 90 degrees which may be expected for maneuvering about a hover point in Phase I. Although wind tunnel data are quite sketchy at the higher β's, these curves are required and are based upon the available wind tunnel data. The estimated constant level of C_ℓ for β's in excess of 18 to 20 degrees causes some small concern, but no real problems, in the low speed Phase I regime where large β's are realistic.

e. Aerodynamic Controls

The effectiveness of the basic aerodynamic control surfaces are estimated. These surfaces are designed primarily for the conventional flight regime (Phase IV) and are completely ineffective at a true hover. The reaction controls supplied for hover requirements are progressively supplemented by the aerodynamic controls as speed is increased in Phases I and II of transition. The build up in interference effects generated by the engine lifting thrust and inlet airflow with increasing airspeed is compensated by the build up in aerodynamic controls with increasing airspeed. Minimum maneuver control margin occurs at low transition speeds approaching hover.

f. Airloads Data

Estimated airloads data include wing spanwise loading and hinge moments of the various doors and landing gear in addition to the normal aerodynamic characteristics supplied for the aircraft which are also required for basic loads determinations. All of the component data required for a complete loads analysis are supplied.

3. PERFORMANCE

This discussion of the performance of the XV-4B covers the complete flight regime from hover through transition up to maximum speed in conventional flight including conventional take-off and landing characteristics. For all operations the angle-of-attack

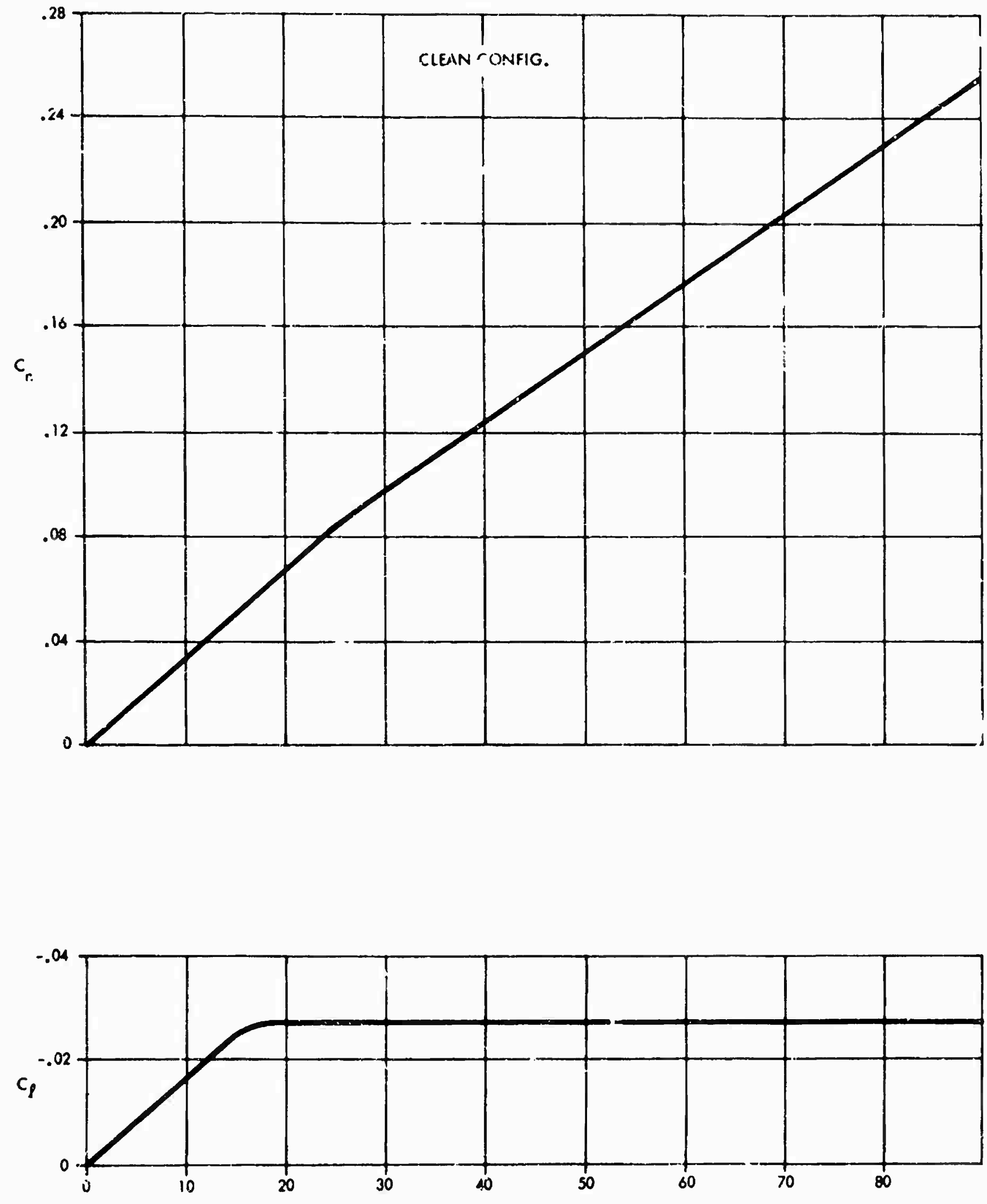


FIGURE 13 - EFFECT OF SIDESLIP ON YAW & ROLLING MOMENT COEFFICIENT

range is restricted to that limited by -10 degrees and +12 degrees although the aircraft capability exceeds these limits. The lower limit of -10 degrees is imposed due to the lack of aerodynamic data in the large negative angle-of-attack range. The upper limit of +12 degrees is imposed due to the aerodynamic pitch-up tendency at the stall predicted for the configuration based on small scale wind tunnel tests. This upper limit may be exceeded at the low Phase I transition speeds approaching hover where the aerodynamic generated forces and moments are negligibly low.

a. Phase I

At the design maximum VTOL weight of 12,800 pounds, the maximum hover time is 16.3 minutes with two crew members. For a thrust to weight ratio ($T/W_{T.O.}$) at take-off of 1.05, typical take-off limits of altitude and ambient temperature for the VTOL weight of 12,800 pounds are:

Sea Level at 103°F
1100 feet at 92°F
5000 feet at 32°F.

Alternately, three typical transitions may be accomplished with an additional hover time of 2.4 minutes. A typical transition consists of: two minutes of hover time at T/W = 1.05 allowance for engine start and warm-up, vertical take-off, and transition to forward flight; three minutes cruise at 200 knots with lift engines at idle, flaps down, and gear up; and one minute of hover time at T/W = 1.05 for retransition to hover and land vertically.

Essentially unlimited combinations of angles of attack and lift nozzle angles within the limits of the two parameters are available to the pilot for operation at all speeds in Phase I of transition. Operation between the limits of maximum take-off transition accelerations and maximum landing retransition decelerations is completely flexible. Either procedure may be reversed at any speed and unaccelerated flight is available at all Phase I speeds. Figure 14 is a typical take-off transition time history using combinations of parameters for maximum acceleration, and Figure 15 is a typical landing transition (retransition) time history using combinations of parameters for maximum deceleration. The take-off transition includes both Phase I and Phase II; whereas, the landing transition utilizes only Phase I.

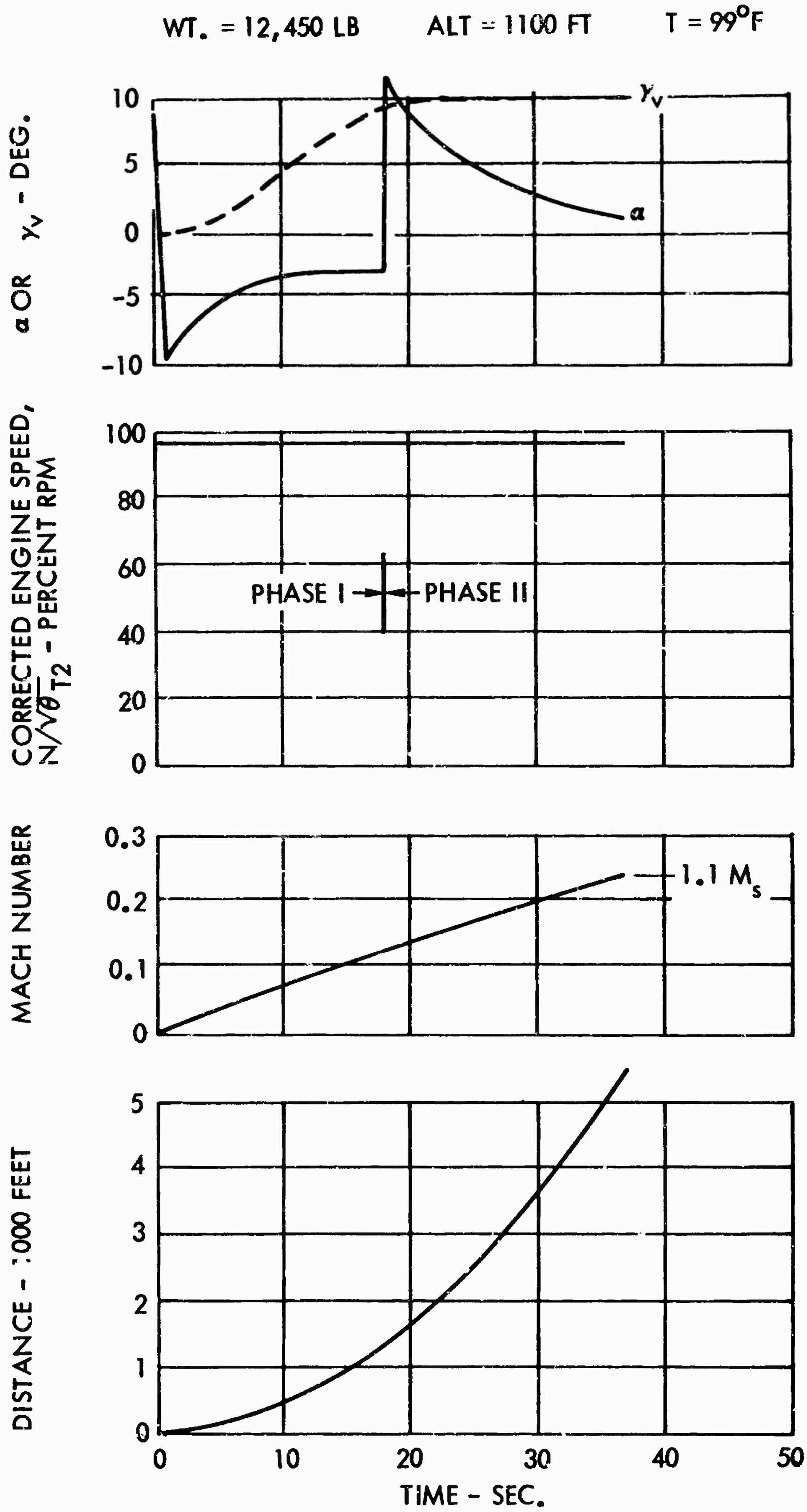


FIGURE 14 - TAKE-OFF TRANSITION TIME HISTORY FOR MAXIMUM ACCELERATION

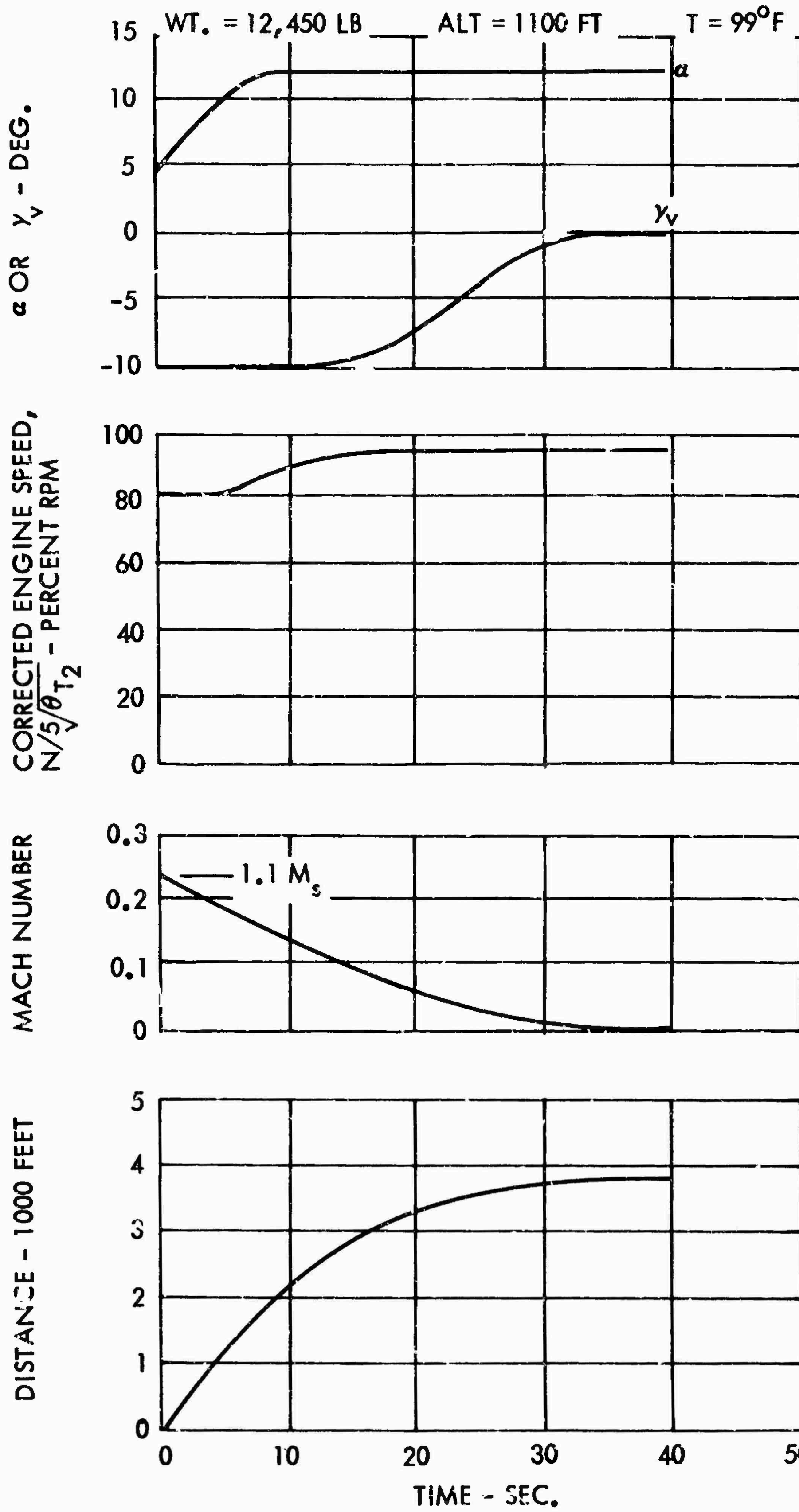


FIGURE 15 - LANDING TRANSITION TIME HISTORY FOR MAXIMUM DECELERATION

b. Phase II

In general, Phase II is of interest only for steady state unaccelerated flight and for take-off transitions. Greater deceleration for retransition is available for the Phase I configuration; however, the Phase II configuration is mandatory at least momentarily following the starting of the lift engines. Similar to Phase I, the aircraft is completely flexible for operation in Phase II. As mentioned previously, Figure 14 shows a typical take-off transition time history for maximum acceleration.

c. Phase III

Since Phase III is only a transient variation of Phase II or possibly Phase IV, Phase III considerations are of limited interest and have not been investigated extensively. However, as a special case of Phases II and IV, acceptable performance characteristics are available to perform both transitions and retransitions as well as steady state unaccelerated flight.

d. Phase IV

When considered as a portion of either a transition or a retransition, Phase IV operations are limited to inflight conditions between stall speed and structural design speed. When considered as normal conventional flight, Phase IV operations include the normal airport performance items as well as normal cruise and climb considerations. The most critical item for conventional airport take-off performance is the single engine rate-of-climb following lift-off with the flaps extended and the landing gear extended. In order to maintain a minimum rate-of-climb of 100 feet per minute under these conditions, the allowable flap deflections must be restricted for the design take-off weight of 12,000 pounds even for standard day at sea level conditions. As the temperature and altitude conditions for take-off increase, the allowable flap deflection continually decreases to the optimum of 10 degrees. Beyond this condition, the allowable take-off weight must be progressively restricted. When the take-off flap and weight limits are recognized, the military take-off field lengths are quite reasonable even for operations up to an altitude of 5000 feet for either standard or hot day temperatures with a maximum length of about 9000 feet.

The normal total landing distances over a 50 foot obstacle are consistent with the military take-off field lengths even when the drogue chute is not deployed; however, the actual ground roll distances are significantly greater for the landing. With the drogue chute deployed after touchdown, the landing ground roll distances are comparable with those for take-off. The brakes are not normally used at speeds above 122 knots. A number of landings with various malfunctions such as no braking or flaps inoperative in the retracted position were investigated. With the drogue chute operational, all of these landing distances were less than that available at Dobbins Air Force Base. However, with a drogue chute failure in conjunction with one of the other malfunctions, the landing distances are considerably in excess of that available at Dobbins.

Figures 16 and 17 give specific range summaries for twin engine and single engine operation, respectively, for cruise at the maximum operational speed of 260 knots as further limited by 0.53 Mach number. Twin engine cruise at heavy weights is not possible at the operational limit altitude of 20,000 feet, and specific range is not shown above an altitude of 15,000 feet.

The limitations in speed and altitude imposed upon the XV-4B are presented in Figure 18. Limits in airspeed as defined by thrust required and available are far in excess of these values for the normal cruise configuration. By the same token the climb capability is outstanding.

4. STABILITY AND CONTROL

This discussion includes the stability and control characteristics of the XV-4B aircraft in all phases of flight. The operational limitations given in the introductory remarks of the performance section are also applicable here. The basic aerodynamic data discussed previously are supplemented by the interference effects and direct thrust generated terms to provide data for the transition regimes. In general, all discussions are applicable to the configuration with the center of gravity at the 10 percent mean aerodynamic chord unless indicated otherwise.

a. VTOL

Operation of the XV-4B from hover throughout transition is covered longitudinally and lateral-directionally. The powered interference effects operated by the lifting engine thrust in both Phases I and II are linear functions of the inverse of the square root of thrust coefficient ($C_T^{-1/2}$) and are various functions of angle of attack, sideslip angle, and lifting thrust exit angle. For all aerodynamic components, the interference

TWIN ENGINE OPERATION
STANDARD DAY
5% CONSERVATIVE FUEL FLOW
260 KEAS CRUISE AS LIMITED BY MACH 0.53

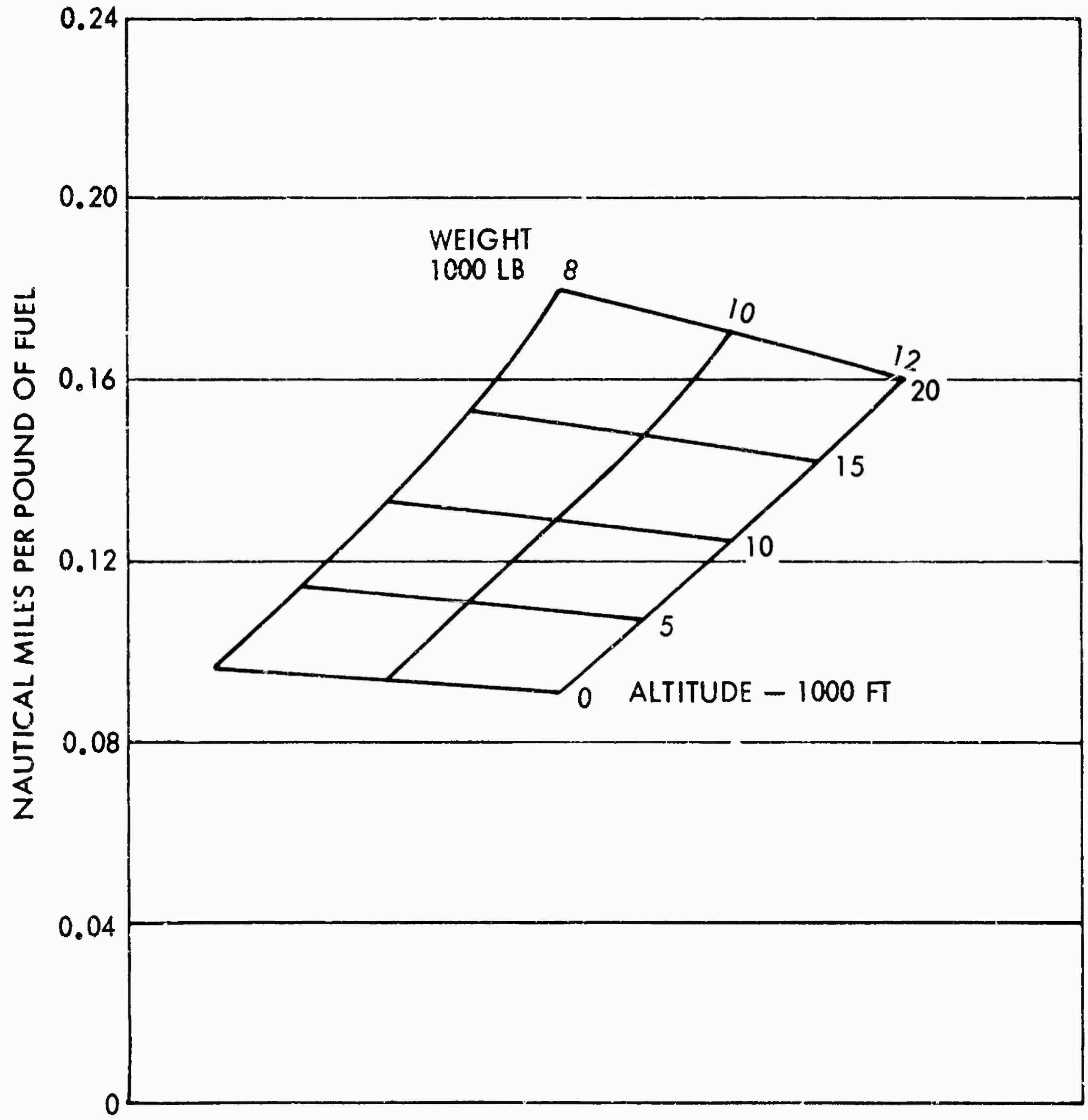


FIGURE 16 - SPECIFIC RANGE SUMMARY, TWIN ENGINE OPERATION

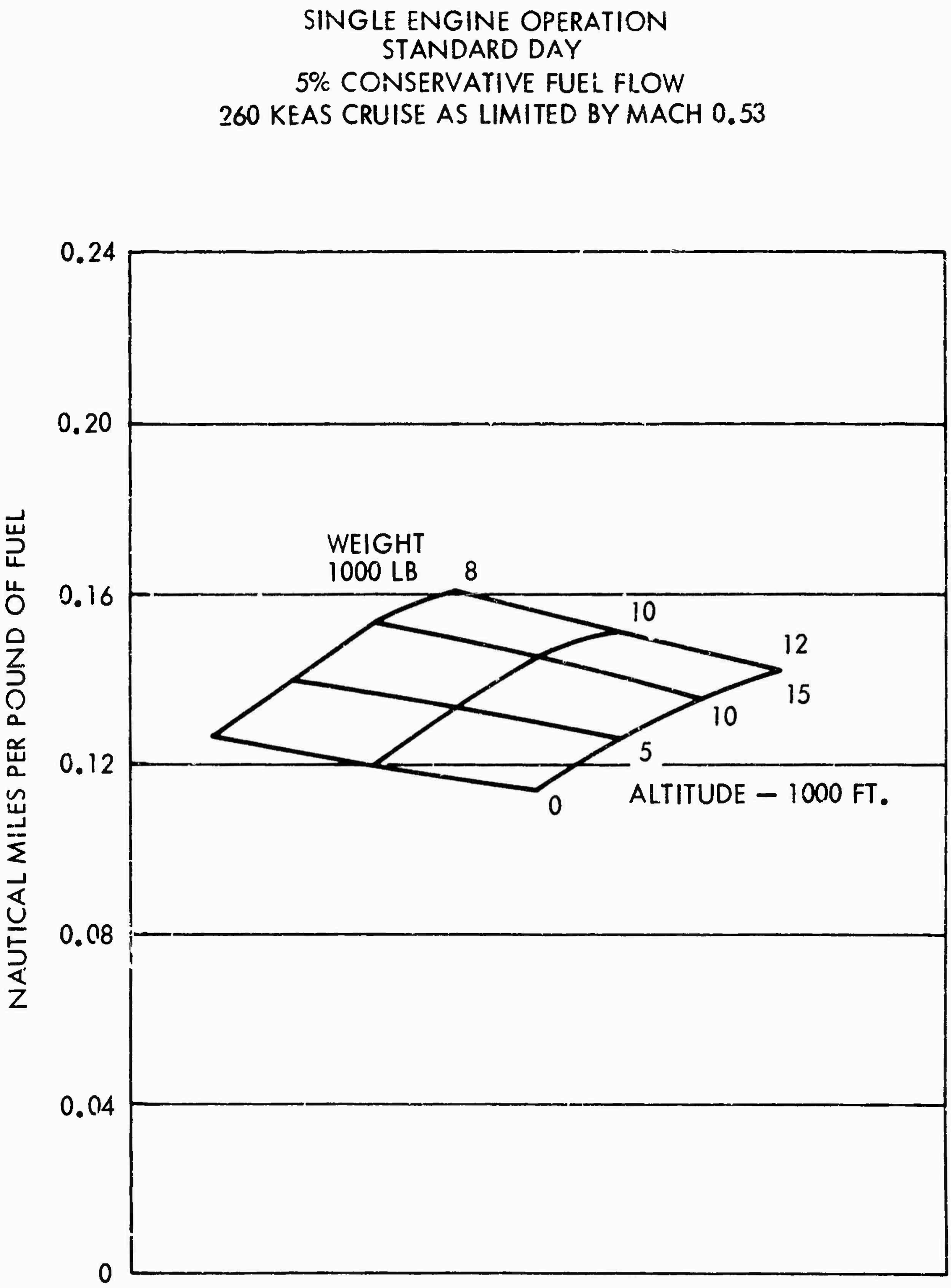


FIGURE 17 - SPECIFIC RANGE SUMMARY, SINGLE ENGINE OPERATION

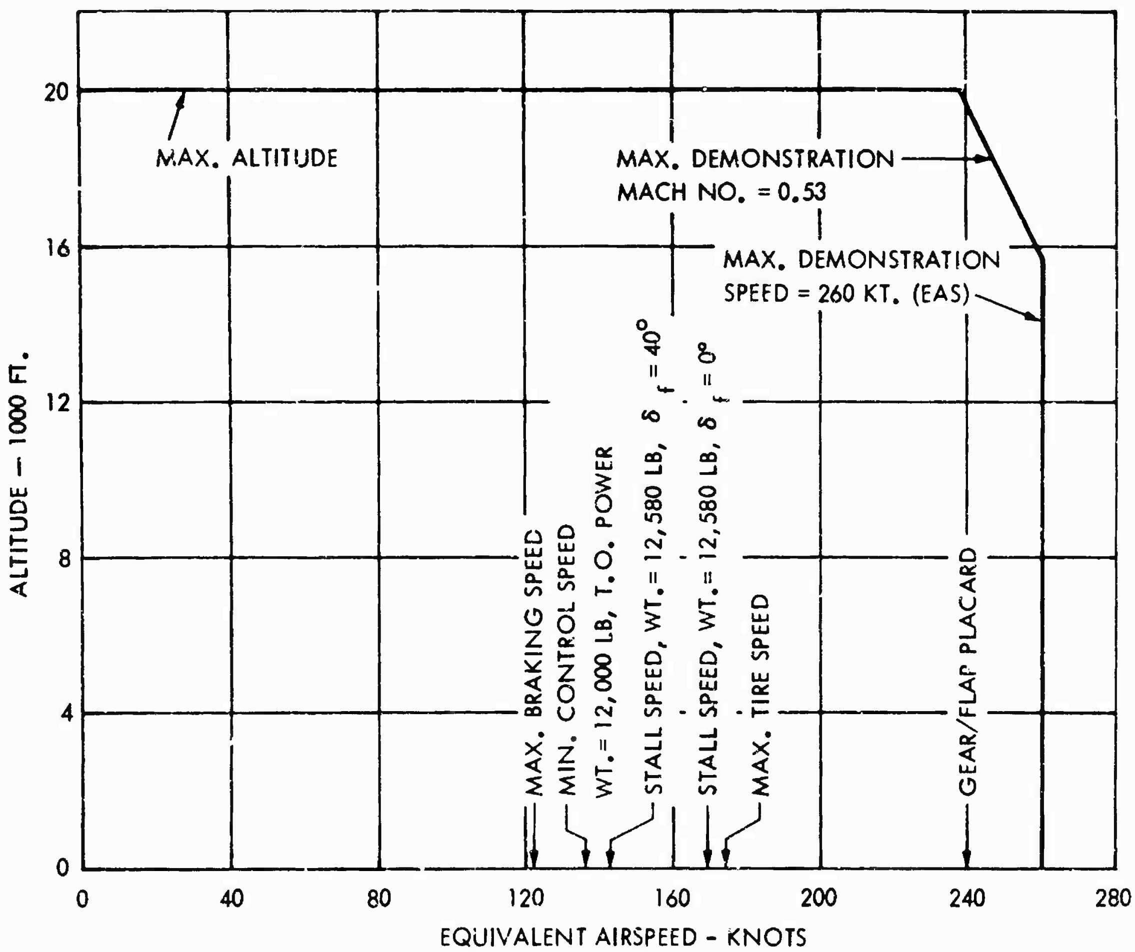


FIGURE 18 - SPEED & ALTITUDE LIMITS

effects are different functions for Phase I and Phase II. In addition for Phase I the thrust is the total of all six lift nozzles; for Phase II the thrust is the total of the four lift engine nozzles.

The minimum, single axis, control powers in still air hover are as follows between sea level and a pressure altitude of 5,000 feet:

Axis	Control Power - Rad/Sec^2
Roll	2.45
Pitch (Nose Down)	0.97
Yaw	1.06

With 50 percent of maximum control in two axes, the control available in the remaining axis is well in excess of 50 percent of the maximum single axis value. After trimming for the critical engine inoperative, the remaining control powers are quite impressive being at least 37.5 percent of the single axis values for all three modes simultaneously.

Analyses of the longitudinal control capability throughout the complete VTOL operational range from hover through Phases I and II of transition and retransition in the form of the pitching moment required and available reveal the following:

(1) The minimum margin between required and available occurs between hover and a speed of about 60 knots in Phase I depending upon angle of attack, lift thrust exit angle, all engine operation, critical engine inoperative, engine speed, and center of gravity.

(2) The most restrictive condition is at hover with one of the forward engines inoperative with a forward center of gravity where a minimum of 93 percent corrected engine RPM is required to maintain a selected 2000 foot pound maneuver margin is arbitrarily selected as a realistic lower limit in excess of all steady state trim requirements. Any SAS requirements must be supplied from this maneuver margin.

(3) Incremental lift nozzle deflection angle is an effective pitching moment generator.

(4) Engine speed exhibits a strong influence on maneuver margin.

(5) Reaction control is not available when all engines are below 80 percent RPM since the bleed valves close at this point.

(6) For a critical engine inoperative condition, the maneuver margin may be maximized by the proper selection of angle of attack and lift nozzle deflection angle.

The longitudinal static stability in Phases I and II of transition is shown in Figure 19 in the form of the slope of pitching moment with angle of attack. Like most VTOL configurations, the neutral stability at hover becomes unstable at low forward speed and then becomes stable with further increases in forward speed. As shown, the degree of instability is quite modest and presents no problems.

Figure 20 illustrates the relative control power required and the level of speed stability for both Phases I and II of transition by presenting the elevator deflections required to trim for steady state flight as a function of Mach number at a corrected weight of 12,580 pounds. Trim curves for variable lift nozzle deflection angles between the limits of +10 and -10 degrees in increments of 5 degrees are depicted. As shown, elevator deflections between -6 and +14 degrees, compared to a maximum of ± 30 degrees, provide longitudinal steady state trim with all engines operating at common RPM's for all combinations of variables.

The lateral-directional characteristics are evaluated by determining the trimmed steady sideslip parameters required for both phases of transition. All engines operating and a single engine inoperative cases show normally expected variations of bank angle, rudder deflection, and aileron deflection with sideslip angle. As discussed previously, the invariant estimate of the basic aerodynamic rolling moment coefficient (C_ℓ) with sideslip angle (β) at large β's causes a reduction in the aileron required to trim to sideslip angles much in excess of 15 to 20 degrees. Since this is predicted only at the higher sideslip angles in excess of those normally of interest, no real problem is envisioned.

b. Conventional Flight

Stability and control of the XV-4B in conventional flight covers operations in the normal wing borne flight regime from stall to maximum operational speed. Nose wheel lift-off speed considerations indicate that this phase of the longitudinal control is somewhat marginal. The maximum flap deflections must be limited for the more forward center of gravity locations if the nose wheel lift-off speeds are not to exceed the normal aircraft lift-off and climb speed of 1.2 times the stall speed. At the forward limit of 4 percent MAC, the flap deflection must be limited to approximately 18 degrees with no flap restriction required for center of gravity locations aft of 8.8 percent MAC.

PHASE I AND II
ALL ENGINES OPERATING

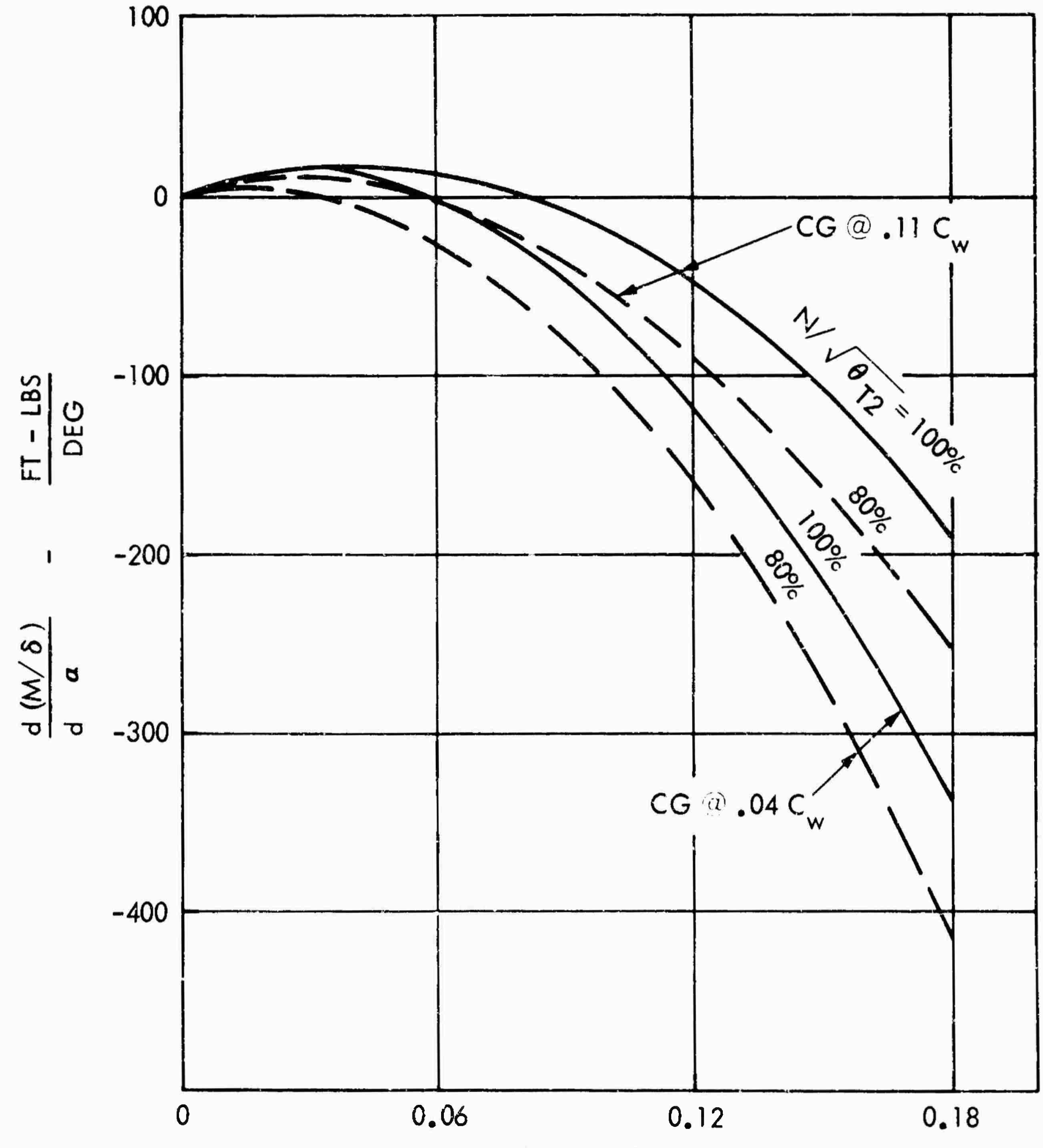


FIGURE 19 - ANGLE OF ATTACK STABILITY

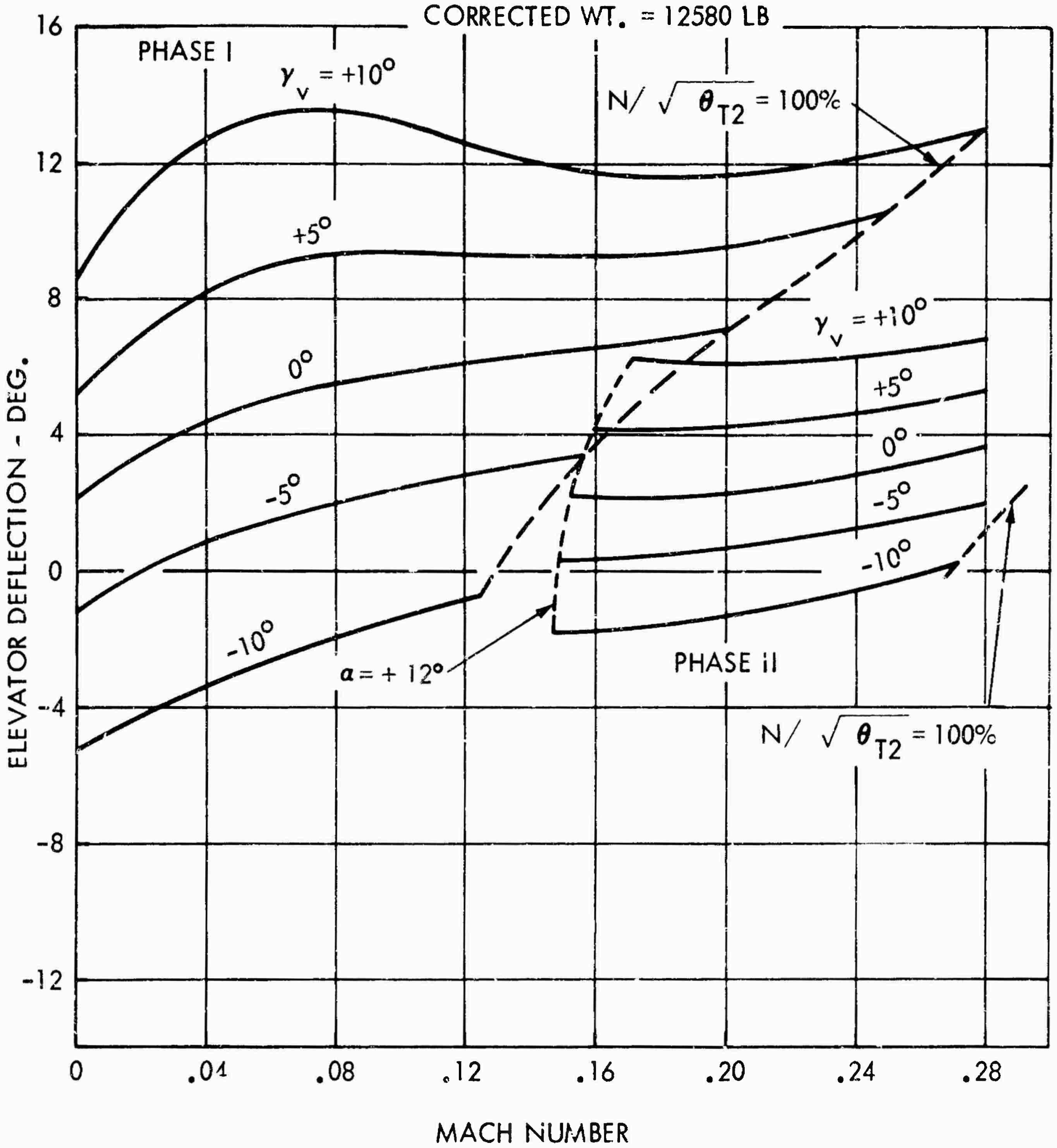


FIGURE 20 - ELEVATOR DEFLECTION REQUIRED FOR STEADY STATE FLIGHT

Positive speed stability in terms of both elevator position and stick force exists for all extremes of center of gravity and weight. Due to the light stick forces chosen for the fully powered control system, the speed stability almost becomes neutral at the higher structural design speeds which are well in excess of the maximum operational speed of 260 knots or 0.53 Mach number. The light stick force level was chosen for compatibility between the VTOL values and the conventional flight values.

When considering only the required elevator deflections, a reasonable level of maneuver stability is exhibited by the XV-4B; however, when combined with the light stick force characteristics of the powered control system, the stick forces required for maneuver load factors are quite low and do not meet the requirements of MIL-F8785 (ASG). Since extremely low stick forces are used in the VTOL mode based on pilot opinion from simulation experience and a large contrast between the stick forces for the VTOL and the conventional operational modes is not desirable, these low stick forces, as incorporated, are deemed to be desirable.

Figure 21 presents a typical maneuver envelope with the required trim elevator deflections superimposed for the XV-4B covering the complete structural design speed range although the aircraft is operationally limited to 260 KEAS or 0.53 Mach number. For the flaps up configuration at sea level, elevator deflections between the values of about -12 and +14 degrees provide trim over the complete maneuver envelope for all combinations of allowable weights and centers of gravity. For the flaps and landing gear down configuration, the corresponding elevator deflection limit requirements are about 0 and -20 degrees. For all cases the stick force per load factor decreases with increasing speed similar to the reduction in elevator deflection shown for the typical case of Figure 21. Due to the stick force characteristics of the powered control system, a constant elevator deflection line represents a constant stick force line if the trim input to the control system remains unchanged.

The estimated steady state rolling rates with SAS inoperative exceed the minimum requirements of AGARD Report 408A by a small margin.

From the estimation of trimmed steady sideslip characteristics, the maximum rudder deflection of 20 degrees limits the maximum steady sideslip angles to about 16 degrees for the flaps-up, clean, conventional flight configuration with both engines operating at the same thrust. With the number one engine inoperative, sideslip angles in excess

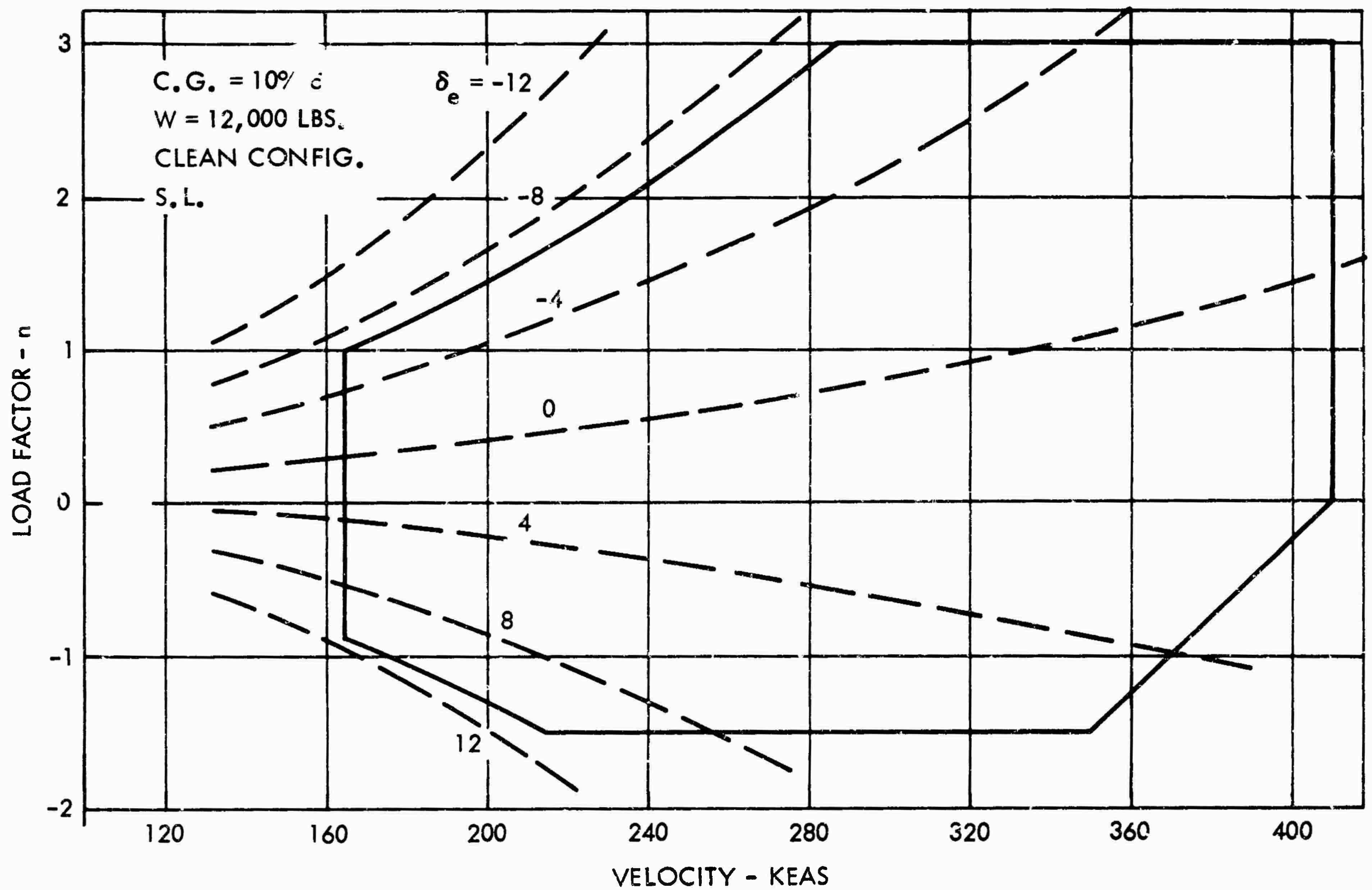


FIGURE 21 - MANEUVER ENVELOPE

of 10 degrees opposing the moment generated by the inoperative engine and less than 20 degrees assisting the inoperative engine are available. For the flaps deflected 40 degrees conventional flight, landing configuration, the available sideslip angles are reduced slightly. At very low speeds the aileron is limiting rather than the rudder.

For both the flaps up and flaps extended 40 degrees conventional flight configurations, the minimum control speeds are below the corresponding stall speeds over the operational weight range utilizing a bank angle limit of 5 degrees.

c. Dynamic Stability

Due to the lack of inherent aerodynamic damping at low speeds, a rate-sensitive stability augmentation system (SAS) is incorporated in the XV-4B control system. Two three-degree-of-freedom dynamic stability digital computer programs utilizing linearized dimensional derivatives were used to solve for the roots of the characteristics equations. For adequate compliance with the AGARD 408A design guide, the SAS gains of the following tabulation were selected:

Mode	Parameter	Gain - deg/deg/sec	
		VTOL	Phase IV
Pitch	$\delta_e / \dot{\theta}$	2.6	0.3
Roll	$\delta_a / \dot{\phi}$	2.9	0.4
Yaw	$\delta_r / \dot{\psi}$	1.16	1.16*

*With 2 second cancellor

Piloted simulation tests using the FDL mechanization of the XV-4B indicated satisfactory handling qualities are provided by these SAS gains with a limited authority of 50 percent. Pilot comments indicated good agreement between actual aircraft flight tests and the FDL simulation tests. As noted in Section VII, however, the above listed gains were modified somewhat as a result of test experience.

SECTION V

THERMO-PROPULSION ANALYSIS

This section presents a summary of the analyses and performance estimates of the propulsion system, reaction control system, and air conditioning system for the XV-4B airplane. Detailed analyses are presented in Reference 18.

1. ENGINE INSTALLATION

The center fuselage cavity, which contains the lift engines and the lift/cruise exhausts, is divided with titanium panels into six vertical rectangular compartments, with the lift engines installed in the forward and aft pairs of compartments. The exhaust ducts from the lift/cruise engines to their lift nozzles pass through the center pair of compartments. Horizontal titanium panels are located at the top and bottom of the compartments, forming a box. These panels have openings fitted with seals to fit around the engine compressor face and the ejector shroud of the lift swivel nozzles.

a. Engine Compartment Cooling

Cooling air for the lift engine compartments is pumped by ejectors attached to each of the lift swivel nozzles. The ejector mixing section length to engine nozzle diameter ratio is 0.25 and the mixing section diameter to engine nozzle diameter ratio is 1.10. The cooling air is taken aboard through flush louvers in the upper fuselage surface forward and aft of the lift engine inlets, and enters the cavity formed by the upper fuselage surface and the top of the tanks. From this cavity, it enters the lift engine compartments through two rectangular doors in each compartment. In the event of fire, the doors are closed by means of actuators before the extinguishing agent is released.

The lift/cruise engines are equipped with ejectors on the lift swivel nozzles and on the conventional cruise tailpipe nozzles to pump cooling air through the nacelles or through the center pair of fuselage compartments, depending on the mode of operation. The geometry of these ejectors is the same as that for the lift engine cooling ejectors except the conventional cruise tailpipe ejector has a mixing section length to engine nozzle diameter ratio of 0.33. Cooling air enters a flush inlet on the forward lower portions of the nacelles for the cruise mode of operation. In the lift mode, the air enters these nacelle inlets and also through two auxiliary inlets on the aft portion of each nacelle, flowing opposite to the normal flow direction.

(1) Cooling Performance--The maximum allowable temperature of the various engine components, as established by the engine manufacturer, are listed below.

MAXIMUM ALLOWABLE ENGINE COMPONENT TEMPERATURES

Component	Maximum Allowable Surface Temperature, °F
Front Frame -	250
Compressor Mainframe -	600
Max. Circumferential Variation -	75
Combustor Casing -	850
Turbine Casing -	1200
Local Hot Spots -	1300
Exhaust Cone -	1300
Accessory Gearbox -	275
Lube Pump -	250
Fuel Control - Operating -	250
- Shutdown -	160
Overspeed Governor -	250
Ignition Generator -	250
Thermocouple Harness -	600
Harness Connector -	350

General Electric also defined the thermocouple locations for each component and in fact installed the engine mounted thermocouples. Analysis showed that the temperature of the structure around the engines, which is titanium, is less critical than the component temperatures and would be satisfactory as long as the components are adequately cooled. Structural temperatures were therefore not measured.

The cooling tests were conducted on the cyclic test rig. The cyclic test rig included the complete aircraft propulsion and reaction control systems, and the engine compartments and nacelles were representative of those in the aircraft. Engine component temperature data were obtained from the left lift/cruise and left forward lift engines.

The temperatures obtained in tests on the cyclic test rig for the various engine components are shown in Reference 18. The component temperatures were measured at approximately five minute intervals between engine speed changes with ascending and descending engine speed. Since the aircraft is to be capable of operating in an ambient temperature of 130°F, the component temperature test data have been corrected by a 1:1 ratio from test ambient conditions to 130°F ambient temperature. These data are presented as a function of percent engine rpm.

All engine component temperatures, when corrected to a 130°F ambient temperature, are within the maximum allowable except for the accessory gearbox and lube pump. Even though the accessory gearbox and lube pump exceeded the maximum allowable temperatures, the engine oil was approximately 50°F below the maximum allowable of 310°F Thus, it was concluded that no action should be taken to modify the cooling system unless a check of component temperatures with engines installed in the aircraft revealed a problem. Data from the flight test program indicated that all components were within temperature allowables.

b. Inlet Loss

(1) Cruise Engine Inlets--The conventional inlets, supplying the cruise engines, were not analyzed for the XV-4B, since the inlets are essentially the same as those of the XV-4A, which had demonstrated satisfactory performance. The only changes were to make the plane of the inlet leading edge perpendicular to the aircraft centerline and to reduce the diameter of the inlet duct (which is a constant-section cylinder) to that of the YJ85 compressor face. Since the lip highlight diameter was unchanged, the lip area contraction increased from 29 percent to 32 percent, which slightly reduces the lip loss during static and low-speed operation. The inlet Mach number at maximum power and static conditions for the XV-4B is 0.40, whereas that for the XV-4A was 0.37. The XV-4B inlet constant cylindrical section length is only 52 percent of that of the XV-4A. Consequently, the net effect of increased inlet Mach number and shorter inlet duct length results in only 67 percent of the duct friction pressure loss associated with the XV-4A inlet.

Since the differences are small, the XV-4A inlet loss curves, shown in Reference 18, were used to determine XV-4B lift/cruise engine performance. This inlet performance represents a total pressure recovery of 0.9955 for the static take-off engine power conditions.

(2) Lift Engine Inlets--The lift engine inlets were designed on the premise that acceptable performance could be obtained with a fixed-geometry minimum-weight

design independent of inlet closure doors. The primary performance requirements of the inlets were to give optimum total pressure recovery in static operation and to give satisfactory levels of total and static pressure distortion throughout the operating envelope arising from transitional flight. To provide good static performance, an area contraction ratio of at least 33 percent is required, as indicated in Reference 18. For the present inlet, the area contraction ratio was designed to exceed this value. For high-speed flight operation the forward lip design is critical. To establish acceptable flow conditions behind this lip, a two-stage approach was adopted. The first was the design of a basic shape with a forward lip radius-to-inlet-diameter ratio of 47 percent; the second stage was the addition of an auxiliary lip designed to suppress separation on the forward lip.

The inlets were developed in a full-scale test program as reported in Reference 19, and a summary of this program is presented in Section VII of this report under Lift Engine Inlet Development. The final basic inlet configuration is the same for all lift engines and the forward lift engine inlets have an auxiliary lip installed. The basic inlet and auxiliary lip geometry is shown in Figure 22.

Figure 23 presents the lift engine inlet total pressure losses used for engine performance calculations as reported in Reference 18. This predicted performance was established after reviewing a number of papers and reports on lift engine inlets for V/STOL aircraft. A comparison of the predicted performance with test results obtained at a later date, as reported in Reference 19, is also shown in Figure 23. This comparison indicates that the test results have a much lower inlet total pressure loss than the predicted total pressure loss used in the performance analyses.

Test results show that total pressure distortion approaches the engine manufacturer's recommended limit of 10 percent only at airspeeds in excess of 185 knots on the aft inlets. The static pressure distortion results indicate that the engine manufacturer's recommended limit of 5 percent is exceeded by both forward and aft inlets, the distortion reaching a maximum value of 13 percent in the aft inlets at 165 knots and maximum power but the engine manufacturer reviewed the data and indicated that these should not lead to engine operating problems. Confidence in the adequacy of the inlet design evolves from the fact that engine stalls and surges were absent throughout the Inlet Development Test Program, numerous engine starts and accelerations at 200 knots relative windspeed were successfully accomplished, and engine vibration levels were always

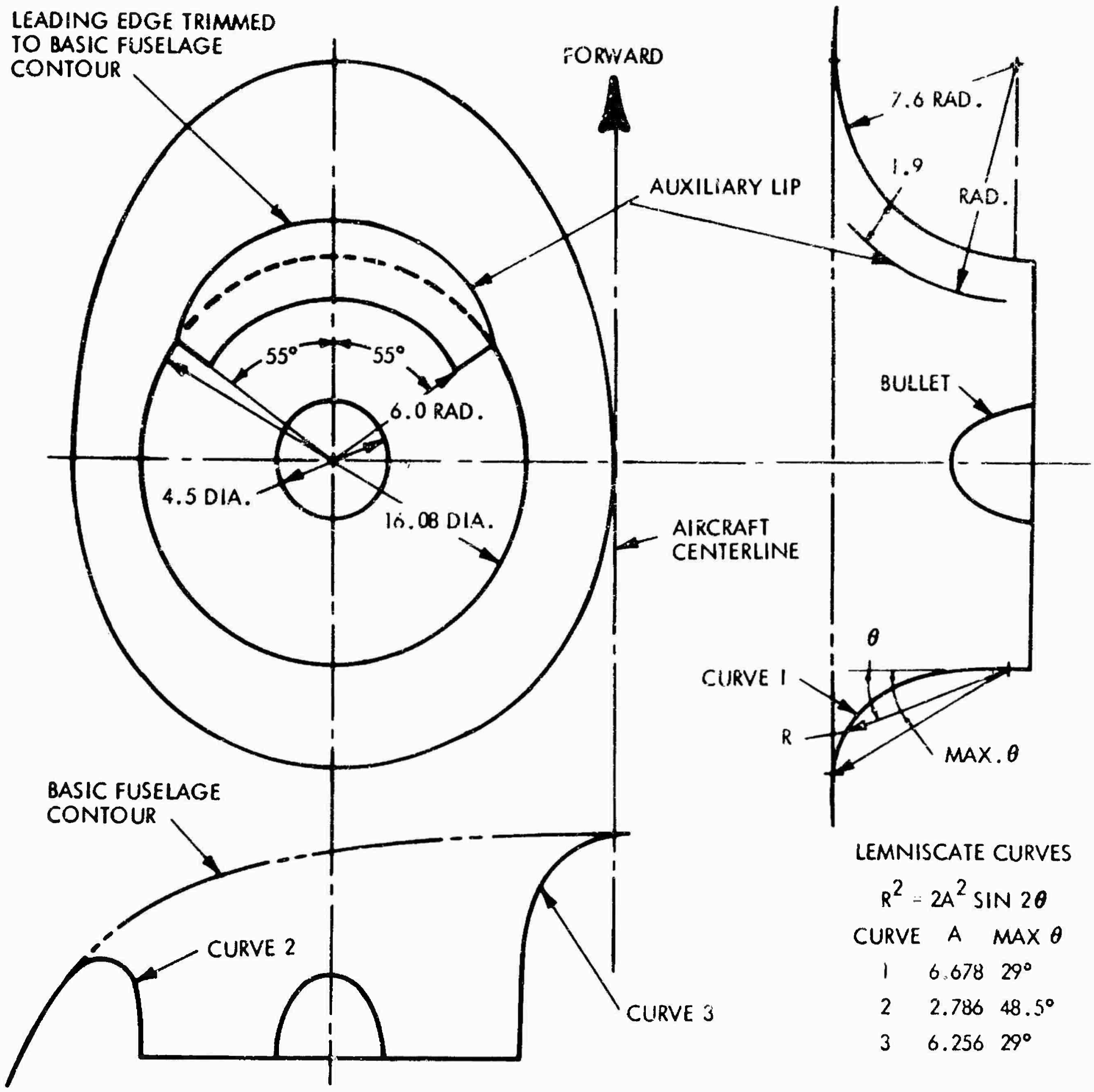


FIGURE 22 - BASIC INLET & AUXILIARY LIP GEOMETRY

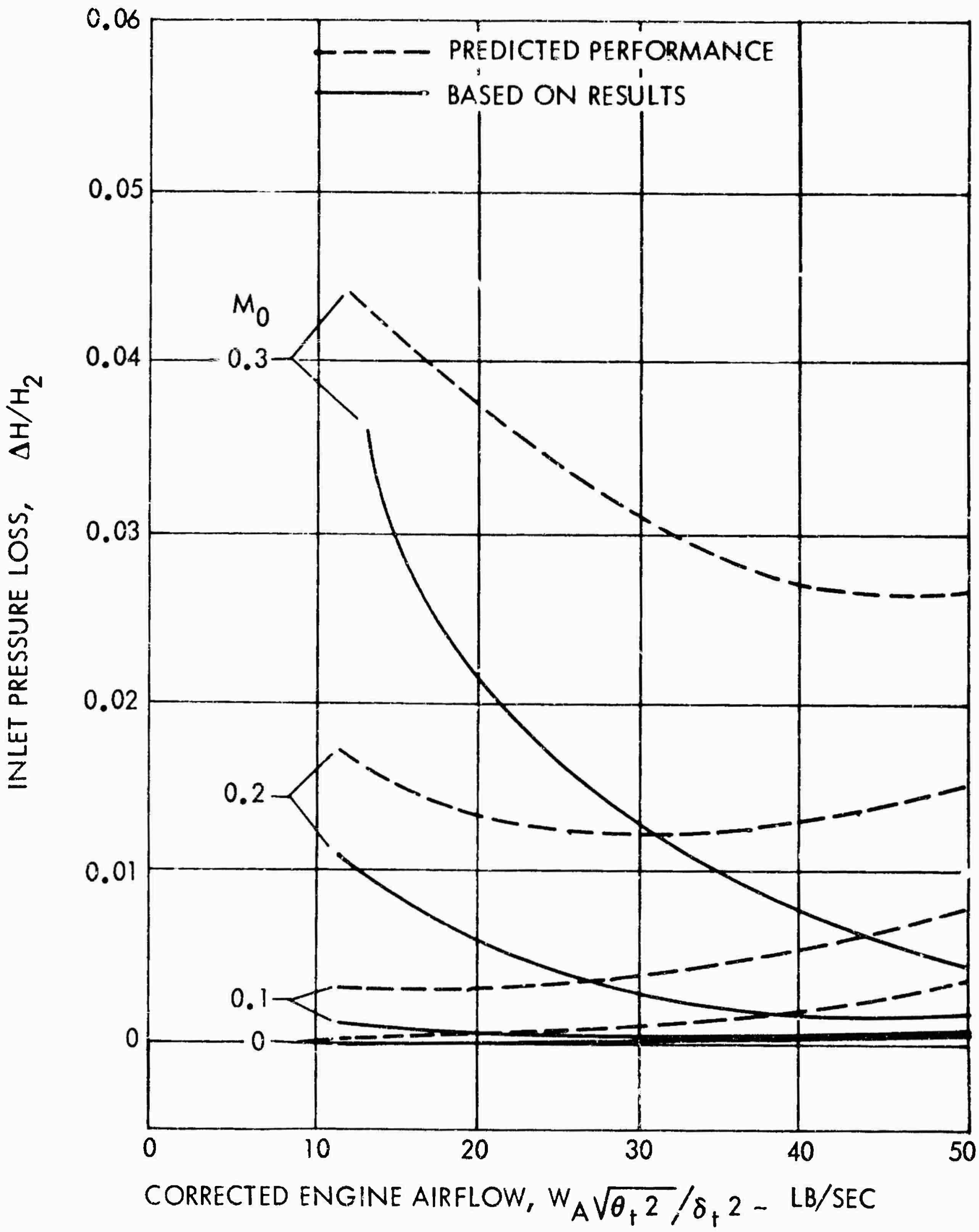


FIGURE 23 - COMPARISON OF PREDICTED LIFT ENGINE INLET PRESSURE LOSS WITH TEST RESULTS

within the manufacturer's recommended tolerances.

c. Exhaust System Pressure Loss

Details of the exhaust system are presented in Reference 18 which describes the duct systems section-by-section and shows the total pressure loss coefficients used in the analysis. The pressure loss associated with each duct component is based on the compressible dynamic pressure taking into account the increase in dynamic pressure due to pressure loss in the preceding components. A summary of percent exhaust system total pressure losses is presented below for engine idle and take-off power levels, and for the different engine exhaust system configurations.

SUMMARY OF EXHAUST SYSTEM PRESSURE LOSSES

Sea Level, Standard Day, Static

	Percent Total Pressure Loss, $\Delta H/H_5$	
Engine	Idle Power	Take-off Power
Lift/Cruise (Cruise Mode)	1.00	5.00
Lift/Cruise (Lift Mode, 0^o Nozzle Deflection)	1.07	5.40
Lift/Cruise (Lift Mode, 10^o Nozzle Deflection)	1.13	6.00
Forward Lift (0^o Nozzle Deflection)	0.32	2.00
Forward Lift (10^o Nozzle Deflection)	0.37	2.55
Aft Lift (0^o Nozzle Deflection)	0.35	2.75
Aft Lift (10^o Nozzle Deflection)	0.40	3.40

The percent pressure loss is referenced to the engine turbine discharge total pressure.

2. ENGINE PERFORMANCE

The XV-4B is powered by six YJ85-GE-19 turbojet engines, defined in General Electric Specification No. E-1129 for a minimum engine. The sea level static standard day uninstalled rating is 3015 pounds thrust at maximum (five-minute limit) power setting. The installed performance of the YJ85-GE-19 engine presented in Reference 18 is derived from General Electric Computer Deck No. PCJ066, which represents the characteristics of an average engine.

a. Nozzle Sizing

If no other values of nozzle area are input, the GE program calculates engine performance for a nozzle area of 104.08 square inches in the lift mode and 105.24 square inches in the cruise mode. These areas give the maximum allowable EGT at the maximum allowable rpm at sea level static hot day (103°F) conditions with no installation losses. The maximum allowable EGT's and rpm's used in the deck to establish installed engine performance are listed below, together with the specification values for comparison.

ENGINE LIMITS

Maximum Rating

Deck				Specification			
T_{t2}, °F	%N	N	EGT, °F	T_{t2}, °F	%N	N	EGT, °F
			Cruise Mode				
≤ 59.0	98.7	16,330	1305				
*	*	*		All	100.0	16,500	1340
≥ 103.0	99.42	16,400	1305				
			Lift Mode				
All	98.97	16,330	1319	All	101.2	16,700	1355

*Interpolate linearly

The program was used to resize the nozzles to allow for installation effects. The lift engine and cruise engine lift nozzles sizes were determined for a constant 7.5 percent customer compressor bleed on a sea level static hot day (103°F) at engine maximum limits of EGT and rpm. The cruise engine cruise nozzle areas were determined for zero customer compressor bleed and all other conditions being the same as those for all other nozzle sizings.

b. Installed Performance

In order to facilitate the calculation of installed thrust and its vector components at a large number of operating conditions, the GE deck was incorporated into a Lockheed program. The resulting program uses the inlet and exhaust system losses determined for the respective systems and iterates the loss calculations and the engine deck to determine installed performance, including an allowance of 0.5 percent of gross thrust for ejector losses. The program also computes the longitudinal and normal components of thrust.

No installed engine performance data is presented in this report, but a description of the data contained in Reference 18 is provided in the following sections.

(1) Conventional Flight--Reference 18 presents standard day installed fuel flow versus installed effective net thrust (i.e., longitudinal component) for flight Mach numbers up to 0.6 at altitudes of sea level, 5,000 feet, 10,000 feet, 20,000 feet, and 30,000 feet; net thrust, fuel flow, and ram drag versus flight Mach number for altitudes of sea level, 3500 feet, and 5000 feet at engine idle and take-off power levels for both standard and MIL-STD-210A hot days; and windmilling gross thrust and ram drag versus flight Mach number at altitudes of sea level, 3500 feet, and 5000 feet.

(2) VTOL and Transitional Flight--Installed engine performance data for VTOL and transitional flight operation are also presented in Reference 18.

Data on a per engine basis are presented for corrected gross lift thrust and corrected fuel flow rate versus corrected engine speed at customer compressor bleed rates of 0, 5, and 10 percent; and similar data are presented for total gross lift and total fuel flow for Phase I of flight.

In Phase II flight, the lift engine data are presented for corrected total gross lift thrust and corrected total fuel flow rate versus corrected engine speed at customer compressor bleed rates of 0, 5, and 10 percent; and similar data are presented for the cruise engines in the cruise mode on a per engine basis.

3. REACTION CONTROL SYSTEM

a. Description

The attitude of the XV-4B is controlled by the ailerons, elevator, rudder, and reaction control valves. The reaction control system consists of the pitch, roll, and yaw control nozzles and the duct system to supply them with compressor bleed air from the engines. The two forward pitch nozzles, located in the nose of the aircraft, provide a nose-up moment. Two aft pitch nozzles, located beneath the tail of the aircraft, provide a nose-down moment. Yaw is regulated by four nozzles located beneath the tail of the aircraft. Roll control is provided by two nozzles on each wing tip. The reaction control system works on a demand basis. When no control is required, the control nozzles are closed. When control is required, the control nozzle areas are increased to provide increased compressor bleed flow from the engines and increased control forces. The control nozzles are supplied by a duct system from the compressors of all six engines. Figure 24 is a simplified schematic of the reaction control system. The design parameters are presented below:

REACTION CONTROL SYSTEM REQUIREMENTS

Control Axis	Angular Acceleration rad/sec^2	Moment of Inertia * $lb\text{-}ft\text{-}sec^2$	Moment Arm ft	Control Force ** lb	Effective Nozzle Area** in^2
Roll	2.630	2,600	13.08	523	11.4
Forward Pitch	0.814	12,600	13.80	743	15.7
Aft Pitch	1.067	12,600	17.60	764	17.1
Yaw	0.792	13,600	16.30	660	14.8

* Based on an aircraft weight of 12,580 pounds.

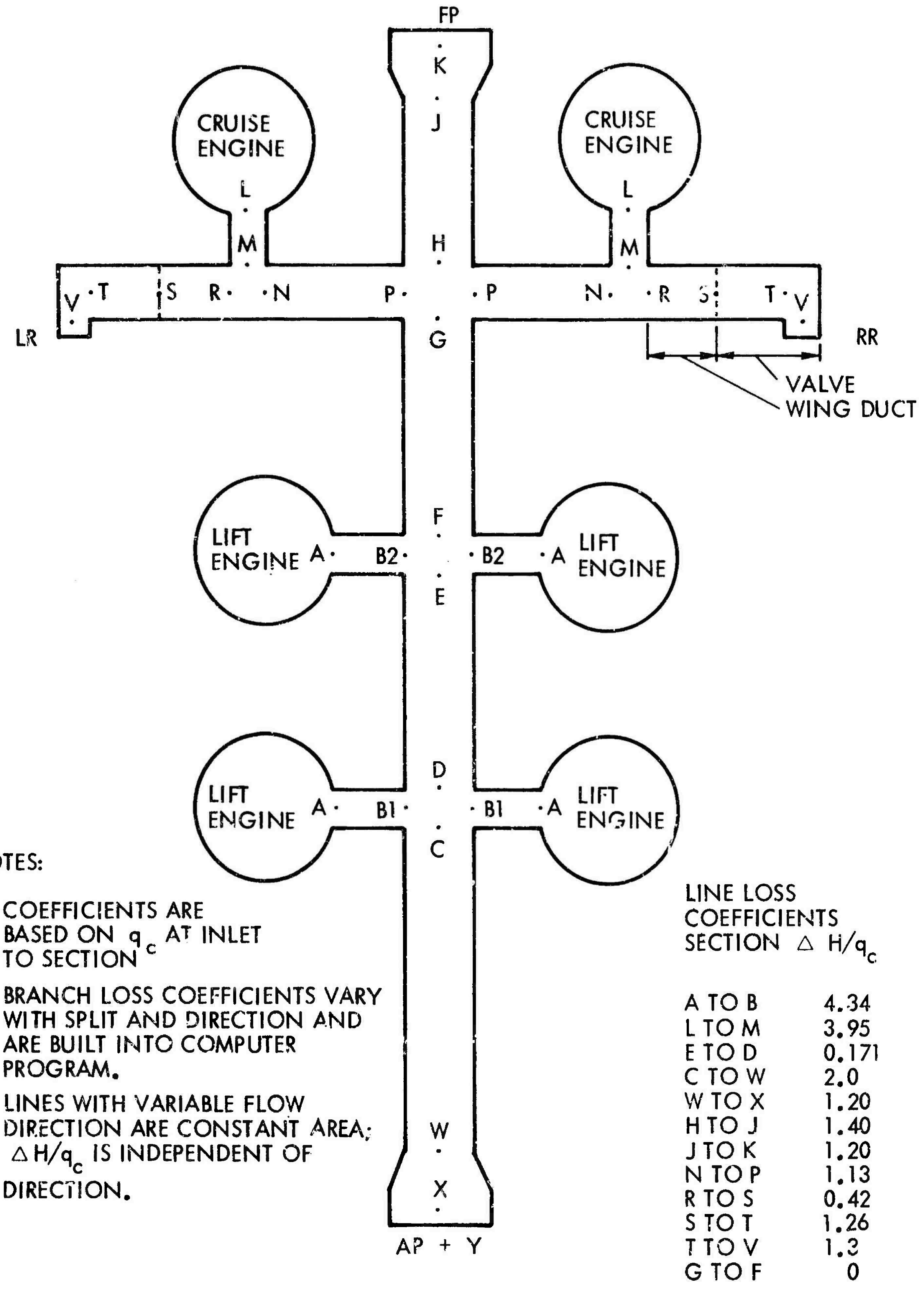
** Based on sea level, hot day, 100% RPM, and 10% compressor airbleed.

b. Computer Analysis

The analysis of the XV-4B reaction control system lends itself well to computer programming. The iterations necessary in the flow analysis, which are not feasible under other circumstances, become quite manageable when performed by computer. For this reason, a program was written for the IBM 7094 computer to calculate the reaction control system performance.

The program calculates specific performance parameters for any given input conditions. The input parameters are: the flight conditions; the six engine power settings; the flow distributions to the forward pitch, aft pitch and yaw, and right and left roll controls; and the initially assumed engine bleed flows. The duct loss coefficients are built into the equations of the program. The branch loss coefficients are contained in stored tables. Both types of losses are discussed later.

The calculated parameters are forward pitch control force, aft pitch and yaw control forces, right roll control force, left roll control force, effective and actual control



NOTES:

1. COEFFICIENTS ARE BASED ON q_c AT INLET TO SECTION
2. BRANCH LOSS COEFFICIENTS VARY WITH SPLIT AND DIRECTION AND ARE BUILT INTO COMPUTER PROGRAM.
3. LINES WITH VARIABLE FLOW DIRECTION ARE CONSTANT AREA; $\Delta H/q_c$ IS INDEPENDENT OF DIRECTION.

LINE LOSS COEFFICIENTS

SECTION	$\Delta H/q_c$
A TO B	4.34
L TO M	3.95
E TO D	0.171
C TO W	2.0
W TO X	1.20
H TO J	1.40
J TO K	1.20
N TO P	1.13
R TO S	0.42
S TO T	1.26
T TO V	1.3
G TO F	0

FIGURE 24 - REACTION CONTROL SYSTEM SIMPLIFIED SCHEMATIC

nozzle areas corresponding to these forces, total pressures at various points throughout the system, and bleed airflow from each engine. The performance parameters are determined from the input parameters by means of tables, equations, and iterations. Because of the many possible variations in the direction of flow through the system, six code numbers corresponding to the six junctions of the system must be input to specify the assumed flow situation.

The tables used in the program consist of four permanently stored tables and four external tables that may readily be changed. The permanent tables represent branch loss curves from Reference 18, in which branch loss coefficients are presented as functions of the pertinent areas, angles, and flow ratios; and a table of compressible dynamic head to total pressure ratio, q_c/P_t, as a function of the flow parameter $W\sqrt{T_t}/AP_t$. The four external changeable tables represent plots of engine performance - corrected compressor airbleed total temperature, corrected compressor airbleed total pressure, and corrected engine airflow, all versus corrected rpm - and a control nozzle thrust coefficient curve, however, a constant value of 0.94 has been applied to all calculated reaction control valve thrust presented in Reference 18.

The iterating process is used throughout the program in order to balance the total pressures of the flows from the several engines. The engine bleeds are varied throughout the system until the total pressures at the junctions are balanced for the required flow distribution to the control nozzles. The control forces and the required actual and effective nozzle areas are then obtained by direct calculation.

In summary, the program operates as follows:

- o Calculation of initial and basic parameters - corrected engine speed, uncorrected values of bleed total temperature and pressure, and uncorrected values of bleed flows.
- o Determination, from input codes, of point in system to begin analysis.
- o Calculation of total pressures at first junction.
- o Balancing of total pressures and flows at first junction if necessary.
- o Determination of next junction to be analyzed.
- o Repetitions similar to Steps (3) through (5) until other junctions in the system have been analyzed and pressures balanced.

c. Loss Coefficients

Branch loss coefficients are stored in the program, as described above. Line loss coefficients for incorporation in the equations of the program were calculated from standard pressure loss data. The various line sections and the overall loss coefficients for each are shown in Figure 24. The components of each section are first analyzed individually. Then the component loss coefficients are added, taking into consideration the increase in q_c due to pressure loss in the preceding components, to obtain the loss coefficient of the whole section.

d. System Performance

It is impractical to present the computer performance of the reaction control system under all combinations of engine power settings and reaction control valve openings. Several hundred curves would be required. The most practical performance presentation is the equations and curves used in the XV-4B simulator, which are presented in Reference 18.

The complexity of a means to predict the reaction control system performance evolves from the fact that all three control axes (roll, pitch, and yaw) are integrated into a common duct system. Thus, the control power about any one axis is affected by a demand for control power about either or a combination of the other two axes. Curves are presented in Reference 18 for control force and percent engine compressor bleed at 100 percent engine rpm as a function of nozzle effective area for the particular axis and also as a function of nozzle area for the other axes. Then there are correction curves to correct roll for forward pitch, yaw for forward pitch, effective engine rpm, thrust for engine rpm, bleed for forward pitch, and bleed for engine rpm. The specific steps are defined in Appendix A of Reference 18 as to the proper use of these curves.

4. AIR CONDITIONING SYSTEM

a. Description

Existing hardware was used wherever possible in the XV-4B air conditioning system. The refrigeration unit, AiResearch P/N 84500, is that used in the Cessna T-37 trainer. It gives satisfactory cooling performance and is compatible with the bleed characteristics of the YJ85-GE-19 engine. Heating is no problem. The temperature control system is identical to that used on the C-140 JetStar. An off-the-shelf pressure regulating and flow control valve is also used, with adjustment capability for varying the flow constant. Since operation of the XV-4B is normally limited to 20,000 feet, pressurization is not provided. The application of the aircraft is such that ice protection is not required.

High-pressure bleed air extracted from the left-hand cruise engine is used as the source of conditioned air. The flow rate varies from approximately 18 pounds per minute at 86 percent engine rpm to 22 pounds per minute at 98 percent rpm regardless of whether the system is in a heating or cooling mode.

Air from the flow control valve enters a two-pass heat exchanger, where it is cooled by ambient air being induced by the refrigeration unit cooling air fan. The cooling air enters the low-pressure system. Hot bleed air flow through this duct is regulated by the temperature control valve, which has a 25-second slew rate. Thus cold turbine discharge air and hot by-pass air are mixed as required to provide the proper temperature for conditioning the cockpit. A temperature sensor is installed in the low-pressure duct downstream from the air mixing point. The signals from the sensor are transmitted to the temperature control box, which in turn pulses the valve to increase or decrease the flow of hot air.

The cockpit is also provided with a ram air ventilation valve which can be activated during cruise if the air conditioning system is inoperative. The manually-operated valve admits outside air through a five-square-inch scoop located in a positive pressure area of the forward fuselage.

b. Cooling Performance

The estimated cooling performance shown below is for a typical low-humidity condition such as is encountered at Edwards Air Force Base and Los Angeles. It is seen that the specification requirement of a 75°F cabin during cruise at 86 percent rpm on a 103°F day is satisfied. Even cooler temperatures are available during hover, due to the higher required engine speeds.

COOLING PERFORMANCE,* DRY CLIMATE

	Spec.	Min. Wt. Hover		Max. Wt. Hover	
Percent Bleed -	0.0	0.0	7.5	0.0	7.5
Lowest Required Thrust, lb (= Weight)		8185	8185	12580	12580
Engine Speed, percent rpm	86.0	89.9	90.9	97.1	98.2
Bleed Pressure, psia	52.3	60.0	56.3	80.6	75.5
Bleed Temperature, °F	397.0	435.0	437.0	533.0	528.0
Cabin Airflow, lb/min	18.3	21.0	19.6	22.6	21.9
Cabin Supply Air Temperature, °F (DAR)	15.0	6.0	14.0	12.0	14.0
Cabin Temperature Capability, °F (at full cold)	66.6	56.8	64.0	58.0	60.5

*Sea level static, 103°F, low humidity, no water separator, full reevaporation of condensed moisture.

Data from the National Weather Records Center at Asheville, North Carolina, confirm that 103°F is a valid maximum temperature for the locations mentioned above, and that humidity levels are in fact negligible. During the hottest period of the year, the average temperature at Edwards AFB is 97.0°F, at a corresponding absolute humidity of 42 grains of water per pound of dry air. The absolute humidity varies little with season and time of day. This amount, plus the 10 grains per pound added by the crew to a cabin flow of 16 pounds per minute, gives a cabin relative humidity of 40 percent at 75°F. These conditions provide a comfortable environment and also prevent fogging of the cabin.

A comparison of weather records at Edwards with the more humid climates associated with Eglin and Dobbins Air Force Bases indicate that a 50 percent effective water separator is necessary to prevent fogging of the cockpit following a rapid descent based on a humidity level of 115 grains per pound of dry air on a 103°F day. However, the use of a water separator compromises cooling performance due to the reduction in evaporation of entrained moisture in the cabin air supply. Thus in order to meet these conditions, a larger heavier refrigeration unit than the 84500 would be required. Since Eglin is not a definitive test site to consider as a design requirement, the use of a water separator does not appear to be justified, based on its relatively low frequency of anticipated need.

c. Heating Performance

Past experience with air cycle machinery has always shown a larger margin in heating capability than in cooling. This results partly from the fact that a larger quantity of bleed air is available for heating since the turbine nozzle area does not restrict the flow. Tabulated below is a summary of the heating capacity of the 84500 unit on a 0°F day during engine operation for an assumed cabin supply air temperature of 150°F. The beneficial effect of crew, solar, and equipment heat input is neglected.

HEATING PERFORMANCE

0°F Ambient Temperature
150°F Supply Air Temperature
75°F Cabin Temperature

	S.L. Hover	1.2 V_{stall}	Max. Speed
Engine Speed, Percent rpm**	82.1	94.0	95.1
Heat Required, Btu/hr	11,600	10,000	6,000
Heat Available, Btu/hr	24,900	19,900	20,700
Cabin Airflow, lb/min	21.8	18.4	19.2

**Engine power setting at lowest thrust required

d. Defogging

Defogging is accomplished by a series of eight openings uniformly spaced in a distribution duct located below the windshield areas. Each outlet is sized to pass approximately 1.2 pounds per minute at design conditions.

In order to prevent fogging, the inside surface temperature of the plexiglass must be kept above the dewpoint associated with the atmospheric humidity. In an aircraft defogging system the most critical requirement occurs at the end of a rapid descent from altitude on a hot humid day. The inner surface of the windshield is fairly cold to start with, and heat from the outside has little time to soak through before humid air begins coming in at low altitude and contacting the windshield.

The analysis indicates that fogging is prevented for all initial altitudes below

15,000 feet for normal descent, even without the defogging system, which extends defogging performance to even more severe conditions. The calculations assume that the inside surface is insulated, which is equivalent to neglecting the contribution of the defogging system entirely. The outside surface is exposed to ram air temperature and the external air film heat transfer coefficient is neglected.

e. Avionics Cooling

The avionics compartment is located aft of the aft fuel cell adjacent to the air conditioning compartment. A summary of the equipment cooling requirements and temperature limits is presented below.

AVIONICS COOLING SUMMARY

1. Upper Compartment -

	Qty	Heat Dissipation, Watts	Ambient Environment Temp. Limit °F
Slaving Accessory and Gyro	1	75	160
VHF Comm., 2 Boxes	1	140	131 *
VOR/LOC	1	22	131 *
Vertical Gyro	1	40	160
SAS Computer and Power Supply, 2 Boxes	1	610	-
Rate Gyros	3	negligible	-
Regulators	2	200 (total)	160
Flight Test Instrumentation (estimate)	-	510	131
Total		1597	

*30-minute limit of 158°F

$$\text{Cooling Air } \Delta T = \frac{Q}{Wc_p} = \frac{1597(3.413)}{18(14.4)} = 21.0\ ^\circ F$$

Air Outlet Temp. = 103 + 21 = 124°F
(SL, 103°F Day)

2. Lower Compartment - 2 inverters, integral - fan cooled, 2820 watts total dissipation, designed for operation with 131°F cooling air inlet temperature.

Equipment cooling is accomplished by drawing ambient air through the compartment by means of an axial-flow fan. Approximately 18 pounds per minute flows over the electronics and avionics components located near the top of the compartment, then is circulated to the inverters, which are located near the bottom of the compartment. Discharge air from the inverters is exhausted overboard through a series of louvered openings in the bottom of the fuselage. It can be seen that maximum ambient temperatures during design conditions at sea level on a 103°F day are below the upper limit for the most critical component.

SECTION VI

INVERTED TELESCOPE AND BALANCE SYSTEM

1. DESCRIPTION

The Inverted Telescope Rig shown in Figure 25 is a project tool designed to support the XV-4B aircraft at various ground plane heights, in such a manner as to permit the operation of all aircraft systems while in a simulated flight position or maneuver. It provides supported positioning of the aircraft for test purposes, such as engine checkout, temperature and pressure survey, vibration survey, control system functional checkout, calibrations, pilot preliminary indoctrination and, with the aid of the balance system, a tool for measuring aircraft performance. The Inverted Telescope and Balance System are more fully described in Reference 20.

The telescope rig permits unrestrained aircraft motions of $\pm 20^\circ$ roll and $+20^\circ$ to -10° pitch, with an additional 10° of snubbing action to all motions; yaw motions are 360° free-swiveling, with a brake to limit yaw rates.

The Inverted Telescope Rig consists of four (4) basic component assemblies: the Fixed Framework structure, the Movable Framework structure, the Aircraft Attachment Linkage and the Balance System components.

a. The Fixed Framework consists of three vertical posts located equidistant on a 49 ft. dia. circle, supported by and braced to suitable concrete foundations. A trifurcated trusswork interconnects between the post tops and supports an electrically powered hoist.

b. The Movable Framework is a symmetrical 3-branch structure supporting a bearing-mounted vertical shaft at its center. The upper end of the shaft is equipped with a hoisting fitting which attaches to the electrically powered hoist on the fixed framework. The outer ends of each branch structure slide vertically in the vertical posts of the fixed framework. At these points, there are mounted hydraulic rams which operate up and down inside the vertical posts with movement of the movable framework.

c. The Aircraft Attachment Linkage consists of a double four-bar linkage, with two horizontal bars pivotally attached to the vertical shaft of the movable framework and two vertical bars extending downward to pivot fittings attached to the aircraft at

FIGURE 25 - INVERTED TELESCOPE SUPPORTING XV-4B

the wing roots. A cantilever arm is also pivotally attached to the vertical shaft and extends aft, its free end being linked to an attachment fitting at the base of the aircraft's vertical fin.

d. The Balance System consists of component parts and assemblies that augment and/or interchange with components of the Aircraft Attachment Linkage to retain the aircraft at a rigid neutral attitude. Load cells, that are integral parts of the Balance System components, measure directionally applied forces.

2. DESIGN CRITERIA

The Inverted Telescope System was designed for safely handling the dynamic loads applied by guaranteed performance characteristics of the completely equipped and fully loaded XV-4B aircraft. Vertical powered lifting provides for a 15 ft. height capability in 1-foot lockable increments.

a. Structural Design Criteria for the entire Inverted Telescope Rig and Balance System is based upon a nominal safety factor of 2, as compared with 1.5 minimum for the aircraft. However, where the design is not restricted by operating clearance, or available hardware, the normal safety factor is 5. Factor of safety is defined as "calculated strength/limit load."

The aircraft may be set at any attitude that can be attained while mounted to the rig with all engines in the lift mode. Adverse combinations of swivel nozzle position may be applied to combinations of pitch, roll and yaw moments. When the cruise engines are used to develop maximum thrust, the aircraft must be at the lowest hoisted position and the thrust increased slowly. At other hoisted positions the cruise engines must not be operated above 85% RPM.

Snubbing loads were based upon the dynamic loads that could be attained by the aircraft and its attachment structure, due to full-open reaction control valve force (as in a "hardover" control maneuver) applied across the unrestrained attitude envelope. For example, full negative pitch moment applied (by aft pitch nozzle thrust) from a $+20^{\circ}$ pitch attitude to a -10° pitch attitude; the dynamic loads developed by this maneuver being damped by 10° of additional snubbing (restrained) motion. These omni-directional design loads are applicable to any two combinations of pitch, roll and yaw dynamics (e.g. pitch-roll, pitch-yaw, roll-yaw). Design loads for yaw dynamics were limited to 1.4 radians/second (80°/sec.; 13 1/3 rpm) by the maximum capability of the yaw brake.

b. The foundation requirements, consisting of suitable footings for the fixed framework, will vary with local soil bearing characteristics and building practices.

The three post footings were equally spaced on a 49-foot diameter circle; the six brace footings on a concentric 74.8-foot diameter circle, about 14.1 ft. from each vertical post. Since it was important that the vertical posts be accurately located, to ensure proper vertical movement of the movable framework, final post locations were established during erection. Structurally integral steel bearing plates on each footing permitted subsequent positioning and weld attachment of the vertical posts and braces.

Each vertical post footing and its attachment fittings were designed to be capable of reacting:

Vertical, down	160,000 lbs.*
Horizontal (all directions)	12,000 lbs.*

Each brace footing and its attachment fittings were designed to be capable of reacting:

Vertical, up	68,000 lbs.*
Horizontal (toward post)	40,000 lbs.*

*All loads included a safety factor of 5.

Consideration was given to installation location and orientation, especially with regard to predominant wind direction. The Inverted Telescope Rig was so oriented that, when the aircraft was headed into the predominant wind direction, the cruise engine exhaust did not impinge directly upon one of the three vertical posts. The control shack, emergency vehicles, stand-by personnel, utility lines, etc. were located on the upwind side, at about 45° to the aircraft centerline.

3. TESTS OF INVERTED TELESCOPE

The following tests of the Inverted Telescope were conducted in order to qualify its functional capabilities and demonstrate the necessary safety requirements.

- Drop Tests - Tests were conducted on the movable framework by dropping simulated loads from various heights. All tests were successful.
- Load Tests - During the drop tests simulated loads equaling full gross weight were used acting as a limit load test for the telescope system.
- Sprinkler Foam Test - For fire protection an automatic sprinkler system was designed, installed and tested by the Automatic Sprinkler Corporation of America. Lockheed-Georgia Company's Fire and Safety Departments approved the design, installation and tests, which were successfully completed by Automatic Sprinkler Corporation.

4. OPERATIONS USING TELESCOPE

The following is a summary of the operations and tests performed on the Inverted Telescope.

- Cruise and lift engine runs to checkout nacelle cooling, exhaust nozzle sizing, diverter valve cycling, engine vibration, etc.
- Electrical system checkout under simulated hover flight conditions.
- S.A.S. checkout under simulated hover flight conditions.
- Landing gear cycling tests.
- Hot tire tests.
- Sonic fatigue tests of reaction control duct system.
- Sonic survey of aircraft structure under simulated hover flight conditions.
- Balance system measurements for demonstration of T/W and control power guarantees.
- Pilot familiarization.

5. OPERATIONAL EXPERIENCE

Several observations concerning the use of the Inverted Telescope were made during the course of the flight test program and are listed below.

- Extended time was required for mounting the aircraft as a result of the importance of proper positioning, complicated structural attachments and time consumed in making electrical and instrumentation installations.
- Additional weight was added to the aircraft by the requirement of mounting hard points on the upper surface as well as the lower surface.
- The desired height location for the movable framework with the aircraft mounted was not readily obtainable because the friction for each post was not the same and one leg would lag or lead the others.
- Inattention to the release of locks prior to repositioning of the movable framework resulted in damage to the vertical posts.
- Movement of the aircraft in relation to the ground plane was experienced during runs at high engine speeds with the reaction control valves operating. This was the result of design clearances, manufacturing tolerances and overall deflections in the aircraft and telescope systems.
- The Inverted Telescope was very useful throughout the test program, although excessive time was required to mount and use it.

SECTION VII
TEST PROGRAMS

1. INTRODUCTION

This section presents a summary of the test programs conducted during the development of the XV-4B aircraft. Programs summarized include: Wind Tunnel Tests; The Inlet Development Test Program; Cyclic Tests to develop VTOL Propulsion and Reaction Control System Hardware; Reaction Control Valve Tests; Structural Proof and Flutter Tests; Acoustics, Temperature, Vibration and Sonic Fatigue Tests; Escape System Tests and the Flight Tests.

2. WIND TUNNEL TEST PROGRAM

The main objective of the XV-4B wind tunnel test program was to obtain powered interference test data. This was necessary to establish the stability and control characteristics of the basic airplane in hover and transition. These tests were required as the first step in developing airframe-stability augmentation system compatibility.

The wind tunnel test programs consisted of what may be considered five separate tests:

- o Langley Test 178
- o University of Maryland Test 488
- o University of Maryland Test 493
- o Langley Test 221
- o Langley Test 226

a. Model Description

A 0.16 scale model of the XV-4B "Hummingbird" airplane was the basic model used in all tests. The fuselage had a steel frame to which the wing, empennage, model engines, and fiberglass fuselage skin were attached. It permitted use of either a conventional aft fuselage sting support or a forward fuselage sting support for 180 degree data runs. It also had provisions for wing trunnion supports. The aluminum wing had provisions for both 0 and 40 degrees of flap deflection. Only positive aileron roll capability was provided, with either left or right aileron deflections of 0, 10 or 20 degrees. The aluminum empennage had movable control surfaces. Rudder deflections of -10, 0, +10 and +20 degrees were available; as were elevator deflections from -30 to +30 degrees in 10 degree increments. Horizontal tail incidence changes were available in 5 degree increments from -10 to +15 degrees. The direct lift jet engine system of the XV-4B was simulated by using externally supplied compressed air to power six ejector units. Each nozzle exit was instrumented with eight total pressure probes manifolded together and one static pressure probe. A detailed description of the model and all of its components used in the Langley tests may be found in References 15 and 16. A picture of the model installed in the Langley 17-foot test section is shown as Figure 26.

Aircraft engine power was simulated in the model power runs by supplying high pressure air (up to 320 psig) to the ejectors. The ejectors were calibrated by recording primary and secondary mass flow rates, nozzle static and total pressures, and resultant thrust as supply pressure was varied. Tunnel power settings were maintained then by controlling the supply pressure and monitoring the ejector primary mass flow rate and nozzle static and total pressures. At a given tunnel speed, airplane power settings were simulated by matching the ratio of inlet mass flow to exit thrust. Details concerning ejector calibration, components and power settings are presented in References 17 and 18.

b. Test Program

A good appreciation of the tests and their significance is best accomplished by discussing each test in some detail. Each of the five tests mentioned earlier are summarized below.

(1) Langley Test 178--Langley Test 178 investigated three areas of flight: hover, powered transition, and cruise. Test objectives for each flight regime were

Hover:

(a) Nozzle incidence study in and out of ground effect at zero and low forward velocities

(b) Horizontal stabilizer effectiveness for various nozzle incidence angles

(c) Elevator effectiveness

(d) Effect of inlet and exit doors and landing gear

(e) Effect of 180 degrees yaw at low velocities

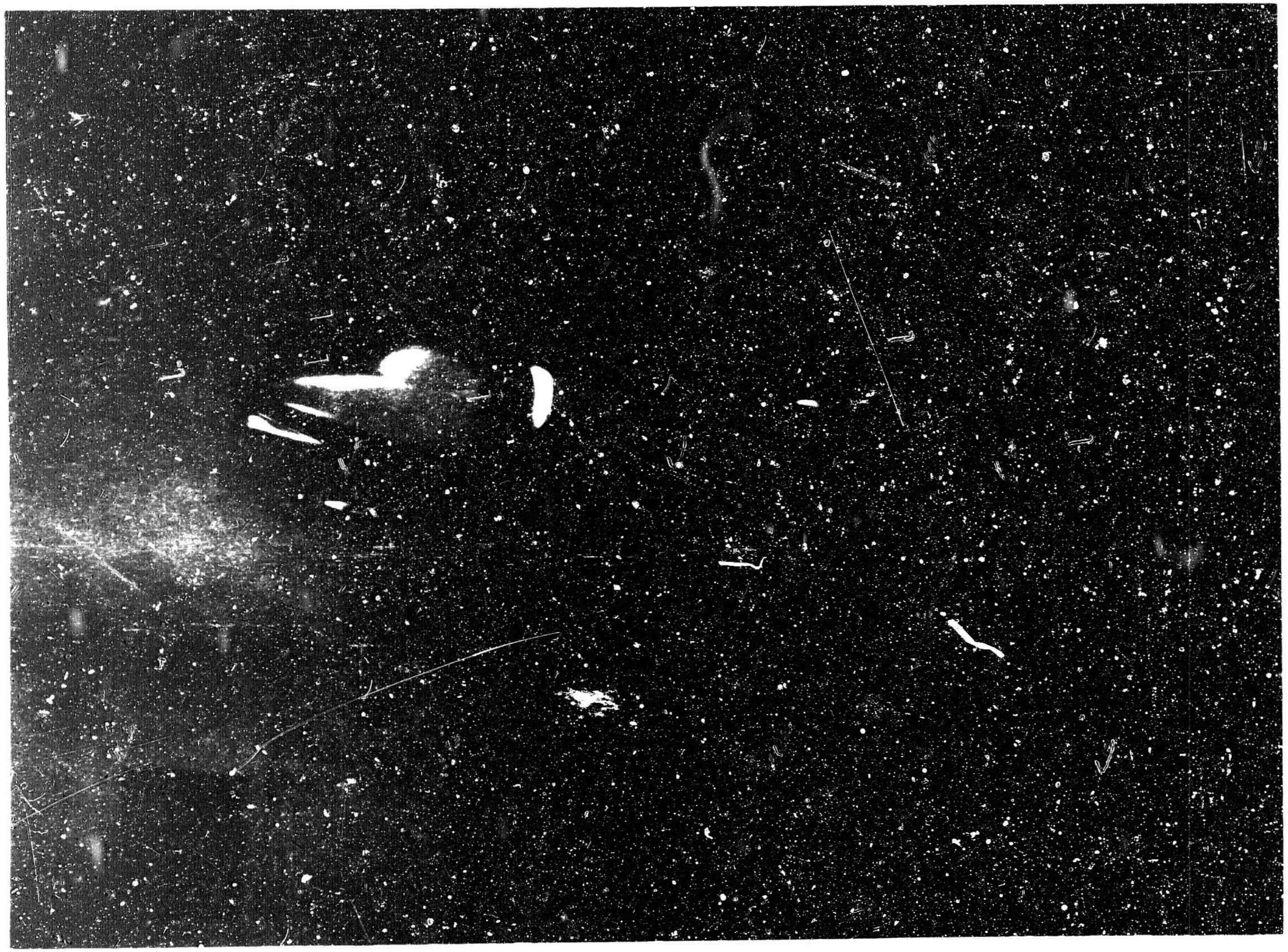


FIGURE 26 - XV-4B WIND TUNNEL MODEL

Powered Transition:

(a) Nozzle incidence study in and out of ground effect

(b) Deep stall with and without power

Cruise:

(a) Rudder effectiveness

(b) Aileron effectiveness

(c) Horizontal stabilizer effectiveness with and without power

(d) Deep stall

(e) Drag study without power

Tunnel speeds selected for the test were 25, 43 and 57 knots, which were used in conjunction with 5 possible power settings. Most pitch runs were made within the angle of attack range of -7 to +28 degrees, although the deep stall study encompassed a +6 to 46 degree range. Beta angles normally varied from +5 to -26 degrees; simulated rearward flight utilized betas from -173 to -208 degrees. A continuous moving belt was utilized to simulate ground effects that corresponded to full scale gear ground clearances of 1 to 31 inches.

The overall test results substantiated the powered lift and drag interference values which had been predicted, but showed that the powered pitching moment interference was less than predicted in the normal flight regime. The test did confirm a deep stall pitching moment problem that led directly to the University of Maryland test.

(2) University of Maryland Test 488--The University of Maryland Test 488 had one basic objective; find a cure for the pitch-up problem. The pitching problem commenced at a model angle of attack of about 12 degrees and was severe in nature. Prompt pilot action would be required to prevent the aircraft from becoming "locked-in" at angles of attack well beyond stall. The fix would need to either increase the angle for pitch-up onset by 4 to 5 degrees, or decrease the pitch-up moment so that recovery at any angle attack could be accomplished with elevator alone.

No attempt was made during the test to simulate powered flight. One beta run was made to check pitching moment in yawed flight. Angle of attack variations were from -8 to +48 degrees to cover both normal operation and deep stall. Runs were made with and without gear, and with flaps both down and retracted. Tunnel dynamic pressure (q) was varied from 10 to 60 pounds per square foot. High q runs were used predominately for fix evaluation, and low q for flow visualization.

The 0.16 scale XV-4B model was tested with numerous modifications. The fixes included low tail location, tail size and span variations, canard surfaces, an aft-fuselage speed brake, part and full span leading edge slats, wing incidence and leading edge radius variations, parachutes, fillets and fairings, vortex generators, spanwise blowing for boundary layer control, and aft-fuselage strakes.

Many of the fixes attempted helped or alleviated the deep stall problem. Unfortunately, most of them also produced large changes in the basic aircraft characteristics. Such a fix would have been highly undesirable because provisions for determining the extent of the changes were not and would not be made. The aft-fuselage mounted strake offered the best solution. Installation was easy to implement, and of all the workable fixes it produced the least change in basic aircraft characteristics within the normal operation flight envelope.

All test data was obtained power off. Interference and tare runs were not made because fix selection was to be based on deltas taken from the basic model data values. Comparisons of test data with other tunnel tests could only be made to the extent judgment permitted.

(3) University of Maryland Test 493--The University of Maryland Test 493 had as its main objective the optimization of strake incidence and size. The desired result was to provide adequate stall recovery capability with the least effect on normal operation aircraft basic data. Tares and interference were not determined. Final selection was to be based on deltas taken from the base run. The strake configuration selected for the next Langley test corresponded to a full scale area of 11.6 square feet with an incidence of 3.4 degrees.

(4) Langley Test 221--In the final Langley test, all investigations performed in the 17-foot test section were designated Langley Test 221. The purpose of the test

was to determine powered interference effects in and out of ground effect for the XV-4B with the aft-fuselage mounted strake installed. The simulated flight conditions were hover, powered transition, and cruise. Angle of attack studies up to +98 degrees were investigated, as were sideslip angles of up to -98 degrees. Some variable q runs were made but the majority of tunnel dynamic pressures were a nominal 11q, with a few high alpha runs made at 6 q. Ground effects at several heights were determined using a continuous moving belt with adjustable height and speed. A few runs were made which simulated engine out conditions to determine powered interference effects for that condition. Additional information on control effectiveness was obtained with the strake installed. A NASA-designed low tail was tested in simulated cruise flight at deep stall attitudes.

Test results disclosed there were no apparent additional power interference effects due to the aft-fuselage mounted strake. It was also shown that deep stall recovery with elevator alone was now possible with all power settings. The engine out power effects were about as predicted. Extreme angles of attack and sideslip did not disclose any problem areas. The NASA-designed low tail cured the deep stall problem, but was inadequate for basic stability purposes. Powered ground effects would also be an unknown quantity for a low mounted tail.

In comparing these test results with the earlier test, it was found that the changes in interference effects due to nozzle deflection were not as great as previously experienced. Discrepancies in static powered data could be attributed to power tares or differences in instrumentation and calibration of primary air.

(5) Langley Test 226--Langley Test 226 was an unpowered test in the 7 x 10-foot high speed test section. The purpose of the test was determination of aircraft basic data. Drag of the gear, doors and strake had not really been determined. This test furnished data at considerably higher Reynolds numbers than had previously been available.

The model was the same 0.16 scale XV-4B that had been used in all previous tests. No new components were tested. Nominal tunnel dynamic pressures were 60, 100 and 150 with Reynolds numbers from 1.00 to 1.57 million. The angle of attack range varied from -7 to +13 degrees, and sideslip angles from -7 to +10 degrees were tested. All data obtained corresponded to the power off, conventional flight mode of operation.

This test substantiated previous estimates for aircraft basic data. Estimated strake effects were validated, and all comparisons with University of Maryland data were quite favorable.

3. LIFT ENGINE INLET DEVELOPMENT PROGRAM

This section presents a summary of the Full-Scale Lift Engine Inlet Development Program for the XV-4B aircraft. The results from this test program are presented in Reference 19.

a. Design Requirements

The design requirements for lift engine inlets in V/STOL aircraft are severe. This severity evolves because the lift engine must be installed with minimum weight, volume, and frontal area; because engine airflow must be decelerated from flight speed to a considerably lower inlet velocity and turned approximately 90°; because engine airflow must enter with minimized pressure loss and pressure distortion; and because an inlet system of simplicity and reliability must be provided to operate over a wide range of relative freestream velocities and engine power levels.

A number of experimental research programs have been conducted related to the design of multiple lift engines in a pod. Studies included tests of inlet configurations involving retractable scoop-type inlet closure doors; individual doors for each inlet; large doors for two or more inlets; and a simple retractable cascade mounted ahead of the front inlet and single auxiliary lips for the remaining inlets in a multiple unit pod. The test results for all scoop-type inlet configurations indicated that pressure-operated louvers would be required in the door or that the door position should be varied as a function of freestream velocity to improve the inlet pressure recovery at low flight speeds and high engine powers.

This effort was concerned with the development of lift engine inlets for the XV-4B aircraft which would provide satisfactory inlet performance for all modes of VTOL flight. The use of ram air for in-flight engine starting was not a requirement since turbine impingement starting using lift-cruise engine compressor bleed air was to be incorporated; however, favorable windmill characteristics were desirable to minimize bleed air requirements. The inlet configurations tested in this program were designed to be independent of inlet closure door considerations and were developed on the premise that a fixed-geometry inlet would satisfy the XV-4B requirements.

b. Model, Test Facility, and Instrumentation

(1) Model--The test model, as shown in Figure 27, represented a full-scale simulation of the XV-4B aircraft front and center upper fuselage geometry and the cruise engine inlet and nacelle geometry. The model structure provided mounting points for the four vertically mounted GE YJ85-5 engines, which were located in a rectangular array with centerline spacings of 1.5 and 2.5 inlet throat diameters laterally and longitudinally, respectively. The complete model was mounted on a movable base which was anchored to the test pad at a number of points. Model position and yaw angle variations were made by translation and rotation of the complete model and base, while pitch angle changes were made by rotation of the model about a pivot point fixed on the base.

The primary performance requirements of the inlets were to give optimum total pressure recovery in static operation and to give satisfactory levels of total and static pressure distortion throughout the operating envelope arising from transitional flight. To provide good static performance, a contraction ratio of at least 33 percent was required. For the present inlet, the contraction ratio was designed to exceed this minimum. For highspeed flight operation the forward lip design is critical. To establish acceptable flow conditions behind this lip, a two-stage approach was adopted. The first was the design of a basic shape with a forward lip radius-to-inlet-diameter ratio of 47 percent; the second stage was the addition of an auxiliary lip designed to suppress separation on the forward lip.

The basic inlet design, shown in Figures 22 and 27, evolved from the above requirements, the constraints of the basic fuselage contours, and the data of previous experimental studies. The auxiliary lip was designed to fit inside the basic fuselage contours and to give a ram scoop effect to unload the basic lip. Three designs of the auxiliary lip were fitted to the two forward inlets during the development program. The first design tested was a short lip fixed in such a position that the leading and trailing edges were approximately 3.2 and 2.0 inches, respectively, from the basic lip. The second lip was mounted on two pivoting brackets near the trailing edge to allow the leading edge position to be varied and locked at a series of settings using a horn-type mounting fixed to the basic lip. The third and final lip design as shown in Figure 22

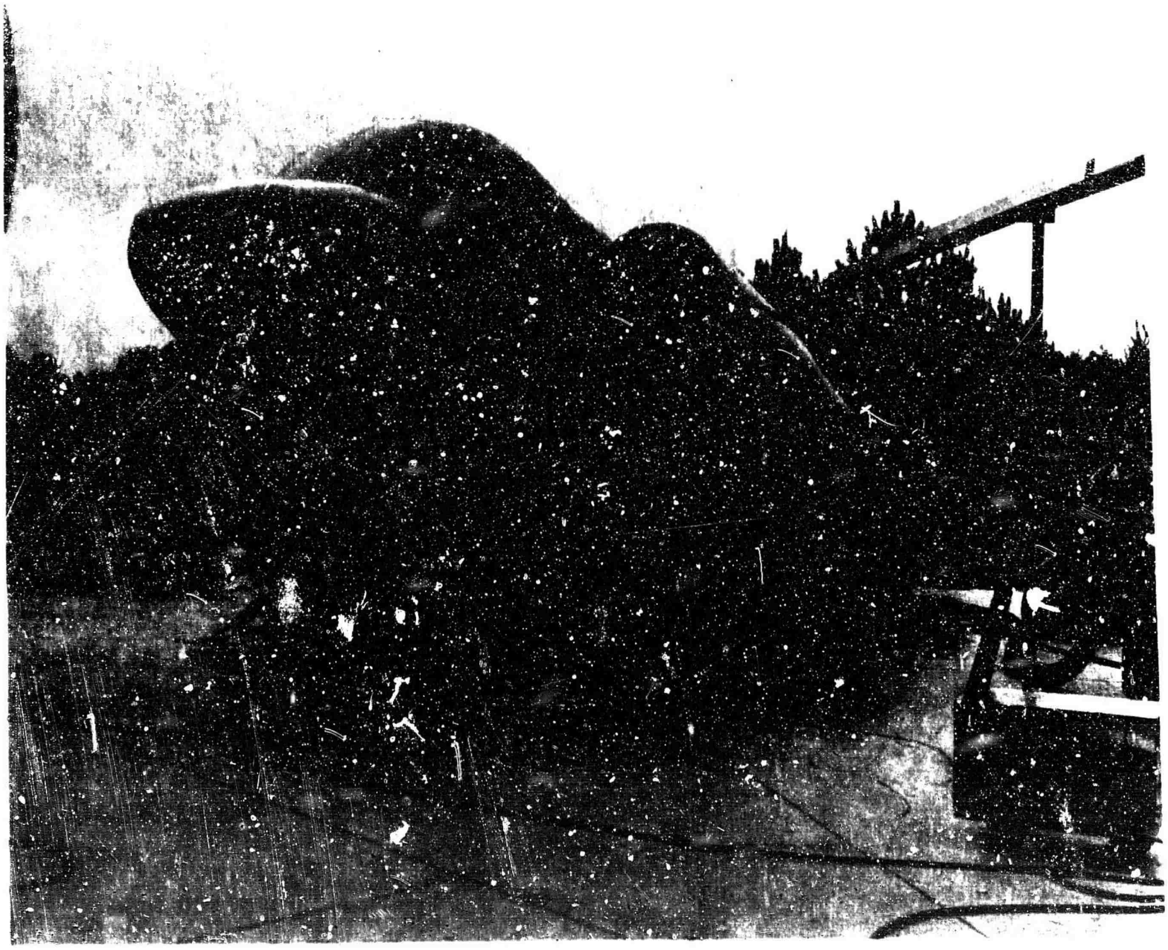


FIGURE 27 - INLET MODEL AND TEST PAD

incorporated the optimum position determined from tests on lip Number 2 and was made from the same lip but with the mountings faired and fixed.

(2) Test Facility--Basically, the test facility (Figure 28) consisted of a VTOL test pad, fuel system, wind machine, control room, and auxiliary equipment (engine start cart, electrical power supply, portable lights, and fire extinguishers). Of these items, the VTOL test pad and wind machine are of primary significance in the investigation reported herein.

The VTOL test pad consists of a concrete pad, a steel grid in the pad center opening, a concrete chamber beneath the pad center opening, and exhaust gas discharge ducts. The test pad was designed to collect the lift engine exhaust gases, divert the flows into three pairs of five-foot diameter ducts, and discharge them at points remote from the engine inlets.

The wind machine was a 500,000 cfm centrifugal blower. The nozzle centerline was located at the same height as the model top surface in the zero-pitch attitude. The blower was fitted with a high-speed rectangular nozzle, 22 by 66 inches, for simulation of wind speeds in the range of 80 to 200 knots, while lower wind speeds were created using the full 80-inch diameter circular nozzle.

(3) Instrumentation--Inlet instrumentation, located in each inlet as close to the engine entrance as practical, was designed to avoid engine-excited resonant frequencies. Total pressures, measured with eight rakes of four tubes each at area-weighted radii, were insensitive to flow angles of at least 20 degrees with the probe head design used. Four static pressure probes, essentially insensitive to flow angles up to 13 degrees, were located (inboard, outboard, forward, and aft) at 94 percent of the inlet radius. Eight thermocouples with bare bead heads were used to measure total temperature. All inlet instrumentation was in the same plane.

c. Test Program

A test program including combinations of lift power, relative freestream velocity, and aircraft attitude for the aircraft hover and transitional flight regime was developed. The program, reported in Reference 19, was limited to those conditions likely to arise from steady controlled maneuvers and did not encompass all attitudes possible during unsteady flight. Some limits on combinations of engine power and freestream velocity

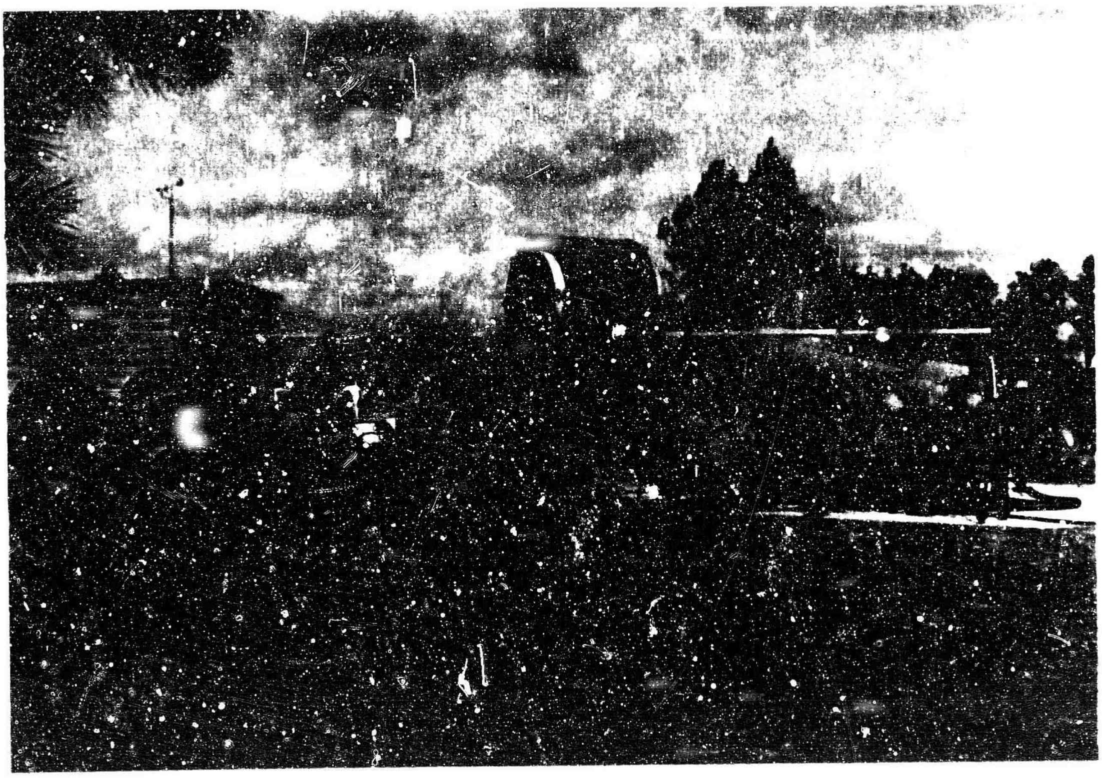


FIGURE 28 - GENERAL VIEW OF INLET TEST FACILITY

were imposed due to the required total mass flow rates approaching the total wind machine flow rate. Separate test series were programmed employing the wind machine equipped with, first, the high-speed nozzle, and second, the larger low-speed nozzle.

Preliminary tests were run to determine wind machine freestream velocity profiles for the high-speed rectangular nozzle, both at the nozzle exit plane and at a station downstream of the nozzle exit corresponding to the location of the model inlets. The exit plane traverses indicated that the flow was substantially uniform at this location. Traverses of the wind-stream at the model location, which were obtained with the model removed, indicated that a high-velocity core was present in the windstream. This core was not large enough to envelope all four inlets. As a result, the model was located in positions such that particular fore-and-aft pairs of inlets were enveloped in the slipstream core. Model positions for all of the high-speed tests were determined as described and only the test data from those inlets in the slipstream core were evaluated. However, all four engines were operated at nominal power settings indicated in Table III throughout the test program. With the low-speed nozzle, the velocity profiles were considerably fuller, and the model center was positioned on the wind machine centerline. It should be noted that throughout the tests, the relative freestream velocity was established at each inlet as described in the following section.

d. Results and Discussion

(1) Reduction and Correlation of Results--The determination of freestream velocity presented a problem since the wind machine slipstream velocity was not completely uniform in either the cross-stream or streamwise directions due to diffusion of the wind machine slipstream. Also, changes in model attitude and engine power levels modified the manner in which the slipstream was deflected and ingested. It was required, therefore, to relate the performance of each inlet to the effective freestream velocity. The assumption was made, and later verified, that a region of full total pressure recovery would occur in all inlets at all test conditions. As a result, the freestream total pressure was determined from inlet total pressure probe readings. The pressure distributions were monitored throughout the program and only probes located in the interior of zero-loss regions were used for the determination of freestream total pressure.

It will be noted that the test results are largely presented in the form of coefficients based on inlet mean dynamic pressure $(\bar{q}_2)$. The inlet airflow dynamic pressures and velocities used in the data reduction process were calculated from the engine

TABLE III

INLET DEVELOPMENT TEST PROGRAM OUTLINE

Yaw Angles Degrees	Pitch Angles Degrees	Wind Machine Nominal Velocity Knots	Range of Engine Power Settings, N% (see note below)
		HIGH-SPEED SERIES	
-15	10	0	48, 64, 80, 90, 100
15	10	120	48, 64, 90
0, -15	10, 0, -10	120	48, 64, 90
15	10	160	48, 64, 90, 100
0, -15	10, 0, -10	160	48, 64, 90, 100
15	10	180	48, 64, 80, 90, 100
0, -15	10, 0, -10	180	48, 64, 80, 90, 100
15	10	200	48, 64, 80, 90, 100
0, -15	10, 0, -10	200	48, 64, 80, 90, 100
		LOW-SPEED SERIES	
0, -15	10, 0, -10	44	48, 64
0, -15	10, 0, -10	70	48, 64, 80, 90, 100
0, -15	10, 0, -10	94	48, 64, 80, 90, 100
-45	10, 0, -10	34	48, 64, 80, 90
-45	10, 0, -10	44	48, 64, 80, 90
90, -90	0	24	48, 58, 80, 90
90, -90	0	31	48, 64, 80, 90

NOTE: In the high-speed section of the program, all engine starting and stopping performed with 200 knots nominal wind velocity.

airflow characteristic using a nominal 16-inch diameter inlet. The following reasoning led to the use of this method of correlation, which was found to be good within the limits of experimental error for all of the results obtained. It is argued that for a given model configuration and attitude, at a fixed freestream-to-inlet velocity ratio (V_0/V_2), the inlet flow pattern will be fixed provided that changes in the actual velocity levels do not modify the positions of any flow separation lines which may exist on any of the inlet surfaces. It is also argued that, should a separation region exist, the pressure losses occurring within this region will be dependent upon the dynamic pressure of the flow adjacent to this separation region. Further, since the flow is essentially subsonic, the dynamic pressure at any point will be proportional to some reference dynamic pressure, such as q_2, for fixed V_0/V_2.

(2) Preliminary Inlet Configuration Test Results--Results obtained from tests on the basic inlet indicated that at zero wind velocity the pressure recovery was optimum; however, at speeds between 120 and 200 knots the pressure distributions indicated that a separation region had developed on the forward lips of all inlets. This separation was caused by the severe adverse pressure gradients on these lips arising from the rapid diffusion and curvature of the inlet airflow. A feature of the distributions, and one which was found for all inlets and all configurations tested, is the region of zero loss in the rear of the inlet. In general this zero-loss zone was located in the back of the inlet with respect to the relative windstream, as was to be expected. As a result of the separation region, total pressure distortion (TPD) and total pressure loss (TPL) increased progressively with freestream speed at a fixed engine power setting. The magnitude of TPD was substantially in excess of the engine manufacturer's 10 percent limit and a configuration review led to the installation of Auxiliary Lip Number 1, which had been designed to suppress this separation, in the forward inlets. The decision on the fitting of auxiliary lips to the rear inlets was postponed as the preliminary test results indicated marginally acceptable distortion levels.

Tests with Auxiliary Lip Number 1 fitted showed that, while the basic lip flow separation was suppressed, a severe flow separation occurred at the auxiliary lip leading edge, and this again resulted in unacceptably high pressure distortions. The lip was positioned to produce a channel between it and the basic lip contracting the 3.2 inches clearance at the leading edge to 2.0 inches at the trailing edge. It appeared that this contraction was too strong and that a more nearly parallel channel might produce more desirable flow conditions. The reasons for this were that the inlet-to-approach streamtube area varied over a wide range, and that the high distortion levels were found to occur at the maximum freestream velocity in combination with moderate to low engine power levels. In particular, very high TPD's were obtained with V_0 = 200 knots in the range $1.2 \leqq V_0/V_2 \leqq 2.6$; in this range the inlet-to-approach streamtube area ratio varied from about 1.2 to 2.6. As a result of these considerations, the auxiliary lip design was modified to control the diffusion on the basic inlet leading lip. To provide this capability Auxiliary Lip Number 2 was built to allow the lip leading edge clearance to be reset between tests and thus provide variable channel contraction or expansion ratio. It was recognized that at the optimum fixed position of the lip, flow separations from the basic lip or the auxiliary lip would be likely to occur at off-design velocity ratios. The approach, however, was to minimize the distortions and eliminate all TPD's greater than 10 percent with a fixed auxiliary lip while maintaining substantially optimum recovery under static conditions.

Two test series were performed with Auxiliary Lip Number 2 with lip leading edge clearances of 1.7 and 2.1 inches and the results, which were similar, indicated significantly reduced TPD levels in the critical range and as a result the TPD levels were within the recommended 10 percent limit. Static tests with these configurations indicated very small reductions in total pressure recovery. Based on these test results the final lip was designed to give a leading edge clearance of 1.9 inches. Further testing indicated that the performance of the clean rear inlets was acceptable.

(3) Final Inlet Configuration Test Results--The final inlet configuration had Auxiliary Lip Number 3 (Figure 22) mounted on faired struts in both forward inlets but the rear inlets were clean.

Test results confirmed that with zero relative wind the pressure loss was substantially zero for all inlets in the final configuration. Tests simulating pure sideward motion (β = 90°) at speeds up to 40 knots indicated that losses in this condition were generally very small, not exceeding a local total pressure loss coefficient (LTPLC) of 0.2. Tests with β = 45° and speeds up to 55 knots indicated losses were associated with the auxiliary lip and the high aircraft centerline lip in the downwind rear inlets. In general, however, the losses were again small in the range of freestream velocity of interest.

Total pressure loss, and total and static pressure distortion coefficient data plotted versus V_0/V_2 (Reference 21) show that, as expected, increasing freestream-to-inlet velocity ratio has a consistently adverse effect on TPLC as well as on TPDC and

SPDC. Test data correlation for the forward and rear inlets, obtained at the engine power and windspeed conditions given in Table III is shown to be excellent for both TPDC and TPLC. Although the SPDC data shows appreciable scatter, this has been shown to be almost exclusively due to errors arising from the combination of the use of mercury manometers and the data collection and reduction process. It is notable that the forward inlet suffers greater pressure losses than the rear inlet despite the lower values of total pressure distortion indicated.

The effect of pitch attitude at high speeds indicate that for the forward inlets, pitch has very little effect. As expected from considerations of flow turning, a favorable effect is noted for nose-down attitudes and an adverse effect for nose-up attitudes.

The influence of yaw angle at high speeds indicate that the upwind inlets invariably suffer greater TPDC's than the zero yaw values but that downwind inlets give reduced TPDC's while TPLC's do not follow this pattern.

Plots of TPL, TPD and SPD obtained from the faired high-speed coefficient results are presented in Reference 18. The plots indicate that for fixed values of V_2, corresponding to fixed engine speeds, TPL, TPD and SPD increase progressively with V_0. While the effects of engine speed on TPL and TPD are variable and dependent upon V_0/V_2 and on the inlet location, the effect on SPD is invariable and SPD increases with increasing engine speed.

Figures are presented in Reference 19 that show envelopes of TPD and SPD for typical accelerating or decelerating transitions at maximum and minimum weights. The plots show that TPD values approach the engine manufacturer's recommended limit of 10 percent only at airspeeds of 180 knots on the rear inlet. The values of SPD are generally higher than the recommended limit of 5 percent but the engine manufacturer has reviewed the data and has indicated that these should not lead to engine operating problems. Confidence in the adequacy of the inlet design evolves from the fact that engine stalls and surges during forward flight were absent throughout the program, and that numerous engine starts and accelerations at 200 knots relative windspeed were successfully accomplished. Also, engine vibration levels were always within the manufacturer's recommended limits.

4. CYCLIC TEST PROGRAM

This section presents a summary of the cyclic tests performed on the propulsive and reaction control systems of the XV-4B research aircraft, utilizing a test stand which permitted simulation of the aircraft installation. The detail results of the test program are presented in Reference 21. The objective of the tests was to provide a base line from which safe flight maintenance and inspection procedures could be formulated for the entire aircraft research program. The tests performed simulated XV-4B hover and transition maneuvers and subjected the aircraft propulsive and reaction control hardware to repeated operating cycles in order to establish reliability of the components. The program consisted of a 35 hour test comprising straight through cruise tests with lift engine starts for 5 hours, reaction control and vector nozzle cyclic tests for 25 hours, and diverter valve cyclic test for 5 hours.

a. Test Stand

To accomplish the cyclic tests, a full scale welded steel test fixture was designed and fabricated. As shown in Figure 29 the test fixture consisted of:

- A simulated mid fuselage section containing the four fuselage mounted lift engines and the diverted tail pipe system for the cruise engines with thrust vectoring nozzles. The nozzle drive system was capable of vectoring the nozzles ±15° from a neutral position.
- Two simulated nacelles containing the lift cruise engines, each with cruise exhaust nozzle and diverter valve.
- A simulated fuselage and wing structure containing the aircraft mounting system for the bleed air manifolding and reaction control valves.
- Appropriate control mechanisms for the engine and reaction systems. The control for the thrust vectoring lift nozzles was the airvehicle system, all others were simulated.

The VTOL Test Facility previously described in Section 3 was also used for the Cyclic Test Program.

b. Test Schedule

Prior to performing the cyclic tests on the aircraft propulsive and reaction control components it was necessary to establish satisfactory operation of the YJ85-GE-19

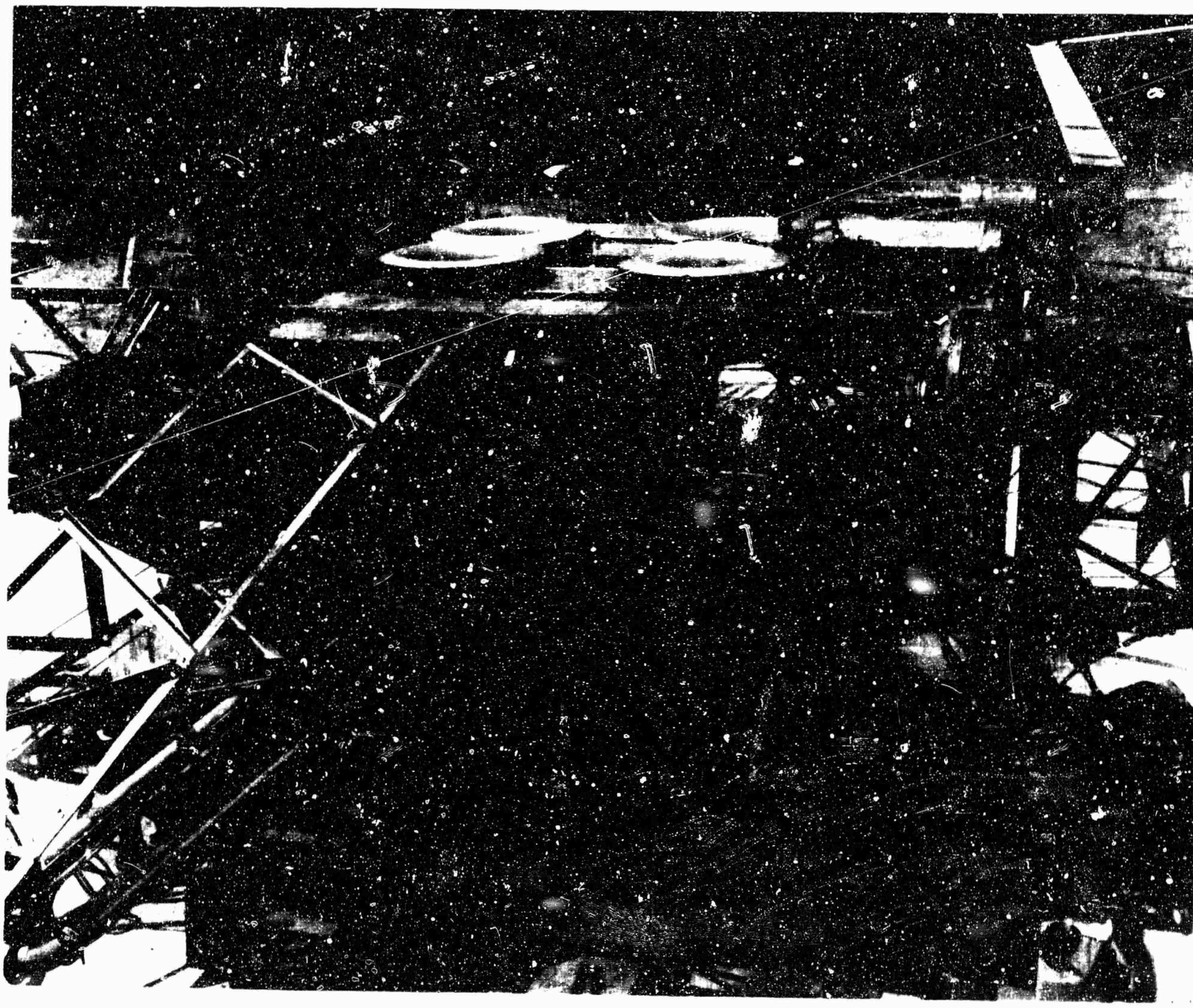


FIGURE 29 - CYCLIC TEST STAND - R/H SIDE VIEW

engines installed in the test stand in a program of preliminary tests. Following installation of the engines and associated hardware, tests were conducted to determine engine bay temperatures, engine vibration, inlet guide vane scheduling and engine exhaust nozzle trimming requirements for all engines for operating with 7.5 percent bleed air in the lift mode and for lift/cruise engine operation with 0 percent bleed air in the cruise mode. Upon satisfactory completion of these tests, control survery were made which indicated that it was possible to create an acceleration stall on the lift engines. The acceleration stall problem was traced to a flow restriction in the inner stage bleed ducting system and subsequently corrected by redesign. As discussed in Section III hot gas reingestion was also encountered creating high power stalls on the #5 and #6 lift engines. This problem was traced to an improperly configured baffle in the test facility exhaust pit and subsequently corrected, alleviating the stall condition.

The Cyclic Test Program was divided into three parts, each designed to demonstrate the operation of the propulsive and VTOL control system components in one operating mode.

Part I Straight through Cruise Testing (transition from conventional flight to hover modes)

Part II VTOL Testing - cruise engines diverted (hover mode)

Part III Diverter Valve Cycle Test (flight to hover mode; lift/cruise engines only)

The tests were performed in the order dictated by availability of test specimens and convenience, thus Part III, requiring operation of the cruise engines only, was run first, followed by Part I and Part II.

(1) Part III, Diverter Valve Cycle Test--Part III consisted of a 5-hour cyclic test of the lift/cruise propulsive exhaust system. Aircraft components tested included the lift/cruise engine assemblies and associated ducting, diverter valves, cruise tailpipes and nozzle assemblies, and lift duct and nozzle assemblies.

(2) Part I, Straight-through Cruise Test--Part I consisted of a 5 hour cyclic test of the cruise propulsive exhaust systems, the (4) lift engine assemblies, associated bleed ducting, tailpipe and nozzle assemblies and the mid section fuselage length bleed air duct.

(3) Part II, VTOL Testing - Cruise Engines Diverted--The Part II test hardware included all lift/cruise propulsive exhaust system components, all lift engine system components, and the complete bleed air ducting system and associated reaction control valves. These components were subjected to 25 hours of testing.

c. Test Results

After numerous hardware failures and subsequent redesign in both the exhaust system and bleed air ducting system, the 35 hour Cyclic Test Program was successfully completed. The test program resulted in the establishment of inspection and maintenance procedures as presented in Table IX of the Appendix and provided a high level of reliability in the aircraft reaction control system. Briefly, the major failures and associated corrective actions experienced during the course of the program are presented below.

(1) The specified five hours of Part III diverter valve cycle testing was completed with no indication of mechanical problems, however, after one hour of Part I straight through cruise testing visual inspection disclosed cracks in the valve casing and in the region of the diverter door. The valve casing cracks, found to be attributable to defective welds, were subsequently repaired and final test results indicated that no further valve casing deterioration had taken place.

Cracks in the diverter valve doors resulting from differential expansion between the door skins and drive axle continued throughout the test program. The final modification, which proved successful, consisted of a plate welded on three sides over the cracked area, in four locations on each door to provide for the required operational differential expansion. To insure safe operation, an inspection period of 8 to 12 operational hours was established for each valve installation.

(2) Both the cruise and lift/cruise diverter duct expansion bellows failed repeatedly in both lift/cruise engine installations and none of the modifications introduced during the cyclic test series provided a satisfactory solution. The established life of the bellows was a minimum of nine hours. As a result of this experience, the originally installed 1 ply bellows were replaced with a dual laminated bellows design and a 5 hour inspection period was established for all installations. A typical bellows failure experienced during the test program is presented in Figure 30.

(3) With the exception of one design change improving the seal, the vector nozzle assembly performed satisfactorily throughout the test program.

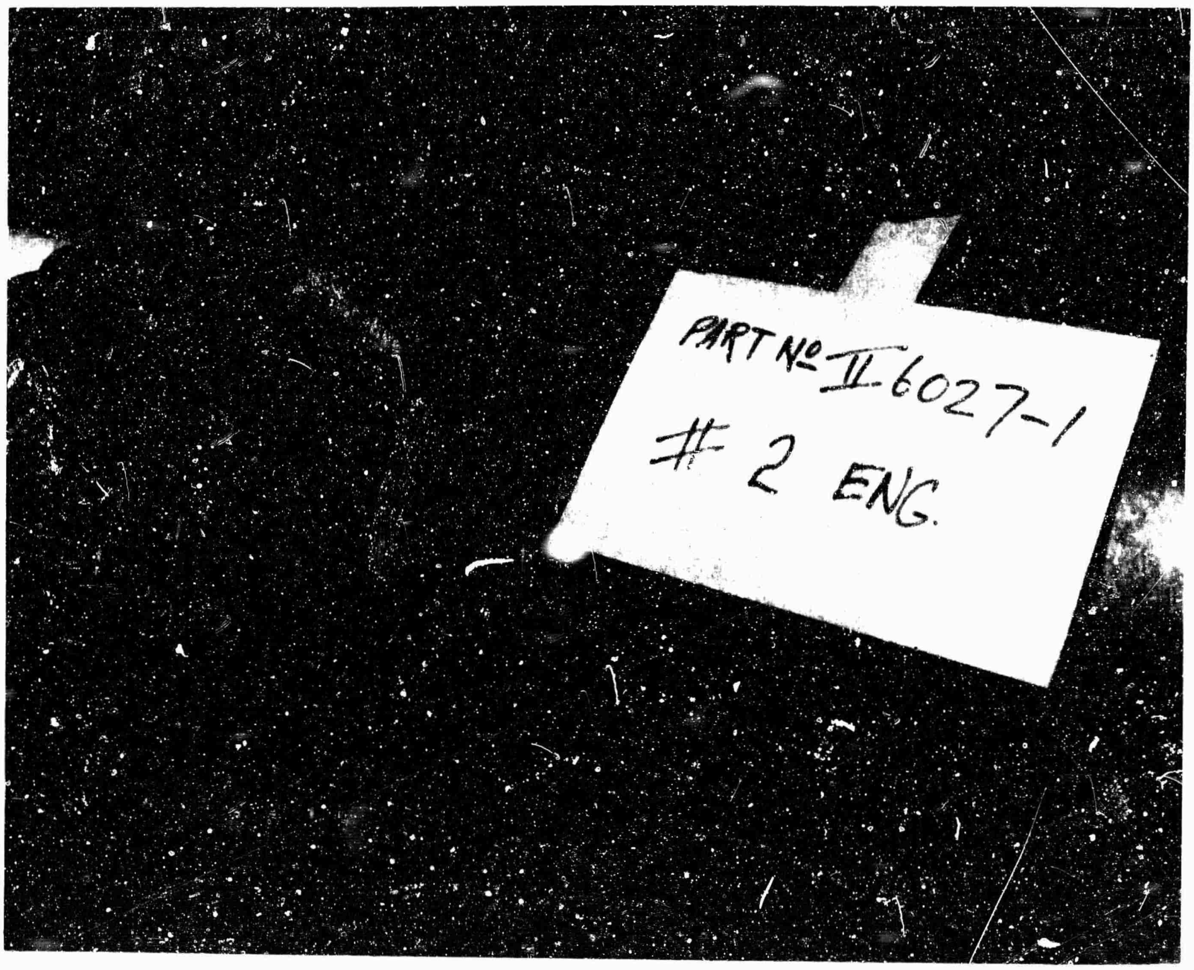


FIGURE 30 - LIFT ELBOW EXPANSION BELLOWS FAILURE

(4) After only 12 hours of testing in the Part II test series a major failure occurred in the Mid Fuselage Length Bleed Air Duct System. The failure occurred following a period of severe engine stalls on lift engine numbers 5 and 6. Following repair of the main bleed air duct, VTOL testing was reinitiated, and after four additional run cycles had been performed it was discovered that the customer bleed duct on the number one engine had failed catastrophically. After general inspection of the test installation revealed multiple failures in the ducting system, testing was discontinued to await a complete redesign and remanufacture of the entire ducting system. Typical failures encountered in the ducting are presented in Figure 31.

With the new ducting system installed, the 25 hour reaction control cyclic testing was completed satisfactorily, and fully demonstrated reliability of the redesigned bleed air ducting system.

Results of the cyclic test program and sonic tests discussed in Section VII-6 fully substantiated that the life expectancy of the ducting system far exceeded the design life of the aircraft.

5. REACTION CONTROL VALVE TESTING

The results of the reaction control valve technology development program reported in Reference 22 were applied to the XV-4B control system. Reference 22 describes the development of a typical VTOL control valve based on the XV-4B forward pitch valve. The tests described breifly here were designed to investigate the thrust performance, leakage, actuating torque characteristics and mechanical behavior of the valves. The resulting valves developed in this program show good thrust performance but have undesirably high leakage and actuating torques. In addition to the tests reported in Reference 22 the XV-4B reaction control valves were calibrated so that thrust and airflow could be determined from reference air flow measurements.

a. Test Setup

The test setup used in both the development and calibration tests is shown schematically in Figure 32. Test instrumentation in the air supply line includes a square-edged orifice with static pressure taps and a thermocouple probe for measuring airflow rate. For the forward pitch and aft pitch/yaw valves a wall static pressure piezometer ring was installed in the air supply duct just upstream of the valve attachment flange for the purpose of determining an effective total pressure based on the measured airflow rate, static pressure, temperature, and flow area of airflow entering the valve.

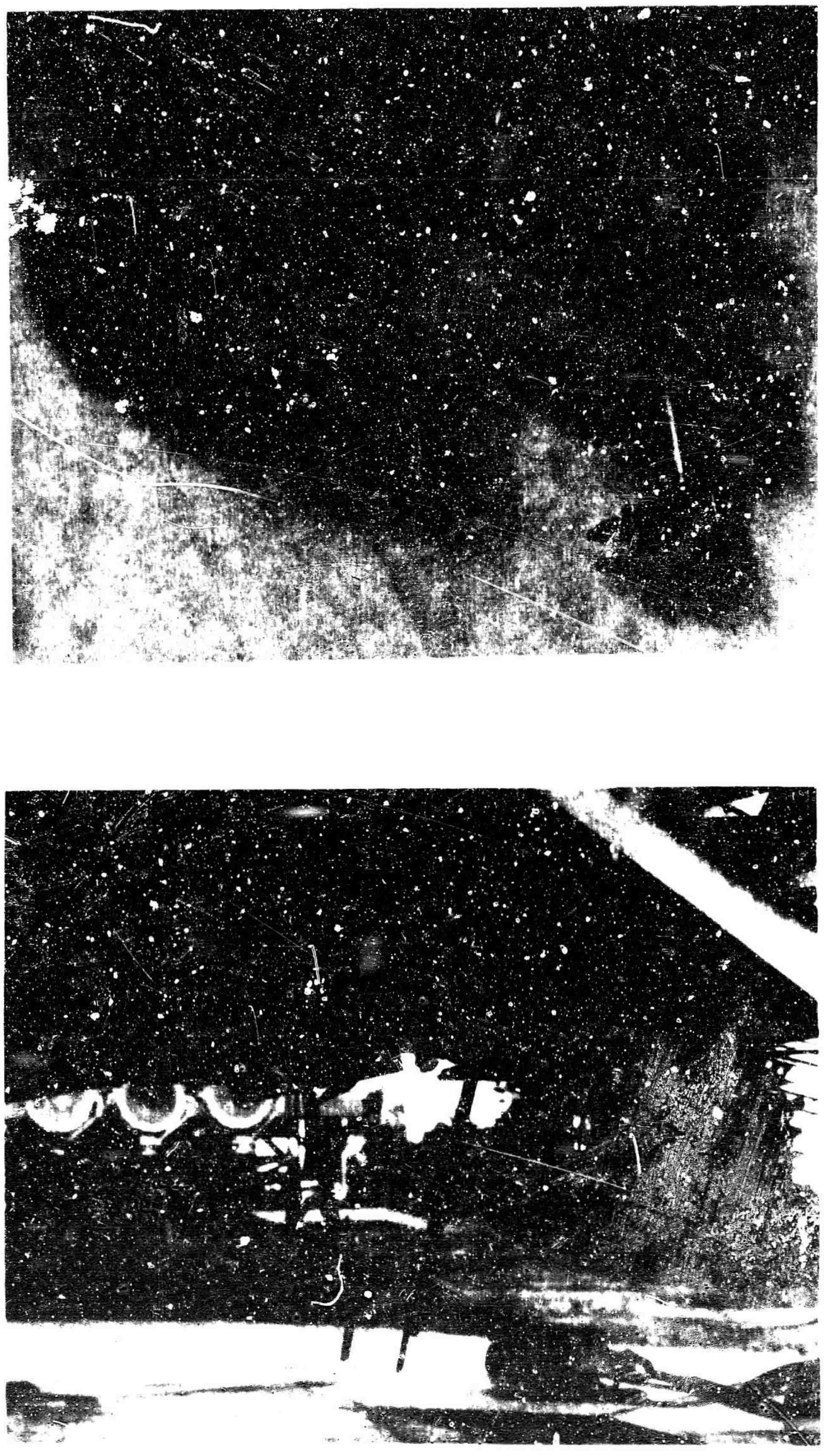


FIGURE 31 - TYPICAL BLEED DUCT FAILURES

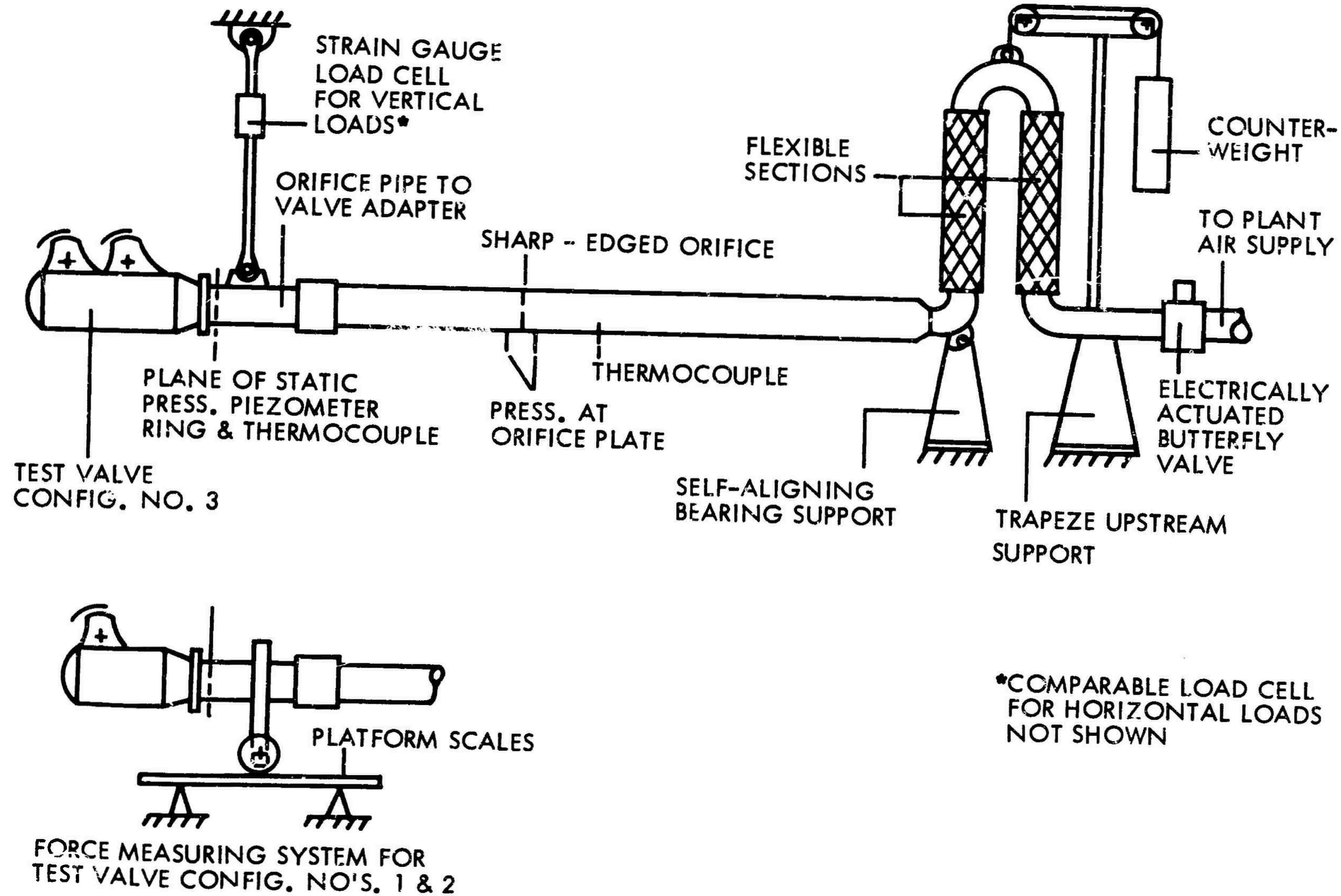


FIGURE 32 - SCHEMATIC LAYOUT OF REACTION VALVE TEST RIG

Strain gage load cells installed in both the vertical and horizontal planes were used for measuring valve thrust. Data readout and recording equipment included manometers and gages for pressure readings, a Brown indicating potentiometer pyrometer for hot air temperatures, a strain gage indicator and channel selector box for thrust measurement, and a recording oscillograph for valve visor position time histories.

The valves were tested from the fully closed position to the fully open position in approximately 10 percent increments, with the valve inlet controlled in 10 psig increments from 0 to 100 psig. The nominal air was 500°F in all tests. In the case of the forward pitch and aft pitch/yaw valve, calibration data were not obtained over the entire range of operation due to air supply limitations at the test facility.

b. Development Tests

(1) Valve Design--The developmental program was comprised of three test series. Between each of these series the valve was modified to correct problems. The first design had two visors, one on each side of the fixed central flow splitter plate/ support strut. To open the valve, the visors moved in opposite directions away from the splitter. Excessive deflections, especially in the visor shaft, bearing supports, and linkages, were encountered in this design. As a result, the visors cocked and did not close evenly against the splitter. Leakage resulted and the valve was redesigned.

The second design used a single visor with stiffener supports and actuating linkage. The clearance between the visor and the nozzle housing was held to a minimum to prevent leakage. Thermal stresses caused visor binding in this design.

In the third design, the nozzle housing incorporated a floating section which was intended to ride on the visor and give minimum clearance and leakage. However, the floating section also cocked and caused the valve to bind. The final tests were conducted with the floating section locked in the position that provided minimum clearance with no rubbing. It is the results of these tests that are discussed below.

c. Test Results

Figures 33 and 34 show typical performance in terms of thrust coefficient and actuating torque as a function of position and pressure ratio or pressure. The thrust coefficients, Figure 33, fall off at the low nozzle openings because of leakage. The actuating torque curves, Figure 34, indicate the high friction encountered in this valve. At any

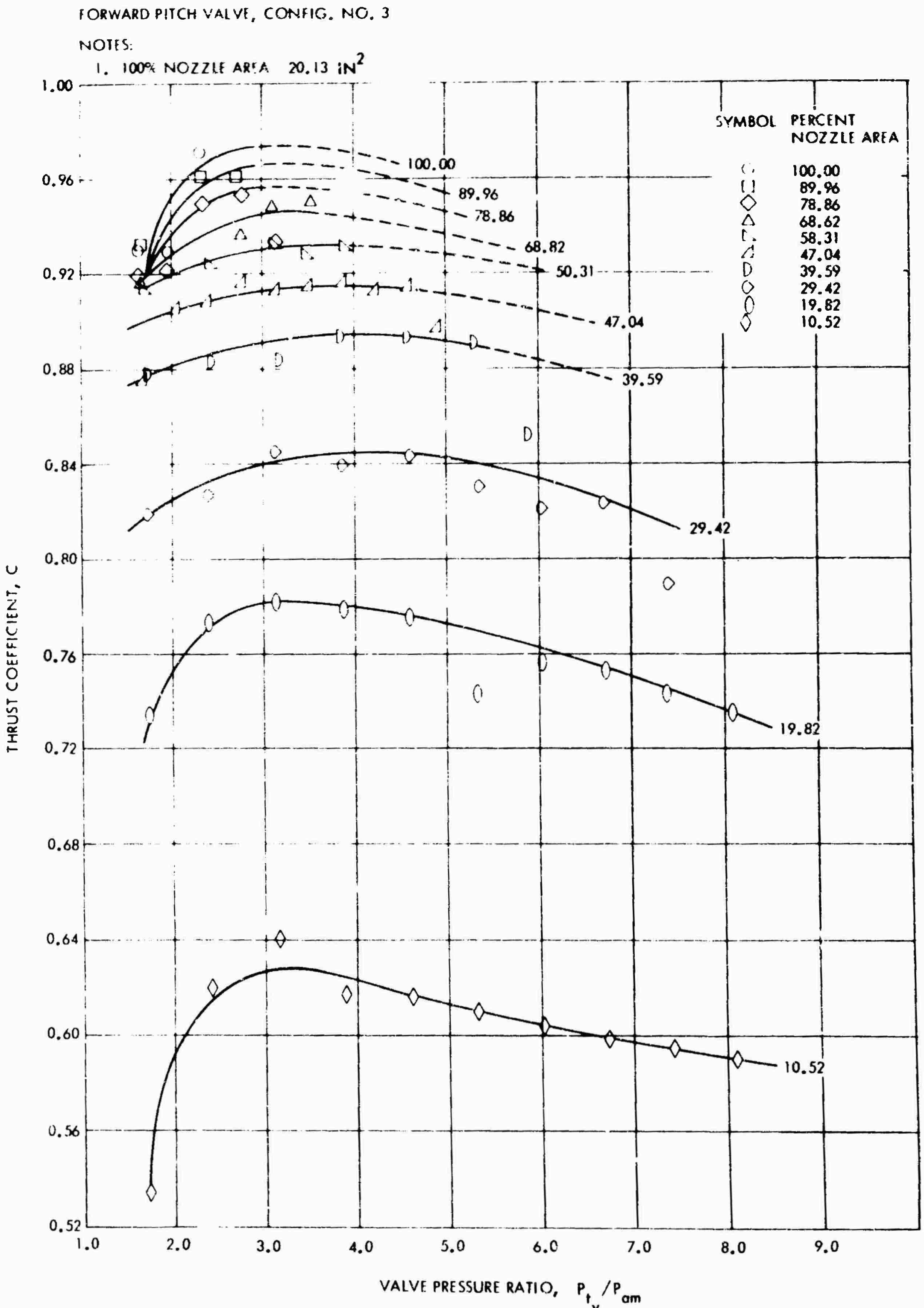


FIGURE 33 - REACTION CONTROL VALVE THRUST COEFFICIENT

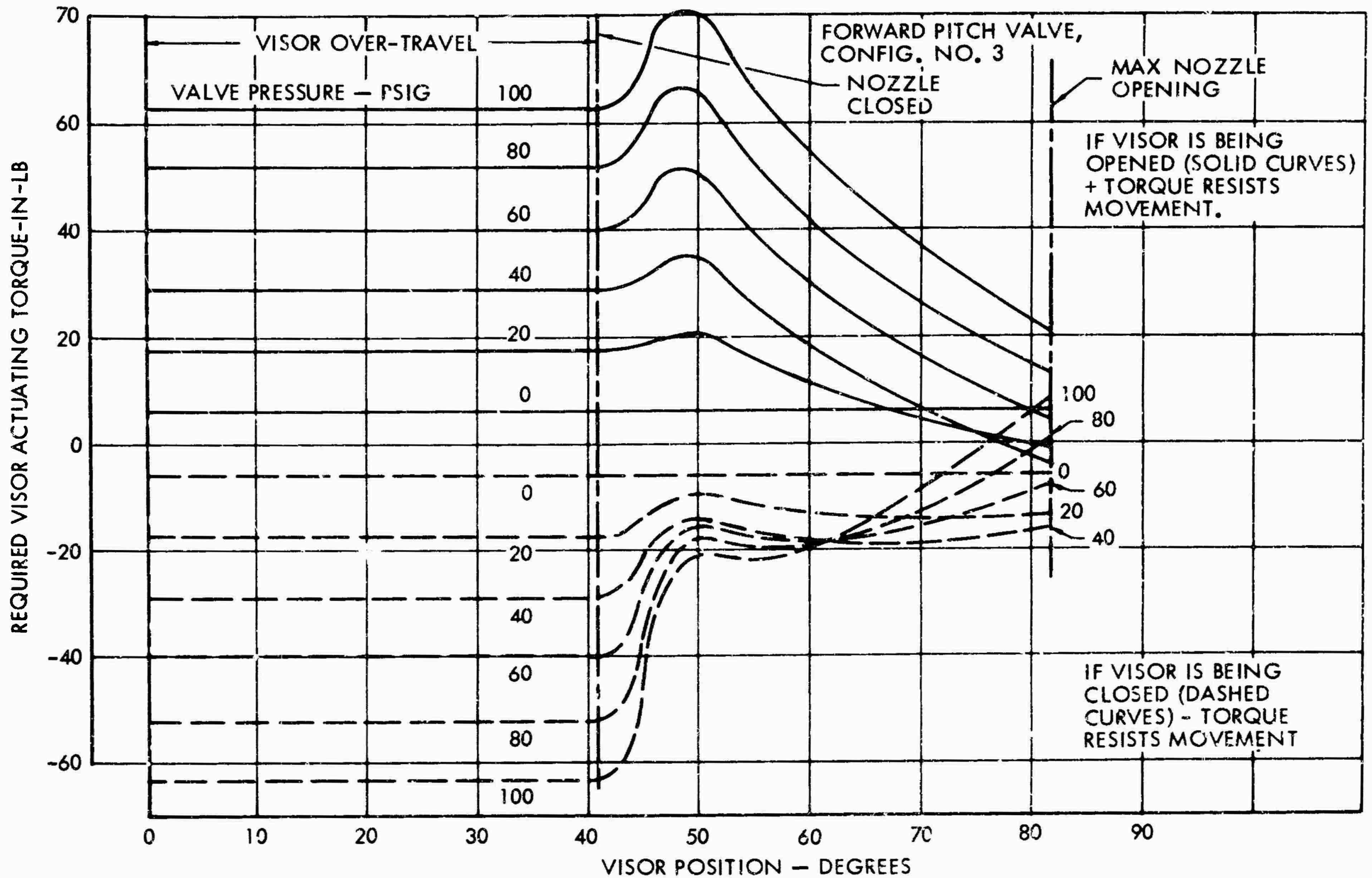


FIGURE 34 - REACTION CONTROL VALVE ACTUATING TORQUE

valve position and pressure, the frictional torque is equal to the difference between the torque required to move the valve toward the open position and the torque required to move it closed. If there were no friction, the two sets of curves would be coincident and aerodynamic forces would move the valve in one direction or the other if it were released.

Accurate leakage measurements were not obtained during the forward pitch valve developmental tests. The leakage was low enough so that valid orifice plate pressure differentials could not be obtained with the orifice plate sized for the normal nozzle flow range, the only orifice plate used in the program. However, leakage was measured by means of the orifice plate during the aft pitch and yaw valve calibration tests described in Reference 18. These results, shown in Figure 35, were used to develop the leakage allowance used in the reaction control system performance analyses presented in Section V.

6. STRUCTURAL TESTS

This section briefly describes the static tests of the control surfaces, control linkages, bleed air ducts, and drop tests of the main and nose landing gears. Dynamic tests of the airframe structure and bleed air system are covered under other headings.

a. Controls Static Tests

Both the aerodynamic surfaces and their actuating linkages were proof tested to their respective aerodynamic limit loads, as reported in detail in Reference 23. Single point chordwise loading was applied at the center of pressure and the spanwise loading was accounted for through a whiffle tree arrangement. Load was applied to the whiffle tree through a cable-pulley arrangement attached to a loading tray. Figure 36 is typical of the loading scheme used in the tests.

Reactions to the surface loads were provided at the cockpit controls, except for the flaps, where a dummy flap actuator was used. Stiffness data were obtained for the complete systems and were also recorded from each of the control servos to its corresponding control surface. This latter data provided a deflection pattern for each control system for analysis of system characteristics in the fly-by-wire mode of operation.

The rudder, elevator, aileron, and flap system tests demonstrated that the structure was adequate for flight. No permanent deformation was experienced in any areas except

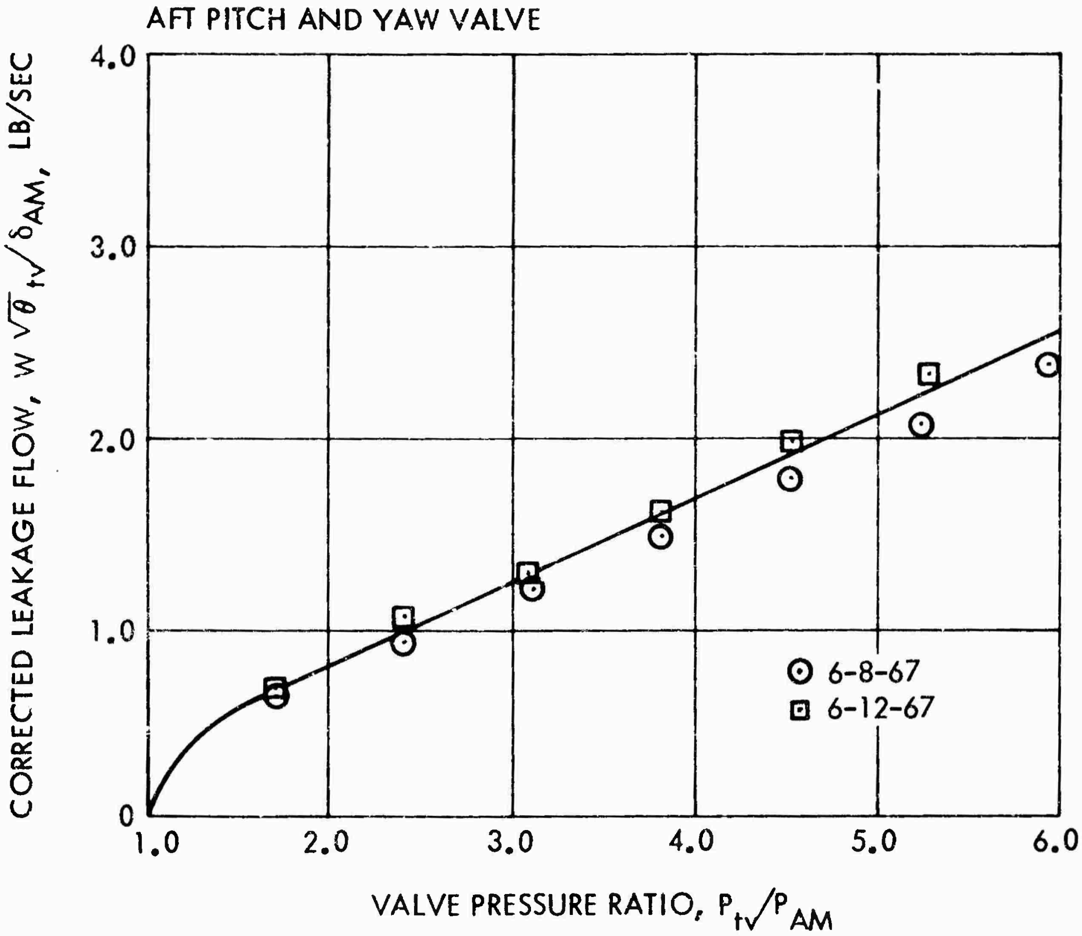


FIGURE 35 - MEASURED LEAKAGE, REACTION VALVE

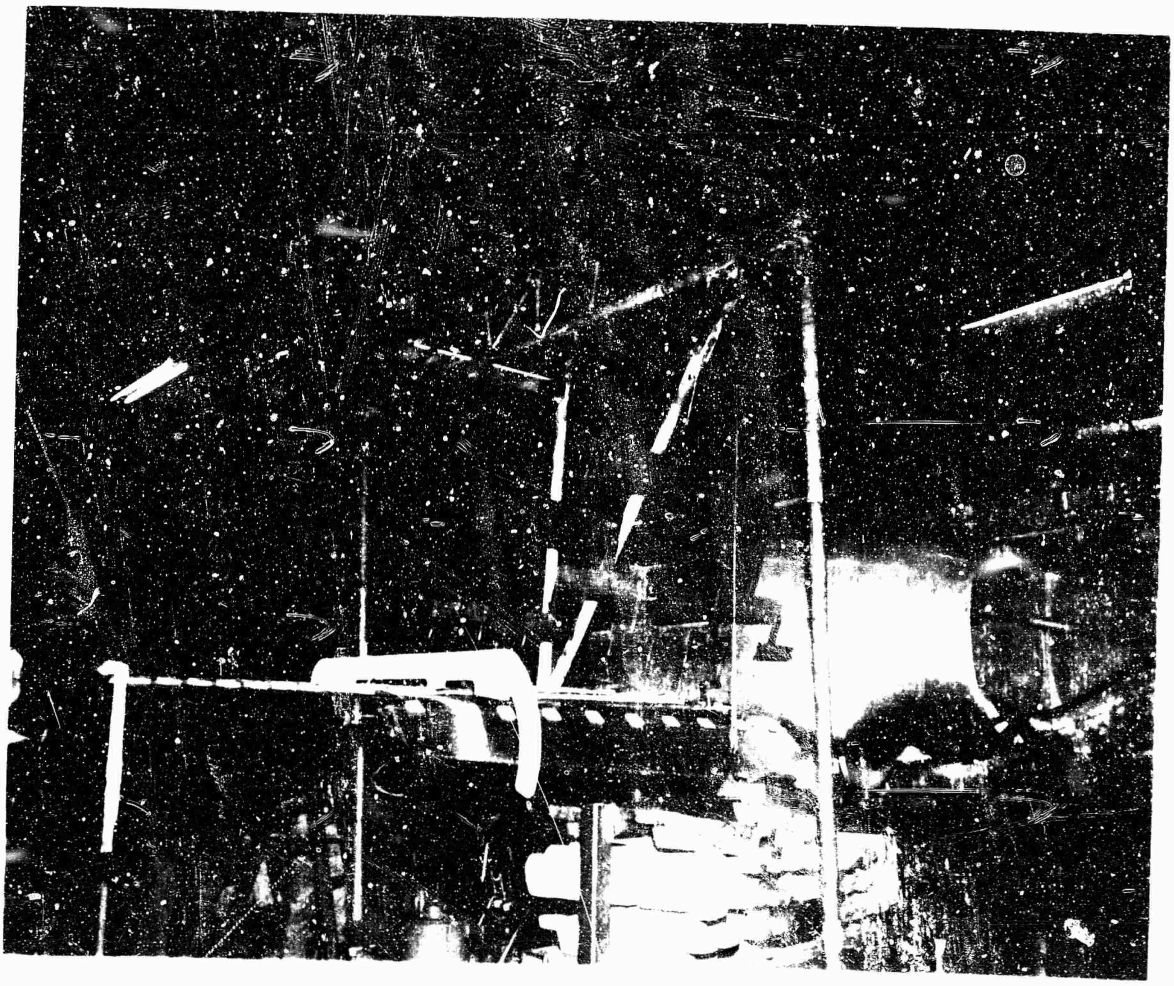


FIGURE 36 - FLAP PROOF TEST FIXTURE

in the elevator system where minor modifications to the original configuration were necessary. Figure 37 shows the results of tests of the rudder system. These results are typical of those of other systems and reflects the incremental deflections attributable to the several parts of the system.

b. Bleed Air Duct Tests

The extensive dynamic testing of the bleed air duct system is described in paragraph 8 of this section. In addition to these tests, and as a consequence of early cyclic testing experience, the final ducting design was also subjected to static proof and burst tests.

The final configuration of the ducting was manufactured and tested by the D. K. Manufacturing Co. of Batavia, Ill. As a result of cyclic test experience this ducting had increased wall thicknesses; improved transitions, and reduced weld joint discontinuity as compared to the original ducts.

All deliverable hardware was proof tested to limit loads based on a maximum operating pressure of 102 psig. The burst tests were conducted utilizing test specimens. To compensate for the operating temperatures while testing at room temperature, the test essures were adjusted to account for the loss of strength at the higher temperatures. In the burst tests the larger ducts, made of Inconel X750 were tested to 292 psig and the smaller ducts, made of 321 stainless steel, were tested to 350 psig. These pressures, adjusted for operating temperatures, corresponded to 2.5 times the maximum operating pressure. None of the test specimens failed and after the pressure had been applied for 5 minutes the ducts were checked for leaks and found satisfactory.

c. Landing Gear Drop Tests

The landing gear was designed and built to the Contractor's specifications by the Howmet Corporation of Pamona, Calif. Major items from other aircraft landing gear that had been analyzed and tested and that could be modified to the XV-4B configuration were utilized in the design.

The Howmet Corp. conducted the drop tests to demonstrate the energy absorption characteristics of both the main and nose gear. The main gear was tested for simulated level and taildown attitudes in both conventional and VTOL landings and the nose gear

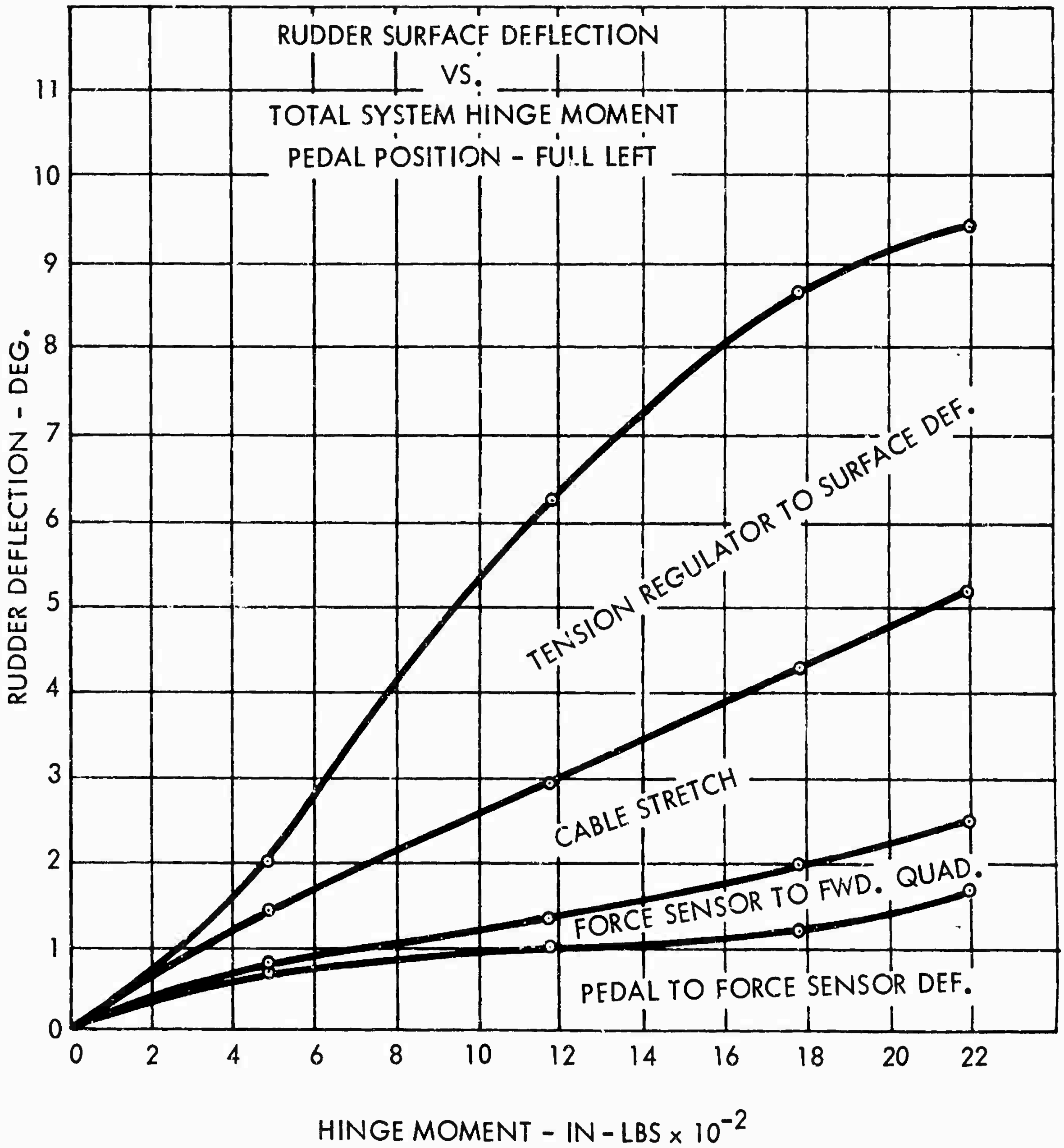


FIGURE 37 - RUDDER SURFACE DEFLECTION VS. TOTAL SYSTEM HINGE MOMENT
PEDAL POSITION - FULL LEFT

was also tested for both conventional and VTOL landing loads. The drop weights for the conventional landing tests corresponded to an airplane weight of 12,000 pounds and those for the VTOL landing tests corresponded to an airplane weight of 12,580 pounds. The drop velocities were 10 and 13 feet per second for the conventional and VTOL tests, respectively.

The results of the test presented in Reference 24 showed that both gear met the specified energy absorption requirements. In addition, inspection of the gear following the tests showed no evidence of permanent deformation or excessive wear.

7. FLUTTER TESTS

Analytical wing flutter, wing divergence, aileron reversal, and empennage flutter analyses of the XV-4B were conducted utilizing high speed digital computers. The results of these analyses showed the airplane to be free of flutter and other aeroelastic instabilities to speeds in excess of 1.2 times the governing limit speed of 260 KEAS or M = 0.53.

To substantiate the analytical predictions ground vibration tests were conducted and a limited in flight flutter investigation was planned.

The ground vibration tests provided data from which the principal resonant frequencies and associated mode shapes could be determined and it was on the basis of these data that lag filters were incorporated in the flight control system to preclude the possibility of airframe/control system dynamic coupling.

The results of the analyses and ground vibration tests are presented in Reference 25.

8. ACOUSTICS, TEMPERATURE, VIBRATION AND SONIC FATIGUE TESTS

The acoustic environment resulting from engine exhaust noise during the VTOL mode was derived from an XV-4B Acoustic Model test. Sound pressure levels in the frequency range of 40 - 10,000 Hz were measured on the model with simulated engines operating in the lift mode. The cruise exhaust noise levels were estimated using measured data from the Lockheed JetStar since this aircraft has similar engines. The engine exhaust noise was combined with the noise contribution from the single engine compressors and the aerodynamic noise to give the total XV-4B acoustic environment.

The predicted acoustic environment was used for sonic fatigue and vibration analyses during the XV-4B design. The predicted vibration environment was derived from these noise levels using empirical noise/vibration correlation methods. Mechanical vibration induced by engine unbalance was superimposed on the acoustically induced vibration levels in the vicinity of the engines. These predicted vibration levels were used as the basis for establishing equipment qualification test criteria for the XV-4B. These criteria were presented in Reference 26.

During the design stage, critical structural and equipment components were analyzed and sized to meet sonic fatigue and vibration requirements. Of necessity, the XV-4B sonic fatigue philosophy differed from that used on a multi-aircraft development and production program. Figure 38 summarizes the steps taken on this program to minimize sonic fatigue problems. Environmental data obtained from the cyclic test stand were used to update the fatigue analysis and resulted in design changes which were incorporated into the aircraft. The most significant of these changes was the bleed air ducting system which was completely redesigned following testing on the cyclic test stand.

Laboratory, engine test stand and aircraft test programs were conducted to substantiate the structure and equipment components. The fatigue life of the bleed air ducting system was verified by a complete evaluation on the cyclic test rig. This program resulted in establishment of operating life times and inspection procedures for the duct systems. A laboratory vibration fatigue test program was conducted to establish the operating lifetime for the wing roll ducts. The results of these test programs were reported in Reference 27.

The aircraft operating environment was measured during testing on the inverted telescope rig. At the same time, dynamic strain measurements were obtained for verification of the structural resistivity to sonic fatigue. This test program was quite extensive and involved measurement of internal noise in the cockpit and aft fuselage equipment compartment, external noise at 15 locations, structural and equipment vibration levels at 37 locations, engine vibration at 20 locations, structural temperature at 56 locations and dynamic strain at 29 locations. The internal cockpit noise levels were utilized in evaluating candidate soundproofing configurations to reduce the operating noise levels. The measured internal noise levels during conventional and VTOL operations are summarized in Figure 39. Exterior noise levels measured during VTOL operation on the ground are also shown in Figure 39 for the areas of the fuselage structure in the vicinity of the lift engine exhausts.

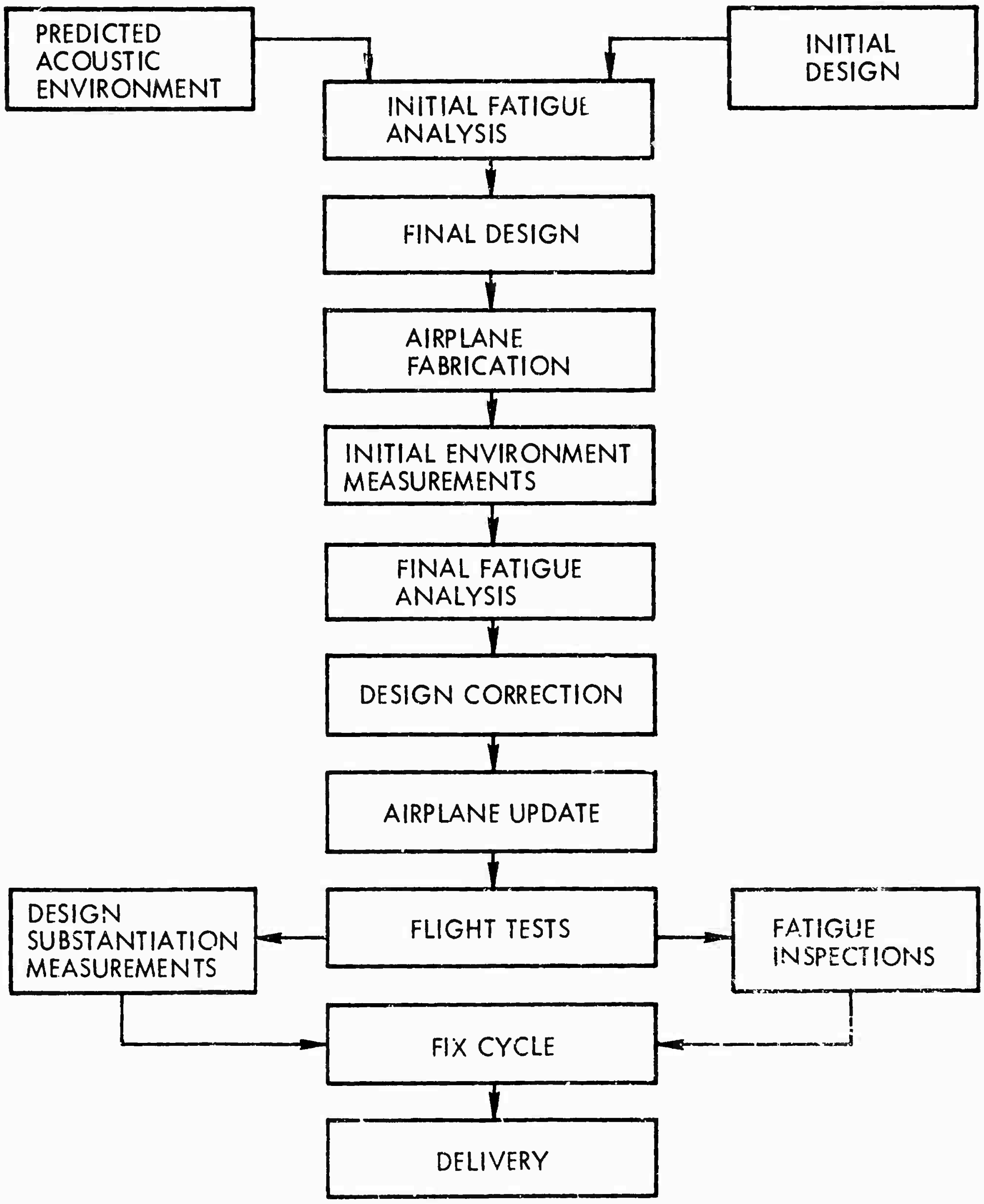


FIGURE 38 - XV-4B FLOW DIAGRAM FOR SONIC FATIGUE DESIGN

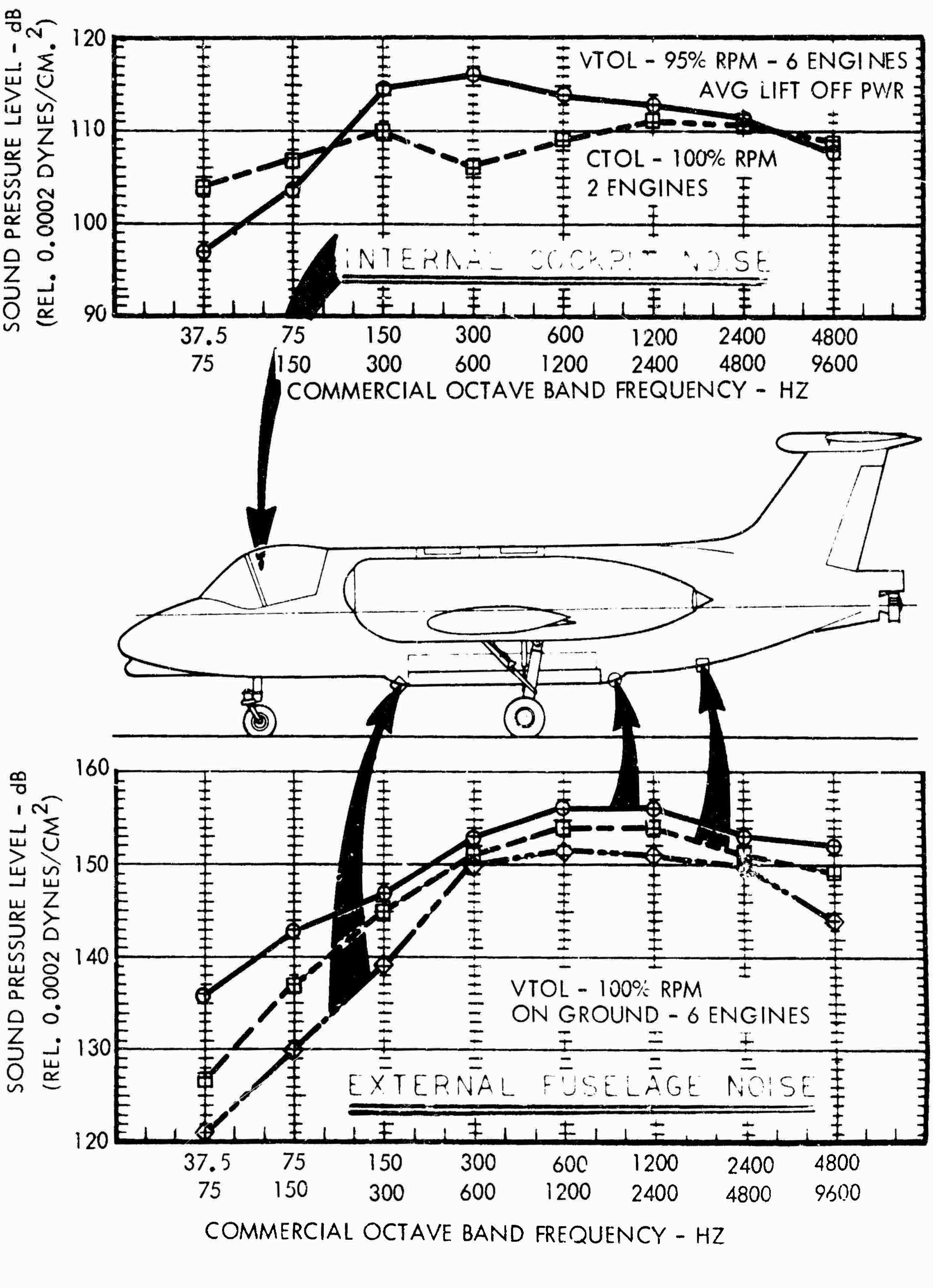


FIGURE 39 - MEASURED NOISE LEVELS

Temperature measurements were made with temperature-sensitive templates located over the aircraft structure and exposed to environmental temperatures resulting from both conventional and VTOL engine operation. The template locations and the temperature contrours determined from the maximum measured temperatures are shown in Figure 40. The highest temperature recorded occurred at the edge of the inner lift engine exit door, at which point the temperature was approximately 300°F.

Dynamic strain data were used with the temperature data in re-evaluating the fuselage structure in the vicinity of the engine exhausts to determine the actual operating lifetime. Engine vibration levels were compared with allowable displacement tolerances for final approval of the installation from a vibration standpoint. Structural and equipment vibration levels were compared with the predicted levels and evaluated for potential fatigue or operating limitations. The acoustics, vibration and sonic fatigue effort was reported in detail and all the aircraft measurements were included in Reference 28.

Vibration data measured on the XV-4B equipment components were used in an extensive qualification analysis of all equipment items, particularly those involving safety-of-flight. Prior to first conventional flight, each item was evaluated for performance and environmental acceptability and the results presented in Reference 11. Following the measurement program, the above data were used to establish the final qualification status of each item, with the results presented in a revision to Reference 11.

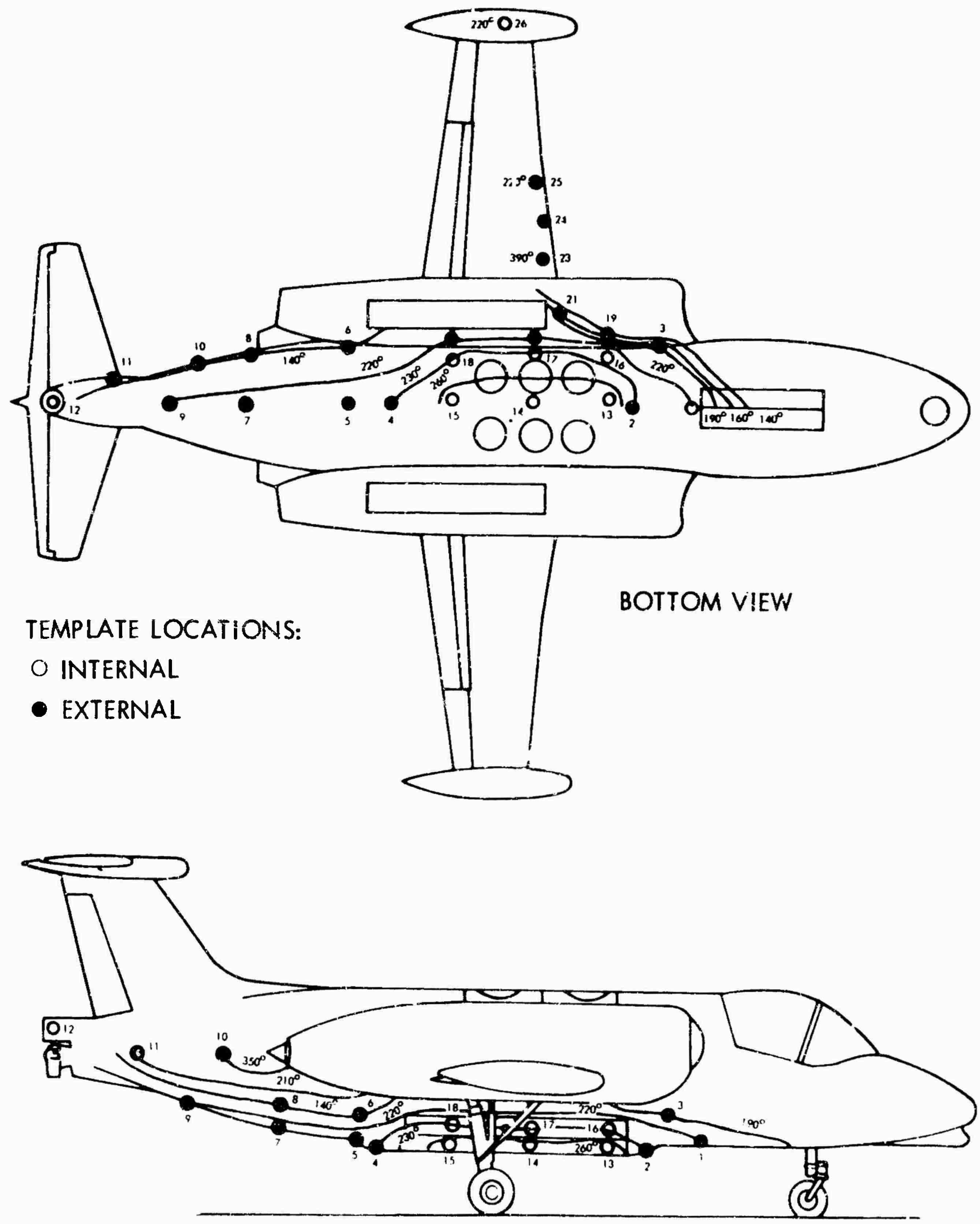
9. ESCAPE SYSTEM TEST PROGRAM

a. Description

An escape system development test program described in Reference 29 consisting of six test firings was conducted to demonstrate:

- Seat-cockpit installation compatibility.
- Trajectory and operational performance of the ejection seat.
- Qualification testing of the total escape system under static and dynamic conditions.

The testing was conducted utilizing anthropomorphic dummies and using an XV-4B forward fuselage test section, as depicted in Figure 41. With the exception of the last firing, all tests were conducted under static conditions. Instrumentation was principally



NOTE: TEMPERATURE CONTOURS ARE MAXIMUM LEVELS EXPERIENCED DURING CTOL AND VTOL MODES. (IN°F)

FIGURE 40 - MEASURED TEMPERATURE CONTOURS

TEST NO. 5, REVISED ONE PIECE CANOPY.

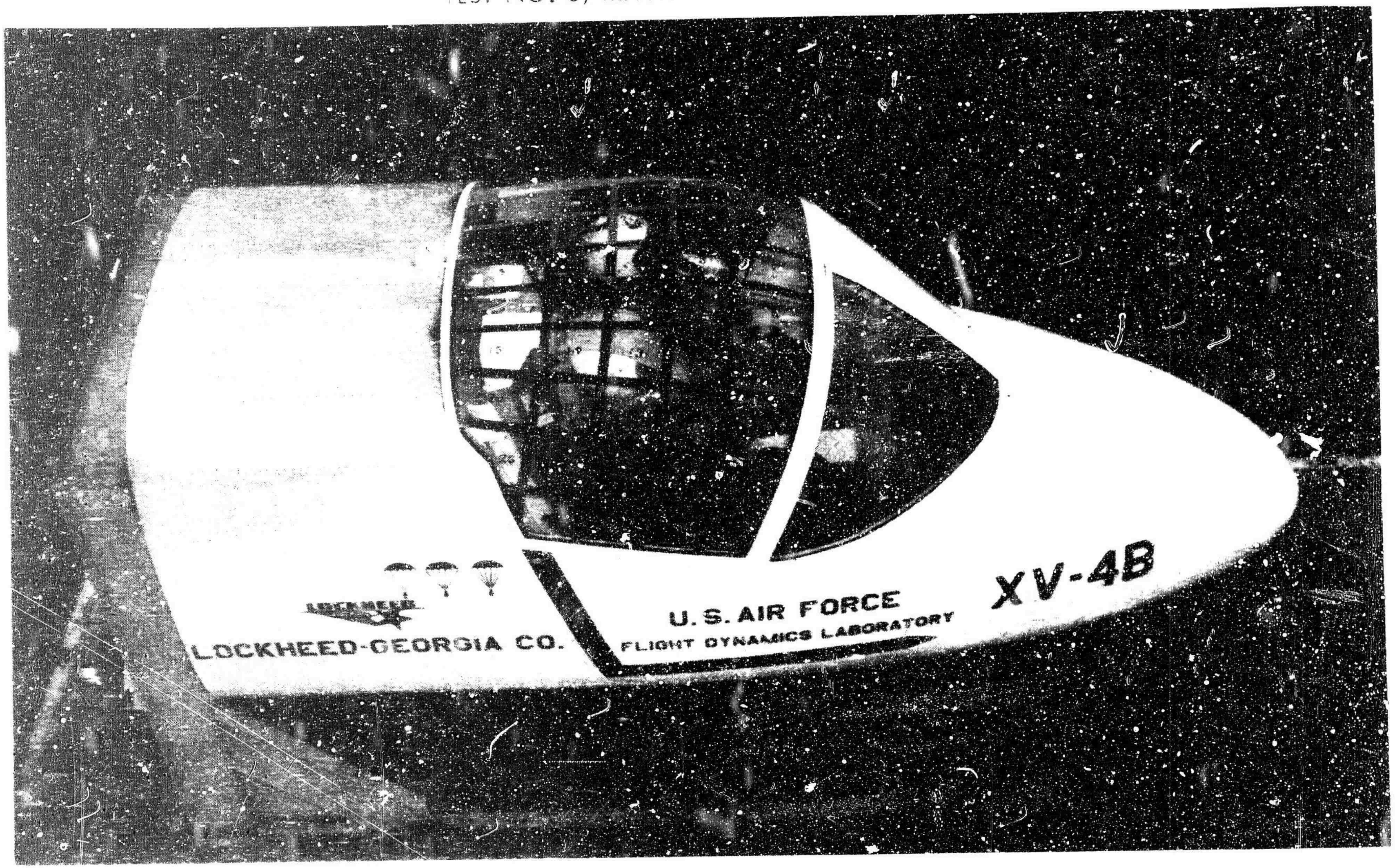


FIGURE 41 - TEST SETUP - COCKPIT TOP VIEW

photographic though various devices were used to measure loads, blast pressures, ejection path clearances and temperature. Detail test results are presented in Reference 13.

b. Test No. 1 (5th Percentile Dummy)

The primary purpose of this and the following test was the demonstration of seat and cockpit installation and structural compatibility. Ballasted for a low CG condition, the dummy was successfully recovered at a height of 63 feet after reaching a peak trajectory of 132 feet.

A failure of the aft canopy frame was attributed to the impinging rocket blast. This also caused buckling of the fuselage skin aft of the crew station. Crewmember clearance was satisfactory and penetration through the canopy glass was without any deleterious effects or physiological damage to either man.

c. Test No. 2 (95th Percentile Dummy)

All escape system functions were normal and clearances were determined to be adequate for the larger dummy. Though the man would have survived, recovery was marginal at 6 feet above the ground. Analysis indicated that the stabilization system was overcorrecting for CG excursions and subtracting an excessive amount of energy from the rocket system. Design changes were initiated to reduce these forces and to incorporate a snubbing seat-man separation system for r e positive operation. The addition of the U. S. Navy qualified NB-11 parachu with ballistic spreading device was also incorporated to reduce parachute opening time.

d. Test No. 3 (5th Percentile Dummy)

Utilizing a high CG to evaluate performance in a critical portion of the envelope and to evaluate the interface compatibility of the various components, this firing resulted in a dummy peak trajectory of 197 feet with a fully-inflated parachute achieved within 4.6 seconds.

A failure was again experienced with the aft canopy frame and necessitated design modifications deleting the center beam and adding a rocket blast shield to each crew station. A failure of the snubber was attributed to the cutting action of the glass though this had little influence on system performance.

e. Test No. 4 (95th Percentile Dummy)

The interface tests concluded with the firing of a 95th percentile with a CG 0.76 inches below nominal to demonstrate a low trajectory. All systems functioned as predicted to attain a peak height of 110 feet, recovery height of 63 feet, and full parachute inflation in 5.3 seconds.

f. Test No. 5 (95th and 5th Percentile Dummies)

To get both crewmen out of the aircraft as quickly as possible, a command sequencing system with a one-half second time delay was developed and utilized in this static, qualification test firing. The right hand crewman was a 95th percentile with a HI CG (0.67 above). The second man was a 5th percentile with a LO CG (0.70 below). In combination with a quick opening parachute, this total system provided capability for two crewmen to exit a disabled aircraft rapidly. The one-half second delay insured that the last man to leave would not be disabled by portions of the first seat system.

Performance during this firing was:

	95th	5th
Peak Trajectory	148 ft.	156 ft.
Recovery Height	80 ft.	136 ft.
Inflation Time	5.3 sec.	3.9 sec.

The time delay between ejections was determined to be 0.56 seconds. The most critical point occurred when the second man passed the empty seat of the first. Separation was 24 feet. The closest approach was further downrange when the distance decreased to 17 feet.

g. Test No. 6 (95th and 5th Percentile Dummies)

A dynamic test firing was conducted on the high speed track at Holloman Air Force Base, New Mexico, utilizing similar dummy configurations as in Test No. 5 and concluded the qualification test program. Using five Lacrosse rocket motors, the test vehicle was accelerated to a speed of 81 knots in one second. This speed approximates the Phase I and II conversion speed in a VTOL transition. The 95th percentile crewman was the first to leave the cockpit and the launch, seat stabilization, separation and recovery were normal.

Approximately 0.57 sec. after initiation of the sequence, the 5th percentile man left the cockpit. Traveling 25 ft. per second faster than the first man, he overtook him 412 feet downrange. Satisfactory performance of all seat functions occurred as anticipated. The second dummy's peak trajectory was 134 feet. Full parachute inflation took place 3.95 sec. after initiation. Recovery height was 103 feet.

As in the preceding test, the most critical point of near-collision during the trajectory came when the second man passed the empty first seat. Separation distance was approximately 6 feet at this point. Both tests validated the necessity for incorporating a one-half second time delay into the command sequencing system.

This test demonstrated that the XV-4B ESCAPAC ID-3 escape system could meet the qualification requirements of the aircraft. During the dynamic tests, the system showed that it had:

. . . . . positive seat-man separation
. . . . . positive parachute deployment
. . . . . ballistic parachute opening, and ....
. . . . . time to full parachute inflation of four seconds.

Upon completion of this test, the overall system showed that it was qualified for the XV-4B. It demonstrated a system that would get one man out of a disabled VTOL aircraft in a minimum time period. In addition, it would get two men out safely without degrading the performance of either one.

h. Test No. 7 (Live Ejection)

An unprogrammed live ejection took place on March 14, 1969. The pilot made an emergency escape while in a left roll leaving the aircraft parallel to the ground. Ejection was approximately 5,000 feet above ground level at an aircraft velocity of 235 KIAS.

The ejection was initiated with the left hand using the lower firing handle. Both feet were on the rudder pedals and the head was slightly down. Launching was solid and the pilot had no particular awareness of going through the canopy except that pieces of glass were observed in front of the face plate. His right knee was bruised through contact and removal of the glass from the canopy. The back sides of the calves were also bruised either by being drawn back during the initial launch acceleration or as the body was blown back by the airstream blast.

Opening parachute shock was moderate to severe and resulted in a momentary loss of visual focus. Descent was at approximately 25 feet per second and accompanied with an uncomfortable parachute oscillation. During descent, the pilot was unable to raise his head due to an interference between the parachute risers and the Robert Shaw-Fulton Company protective helmet (Air Force designation HGU/15/P). The close spacing of the attach fittings on the torso harness and oxygen hose connections were apparently the cause of the problem in this area.

All seat functions operated normally. Separation of the snubber lines was experienced in the same manner as the test firings. From cuts on the pilot's boots, the clearance envelope in the XV-4B may have been marginal when in an uncontrolled maneuver. This phenomenom was not observed in the tests.

10. FLIGHT TEST PROGRAM

This section summarizes the Engineering Flight Test program on the XV-4B aircraft. The test program and results are fully detailed in Reference 30. The program was conducted in accordance with Reference 31 which presents the tests required to define the handling qualities and minimum performance specified for the airplane. Prior to and during the flight test portion of the program an extensive series of ground tests were performed to ensure systems functional integrity and to evaluate total airplane environment in all flight configurations. During the flight portion of the test program the airplane was operated within a velocity-load factor envelope limited by conservative flutter and structural restrictions. Before the loss of the aircraft twenty three flights were completed for a total of 16 hours and 59 minutes flight time.

a. Airplane and Instrumentation

The tests were conducted on the XV-4B aircraft, Air Force Serial No. 24504. The basic airplane configuration is described in Sections II and III of this report. As noted in these sections numerous developmental type changes were made to a number of the airplane systems during the course of testing. Where these changes are significant they will be discussed in the following sections.

The airplane was comprehensively instrumented for all phases of the ground and flight tests and a considerable quantity of data were recorded and compiled. In general, for the ground tests, specific instrumentation peculiar to the particular test was temporarily attached to the airplane and/or components and remote recording equipment utilized. The instrumentation included accelerometers, thermocouples, templates, strain gages, and velocity pick-offs. The devices used to record and/or monitor the instrumentation output included Brown recorders, frequency meters, precise angle indicators, oscillographs, Consolidated Electrodynamics Corporation vibration meters, and magnetic tape recorders. Flight test instrumentation semi-permanently installed in the airplane included provisions for measuring bleed duct pressures and temperatures, aircraft attitudes and attitude rates, airspeed, angle of attack and sideslip, cockpit control surface and actuator positions and numerous other parameters as detailed in Appendix B of Reference 30. In addition to the semi-permanent instrumentation, temporarily installed instrumentation was used on occasion for specific tests. Although all parameters were not available for every flight a total of 69 variables were recorded during the flight program. The recording medium was a tape recorder interfaced with an industry standard IRIG narrow band FM subcarrier oscillator multiplex and other necessary signal conditioning equipment. The recorded data were estimated to have an accuracy of ± 4 percent.

b. Test Facilities

All ground and flight testing was conducted at U. S. Air Force Plant No. 6 which is operated by the Lockheed-Georgia Company. A majority of the static ground tests were performed at the Contractor's VTOL Test Facility described in Section VII. A number of these tests utilized the Inverted Telescope and Balance System described in Section VI. The runway facilities of Dobbins Air Force Base, Marietta, Georgia were used for the ground handling and flight test portions of the program.

c. Ground Tests and Results

(1) Systems Functional Testing--All systems were functionally tested in accordance with the detailed test documents presented in Reference 3. Compatibility problems were cleared by basic design changes or by changes in test procedures. Typical of these problems was the difficulty encountered in functional testing the engine tape instruments. This problem was resolved by a change of test equipment and procedures.

(2) Flight Control System Calibrations and Dynamic Response--Detailed calibration and response testing of the "fly-by-wire" control system was performed to verify control force/surface position relationships and to establish satisfactory gain settings and acceptable response performance. The calibration tests revealed that while the pitch and roll control systems were quite satisfactory the original rudder pedal system was unsatisfactory due to very high friction loads associated with the sliding push rod mechanization. The addition of a supplementary centering spring as described in Section III was partially successful in that centering capability was achieved but the effects of

friction were still apparent as evidenced by non-linearities and a relatively large hysteresis loop. Figure 42 which shows the elevator control system calibration for the right hand stick is typical of the results obtained for the left hand stick and for either stick for aileron control. Figure 43 shows the calibration of the modified rudder control system.

Rate feedback gain calibrations were performed by measuring surface response to simulated rate gyro deflections. All gains were within 10 percent of both the nominal values and these values factored by the gain box adjustment. Table IV shows the original design and actual nominal gains at the time of program termination. The figure includes a similar comparison of other pertinent control system characteristics including authorities, force gains, and rate lags.

Dynamic response tests were performed on each axis by substituting external sinusoidal electrical signals for the force transducer inputs to the computer. A typical example of the results of the tests is presented as Figure 44. The effects of the nominal 0.2 second lag applied to the force input signals is clearly shown. The curves lie close to the theoretical values for the 0.2 second lag up to 2 cps inferring that no other significant lags are present. At higher frequencies some actuator lag becomes apparent. The figure also shows some loss of response performance at low amplitudes. This is common to both the roll and pitch axes and is attributable to lost motion between the servo actuator and control surface. This lost motion is not present on the yaw axis.

Calibrations of the mechanical back-up systems were also made. Figure 45 presents the results for the elevator system and is typical of that obtained for the aileron and rudder systems. The figure shows the high frictional forces existing and the resulting degree of hysteresis. These characteristics are attributable to the factors discussed in Section III.

(3) Structural Shake Tests--A summary of the structural shake tests has been presented in paragraph 7 of Section VII. As noted there, no problem areas were revealed and it was concluded that the airplane would be free of flutter and other aeroelastic instabilities within the demonstration flight envelope.

Control system frequency sweeps between 2 and 100 cps revealed a slight possibility of structural coupling due to rate gyro pick-up of structural feedback. As a precaution, the lag filters discussed in Section III were incorporated in the rate feedback loop of the flight control system.

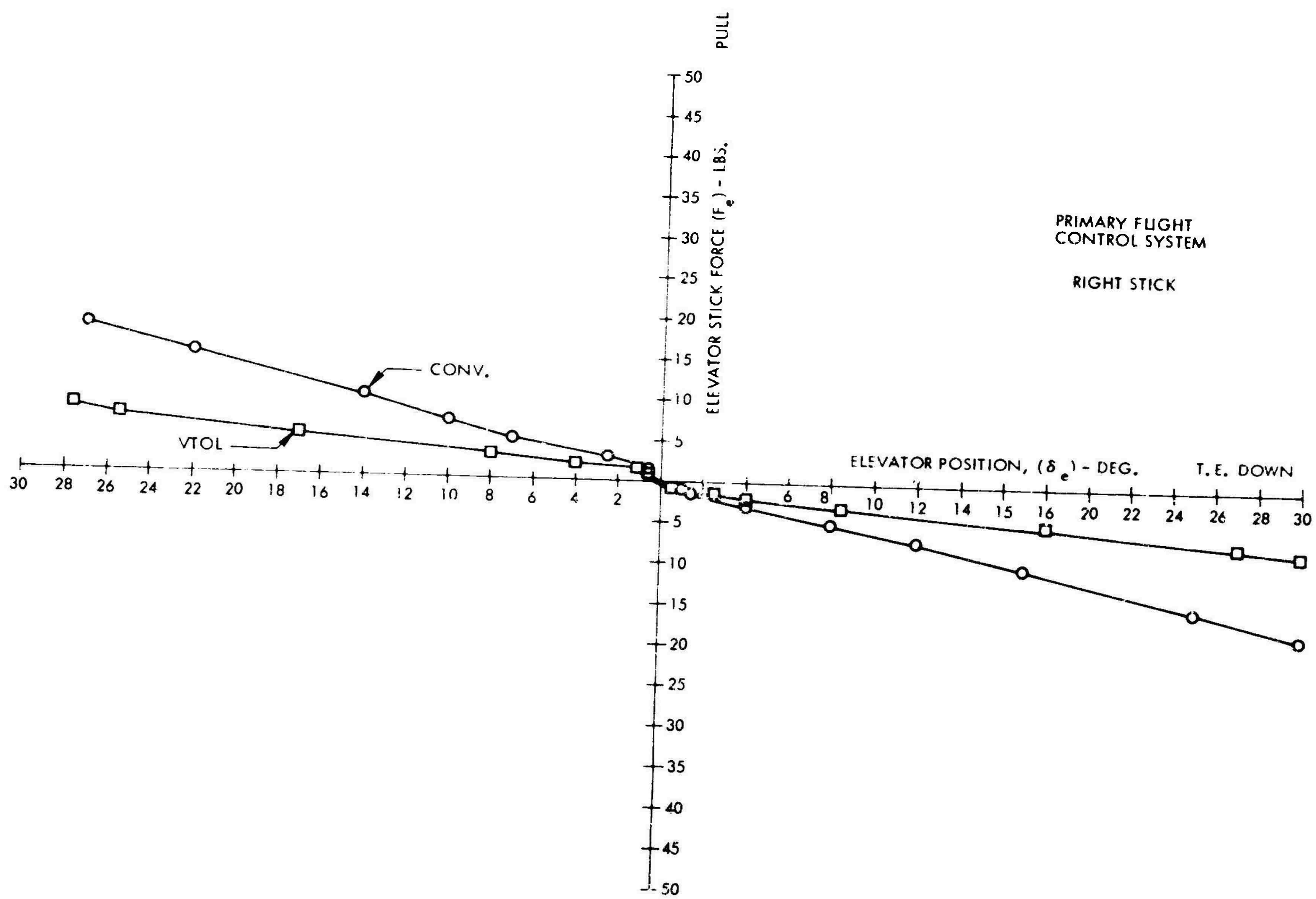


FIGURE 42 - ELEVATOR CONTROL SYSTEM CALIBRATION

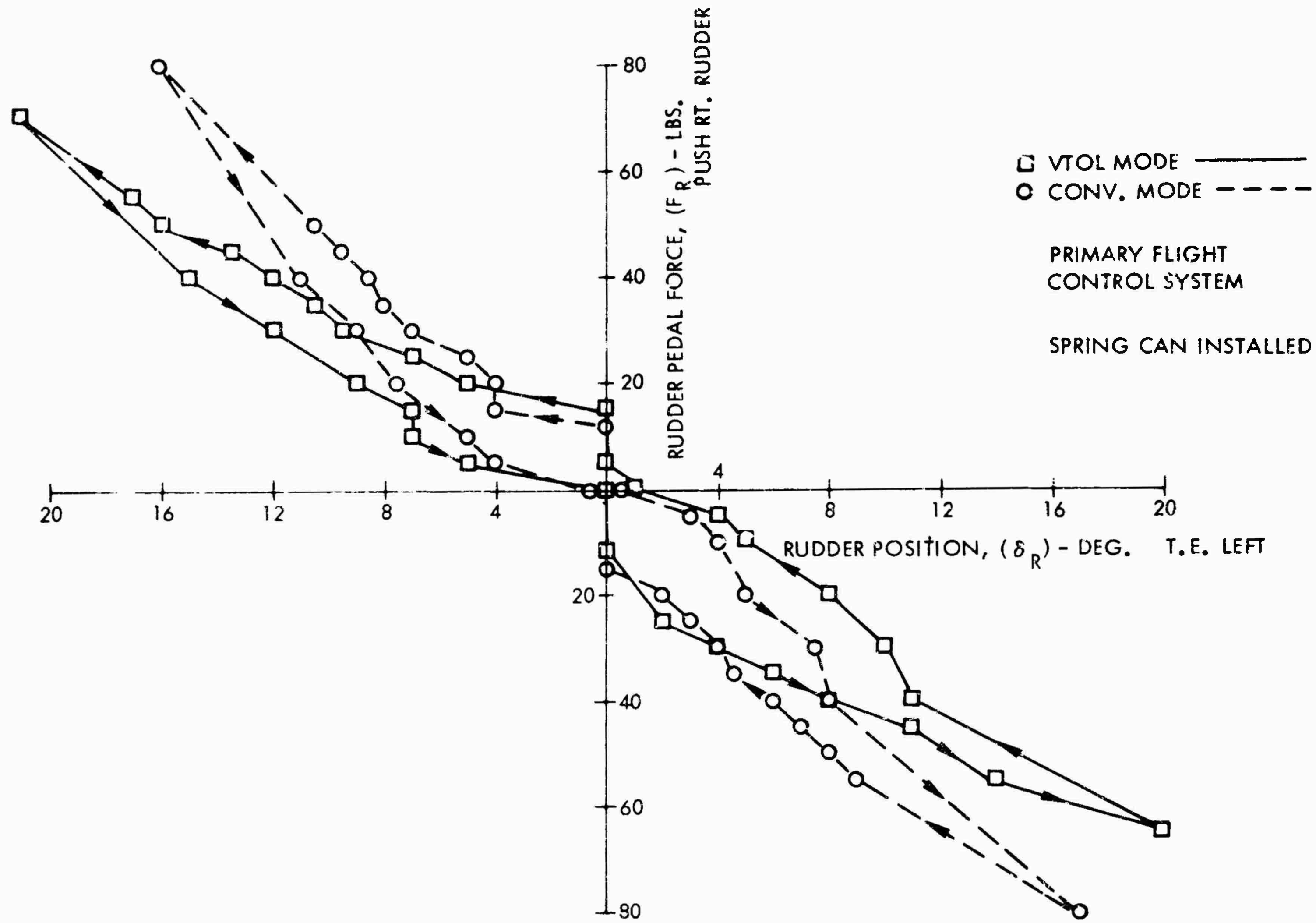


FIGURE 43 - RUDDER CONTROL SYSTEM CALIBRATION

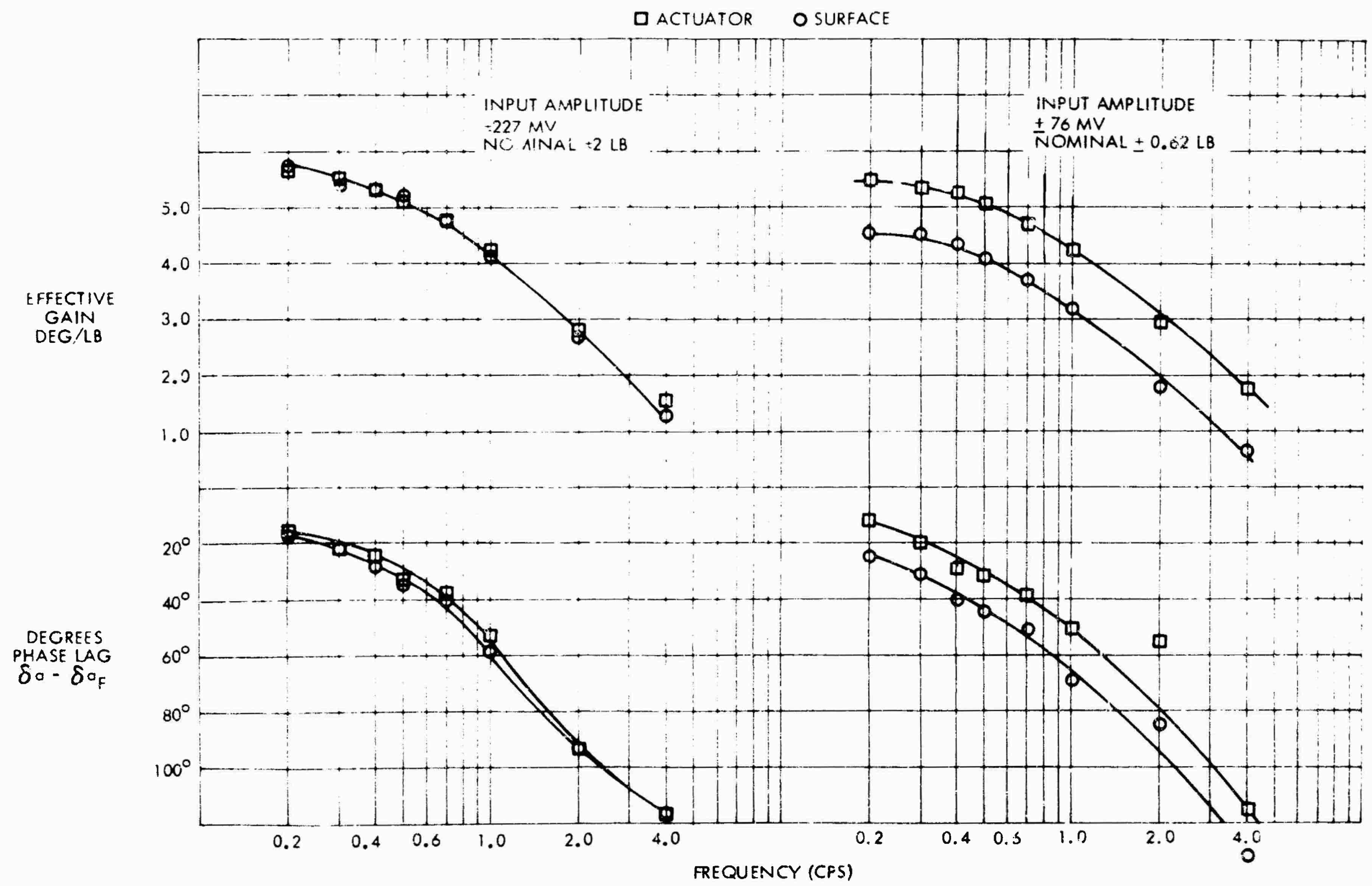


FIGURE 44 - XV-4B P.F.C.S. FREQUENCY RESPONSE - VTOL MODE

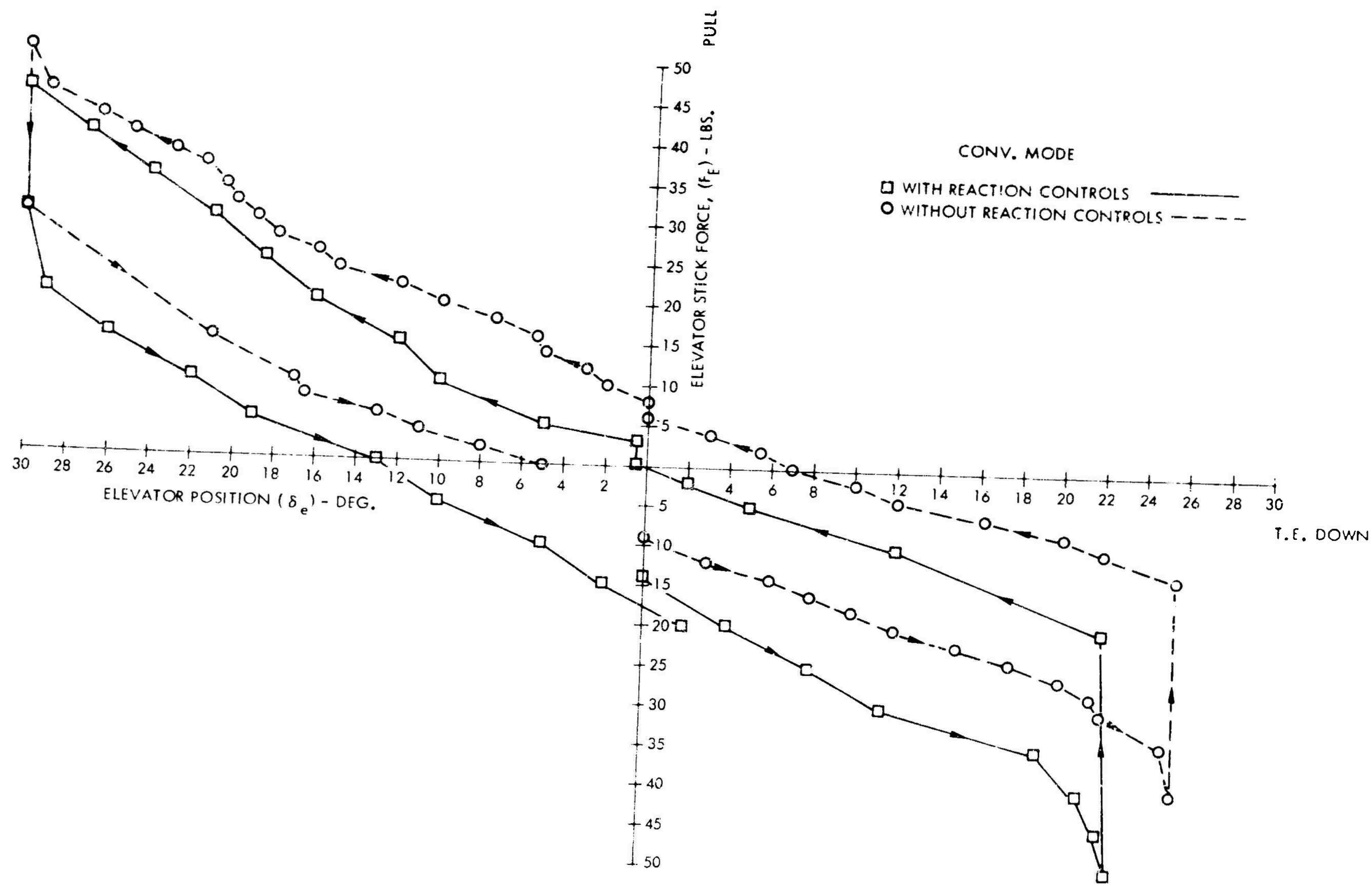


FIGURE 45 - ELEVATOR CONTROL SYSTEM CALIBRATION (MECHANICAL CONTROL SYSTEM)

(4) Environmental Testing--The temperature, vibration and sonic environment were monitored during an extensive series of engine run tests in both the conventional flight configuration and with the airplane mounted in the inverted telescope test rig under simulated hover conditions. This testing established a satisfactory engine installation and although modifications as described in Section III were found necessary to avoid localized overheating as a result of diverter valve leakage, the overall temperature environment was satisfactory. With the airplane close to the ground plane in the VTOL configuration, overheating of the wheels and tires and of the lower fuselage skins was encountered. A tire shield assembly, described in Section III, and insulating skin structure had been devised to obviate this problem but complete evaluation of these modifications had not been completed at the time of program termination.

The vibration and noise environment with all six engines operating was generally higher than predicted but this was not anticipated as a problem with proper protective apparel for the aircrew and with regular inspection of certain areas of structure. A more complete summary of this phase of testing is presented in paragraph 8, Section VII.

Engine re-ingestion was observed to a limited extent during the above testing and stalling of the lift-cruise engines was experienced on a few occasions. As discussed in Section III the lift/cruise engines were modified to improve stall margins and no further problems were experienced. Although re-ingestion remained a potential hazard during near ground operations, it was considered that with the use of special instrumentation for monitoring engine intake conditions, it would have been possible to develop procedures and techniques to avoid it.

(5) Thrust to Weight Ratio and Reaction Control Power--Quantitative measurements of thrust and reaction control moments were recorded with the airplane mounted in the inverted telescope test rig in the simulated hover configuration out of ground effect. Readouts from the integral balance system were used for calculations of thrust to weight ratio and reaction control power. These data are fully detailed in Reference 30 and were used to show compliance with the contract guarantees.

d. Flight Tests and Results

(1) Conventional Flight--Takeoff, climb, and handling performance generally agreed well with the predicted data presented in Reference 14. Testing in the conventional

configuration included longitudinal and lateral directional static and dynamic stability evaluation up to the maximum demonstration speed together with a limited assessment of longitudinal maneuvering stability and aileron response. Figures 46, 47, and 48 show typical lateral-directional static stability, and longitudinal static and maneuvering stability test results, respectively. Both Figure 47 and 48 show the anticipated low levels of force stability associated with the low stick force gradients selected for the conventional configuration. While these levels are lower than those of a conventional airplane they were considered satisfactory for the research mission of the XV-4B.

Although not taken completely into the stall, several low rate approaches were demonstrated up to predicted limiting alpha conditions in both the clean and landing configurations. No evidence of pitch up or wing drop was experienced and recovery was straightforward with nose down elevator and power increase. Light elevator buffet was noticeable at speeds below 1.05 V_s.

Cruise engine response was satisfactory throughout the flight envelope and an airstart was satisfactorily demonstrated. Oversensitivity in control together with some tendency for pilot induced oscillation necessitated increases in stick force gradients above the original nominal values. Rate feedback gains were also increased on the pitch and roll axes to reduce the airplane response. Control system performance was generally considered satisfactory and no inflight problems were encountered. Table IV presents a listing of control system parameters as they existed at the time of program termination.

Two flights were devoted to calibration of the airspeed and altitude displays. The results were typical of those expected of a boom system.

(2) VTOL Flight - Phase III--The Phase III flight envelope between wing stall speed and the flap limit speed of 240 KEAS was fully evaluated from a stability and handling viewpoint. Both lateral and longitudinal static stability compared well with predicted data. A low amplitude limit cycle in roll at 3 cps necessitated reduction of both the VTOL rate gain and the structural feedback filter previously discussed. Even with these changes it was found necessary to limit selection of VTOL control system gains to below 200 KIAS to prevent excessive aileron activity. No further improvement was attempted pending system evaluation under low speed and hover conditions.

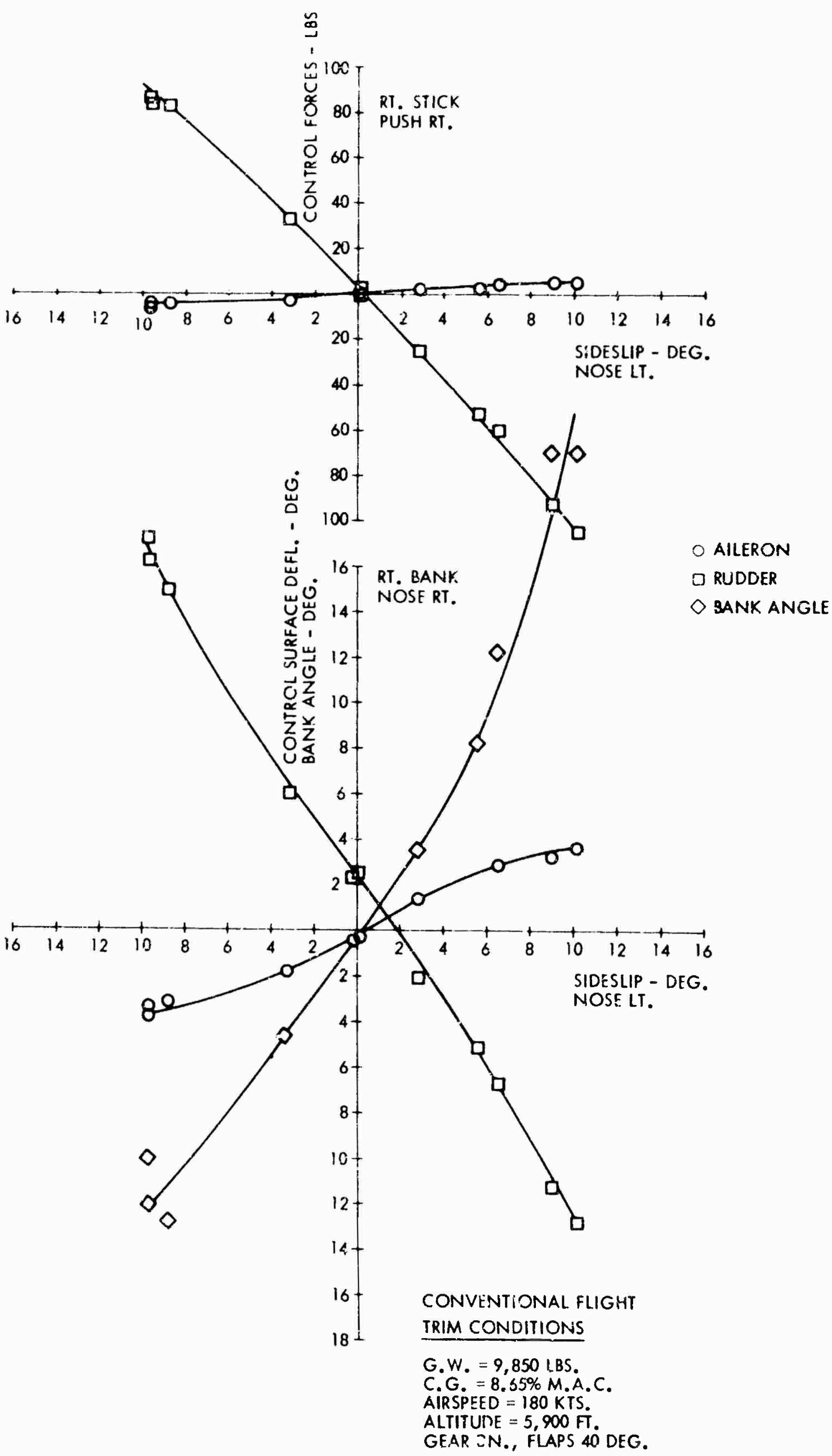


FIGURE 46 - STATIC LATERAL-DIRECTIONAL STABILITY

CONVENTIONAL FLIGHT

NOTES

1. PFCS
2. CONVENTIONAL GAINS
3. F_e DOES NOT INCLUDE 1 LB TRANSDUCER BREAKOUT FORCE
4. SOLID SYMBOL DENOTES TRIM POINT

CONFIGURATION

1. FLAP POS - 0 DEG.
2. GEAR POS - UP
3. GROSS WT. - 11,400 LBS.
4. CG - 7.90% MAC
5. TRIM ALT. - 4000 FT. AVG.

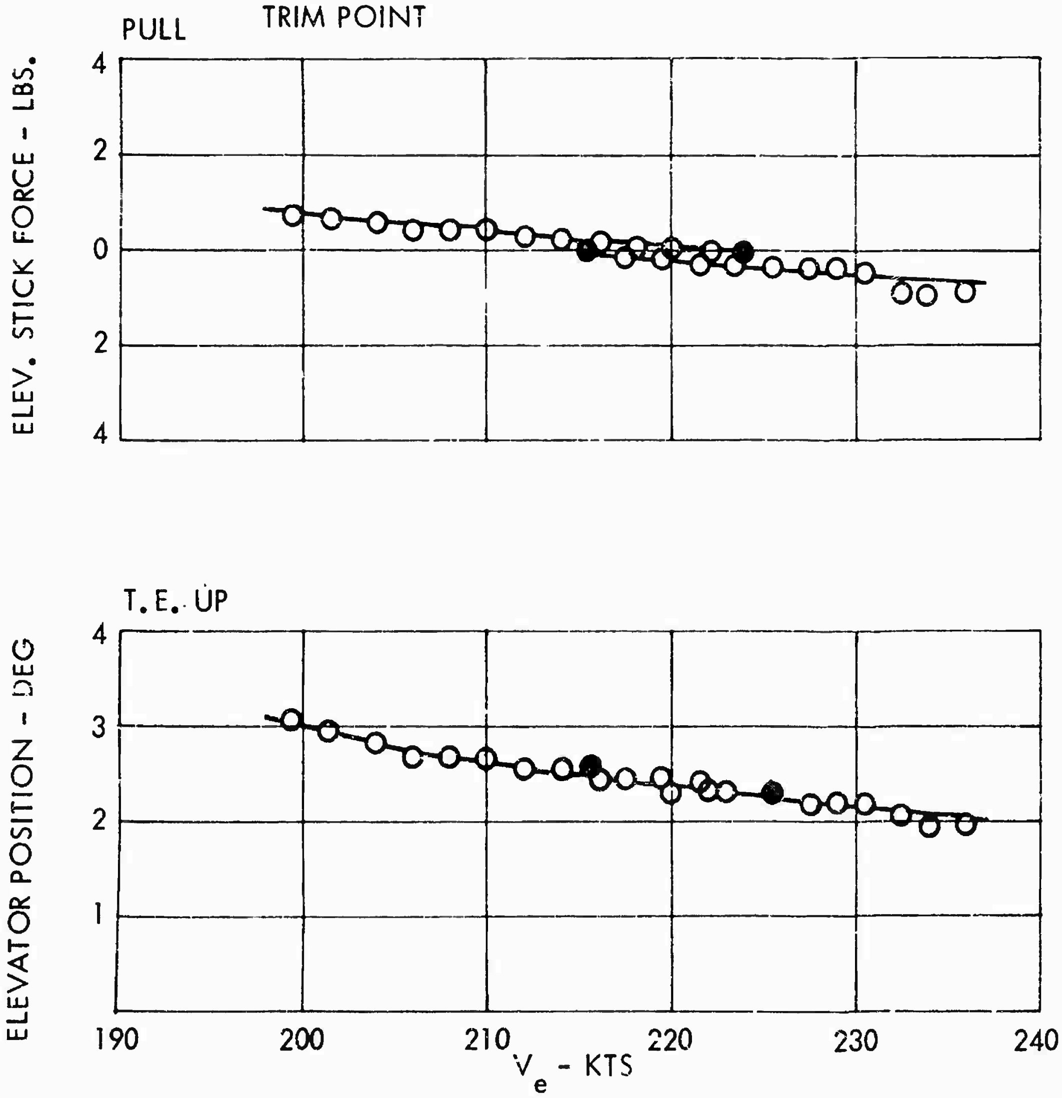


FIGURE 47 - STATIC LONGITUDINAL STABILITY

CONVENTIONAL FLIGHT

TRIM CONDITIONS

AIRSPEED (V_c)	-	247 KTS
ALTITUDE (H_{pc})	-	9650 FT
GROSS WEIGHT	-	10,990 LBS
C.G.	-	8.22% MAC
GEAR & FLAPS	-	UP

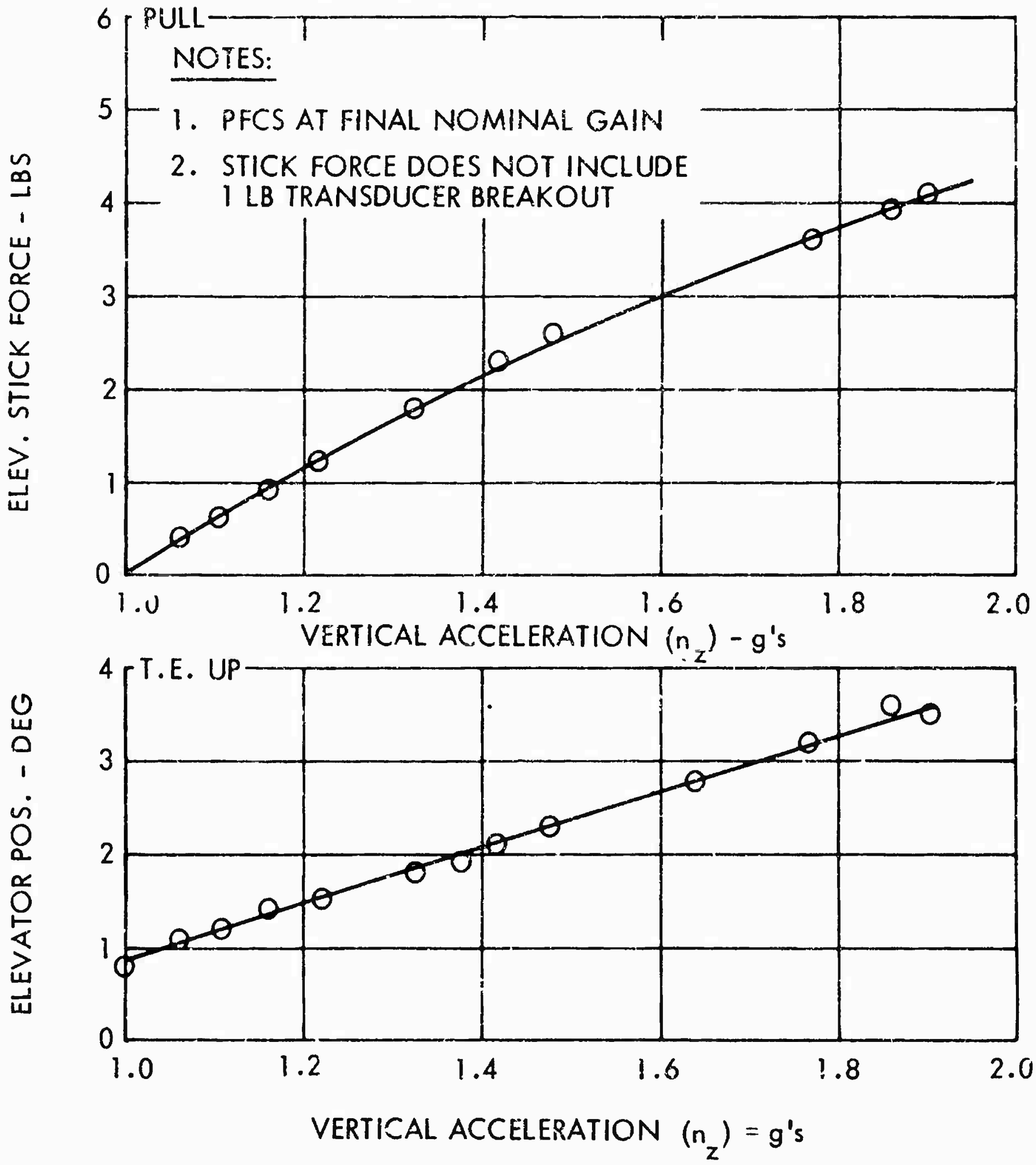


FIGURE 48 - ACCELERATED LONGITUDINAL STABILITY - TURNING FLIGHT

TABLE IV

P.F.C.S. NOMINAL GAINS AND AUTHORITIES

		ORIGINAL DESIGN		ACTUAL NOMINAL AT TIME OF PROGRAM TERMINATION	
		VTOL	CONV	VTOL	CONV
Pitch -	Breakout force - lb	1.0	1.0	1.0	1.0
	Force gain - Degs/lb.	4.3	1.58	2.66	1.25
	SAS Gain - Degs/Deg/ Sec	2.6	0.3	2.4	0.325
	Force lag-radians	2.5	2.5	5.0	5.0
	Rate lag - radians	-	-	30	30
	SAS authority - % Servo Stroke	70	70	70	70
	Trim authority - % Servo Stroke	50	50	50	50
Roll -	Breakout force - lb.	1.0	1.0	1.0	1.0
	Force gain - Degs/lb.	13.3	5.0	8.5	3.65
	SAS gain - Degs/Deg/ Sec	2.9	0.4	2.0	0.54
	Force lag - radians	2.5	2.5	5.0	5.0
	Rate lag - radians	-	-	30	30
	SAS authority - % Servo Stroke	70	70	70	70
	Trim authority - % Servo Stroke	50	50	35	35
Yaw -	Breakout force - lb.	5.0	5.0	15.0	15.0
	Force gain - Degs/lb.	1.33	0.57	0.31	0.2
	SAS gain - Degs/Deg/ Sec.	1.16	1.16	0.81	0.81
	Force lag - radians	2.5	2.5	5.0	5.0
	Rate lag - radians	-	-	30	30
	SAS authority - % Servo Stroke	70	70	50	50
	TRIM authority - % Servo Stroke	50	50	50	50

NOTE: (1) Roll gains are in terms of total aileron
(2) Yaw SAS has a 2-second canceller in Conv. mode

Lift engine airstarts were satisfactorily demonstrated within the speed range 160 to 220 KIAS.

(3) VTOL Flight - Phases I and II--The Phase II flight condition with all lift engines operating was examined on several occasions down to a minimum speed of 95 KIAS. Handling was generally considered satisfactory although changes in techniques were required as a result of the quasi attitude hold characteristics of the high rate gain control system and the very low stick forces. A very stable platform was apparent down to the minimum speed tested although speed/altitude control was not very tight and fairly large control displacements were required for maneuvering. This was particularly true at low speeds with low manifold bleed pressures. Further increase in stick force gradients were necessary to alleviate P.I.O. tendencies induced apparently unconsciously as a result of the inertia forces from the pilots grip.

Diverting the lift/cruise engines down at 140 KIAS to transfer to the Phase I configuration as shown by the time history of Figure 49 was easily accomplished with no significant airplane transients. An improvement in control response was apparent as a result of the higher engine RPM required to trim and the resultingly higher compressor bleed. Again a very stable platform was reported with no handling problems apparent. No quantitative static stability data was recorded in the Phase I configuration and only isolated points were completed in Phase II. No potential problem areas were revealed, however, within the speed range covered.

An empennage buffet phenomenon was apparent in both Phase II and Phase I configurations in the 120 - 140 knot speed range. This buffet (at 10 cps) was of fairly low amplitude and was only apparent at swivel nozzle positions forward of nominal. It was being quantitatively examined at the time of program termination.

Additional immediate program objectives at this time were continued slow down in the Phase I configuration and concurrent development of V/STOL take-off and landing procedures.

(4) Significance of Tests--The portion of the test program completed had indicated that despite several developmental problems, the airplane should have met design objectives and satisfied all specification requirements.

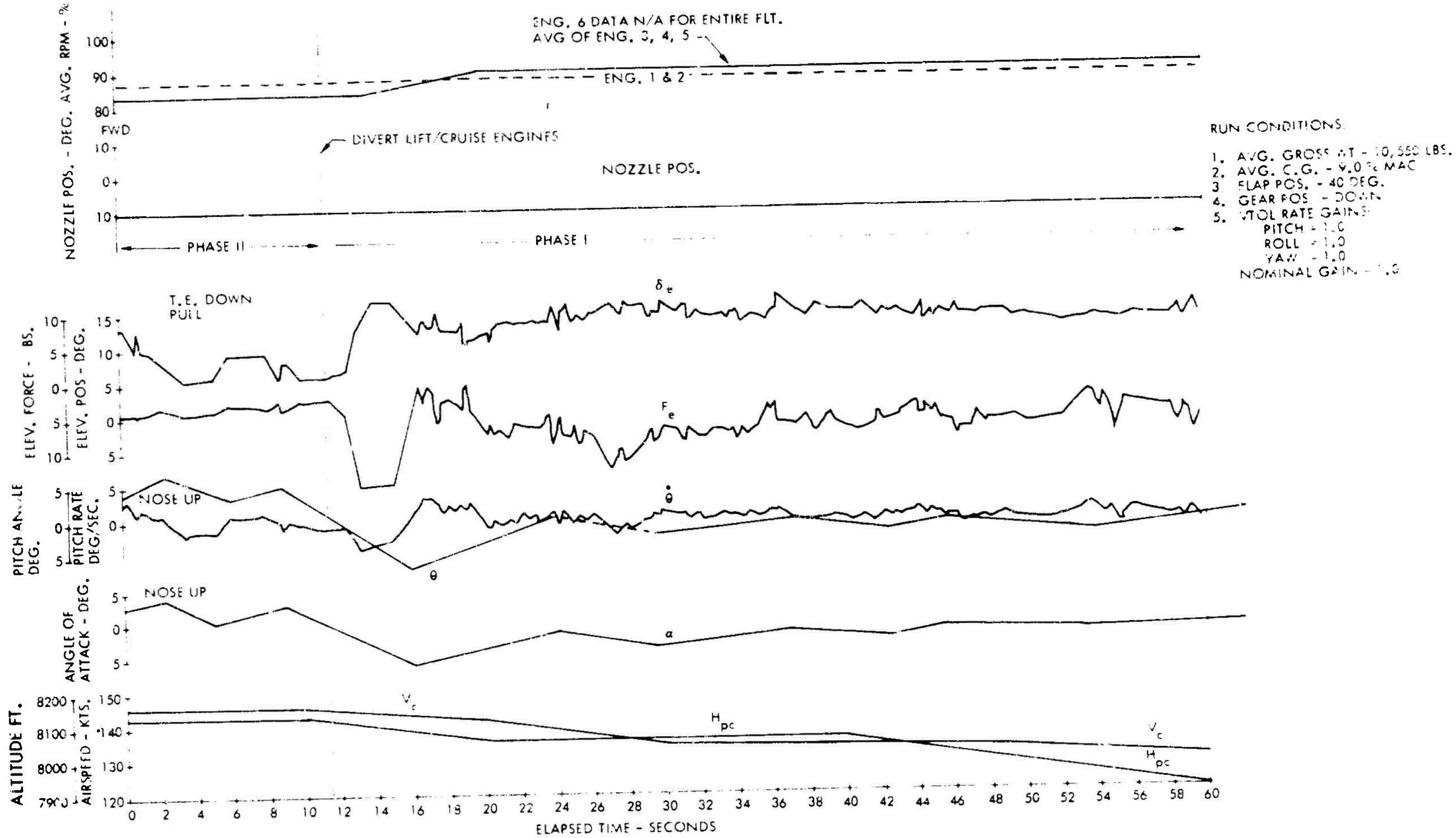


FIGURE 49 - PHASE II/PHASE I SLOWDOWN

The "fly-by-wire" control system had been developed to the point where satisfactory stability and handling qualities were achieved throughout the envelope tested and design modifications, though not fully evaluated, were in hand to resolve the developmental problems.

Engine reingestion remained a potential hazard, but it was considered possible to develop procedures and techniques to avoid it.

SECTION VIII

CONCLUSIONS

The significant conclusions derived from the work accomplished within this program and from the data presented in this and the referenced reports are presented below. Since the program was not completed, a number of conclusions must be considered tentative. In addition, it should be recognized that some of the negative conclusions, though valid, are a direct result of configuration constraints.

1. The portion of the flight test program completed indicated that despite several development problems, the airplane should have met its design objectives and specification requirements.

2. Test data obtained in the flight test program showed good agreement with aerodynamic predictions and no stability problems were revealed.

3. The "fly-by-wire" control system was developed to the stage where satisfactory stability and handling qualities were achieved throughout the envelope tested. For a development system undergoing constant modification, reliability was satisfactory and no known in-flight failures were experienced.

4. Environmental testing in the conventional and hover modes established a satisfactory engine installation and indicated no initial structural fatigue conditions.

5. Noise environment with all six engines operating was higher than predicted but was not expected to be a problem with proper protective apparel for the crew and with regular inspection of certain areas of structure.

6. Simple bell-mouth inlets provide acceptable levels of pressure distortion and pressure loss in a speed range from 0 to 240 knots.

7. The development of the Escape System resulted in a system that provided satisfactory single crew member escape under a broad set of operating conditions and test results indicated that an acceptable confidence level had been established for the escape of two crew members.

8. The XV-4B reaction control valves demonstrated thrust performance close to that predicted but the mechanical performance, particularly as related to actuating torque and leakage, was less than desired.

9. The YJ85-19 engines performed satisfactorily under varying and high bleed flow rate conditions. Inspections revealed no adverse effects of these bleed flow conditions on the engine hot section components.

10. High efficiency axial flow engine compressors have adequate stall margin for conventional applications but these margins are only marginally acceptable in VTOL installations because of the lack of tolerance to rapidly varying and distorted compressor inlet temperatures. For the XV-4B, engine stalls due to hot gas ingestion remained a potential hazard but it was considered that stalls could be avoided by the development of operational techniques based on careful monitoring of inlet temperatures.

11. The lack of tires capable of tolerating temperatures of the order of 600°F for usefully long periods of time proved to be a program constraint and will be a configuration constraint in future designs. For the XV-4B, limited testing indicated that a specifically developed wheel and tire shield, together with operational techniques, would probably be successful in alleviating tire overheating problems.

12. Because of tire overheating problems and the hazards of possible engine stalls true XV-4B VTOL operation at ground level would most probably require a VTOL operating pad to conduct hot gases away from the tires and engine inlets.

13. Bleed air and exhaust gas ducting systems in VTOL aircraft must have prime reliability. Even if rational design criteria are applied first order reliability must be demonstrated through qualification testing under operating environmental conditions.

14. Maintenance manhours for the XV-4B were large because of the high density and compactness of systems installations and because of limited accessibility to many areas of the airplane.

15. Maintaining a high degree of cleanliness in the hydraulic system of a developmental airplane will always be a problem. Consideration should be given to the development and use of components having less stringent cleanliness requirements. Cleanliness could also be improved by providing self sealing quick disconnect couplings at major components.

16. Handling diverter valve leakage will be a continuing problem on aircraft utilizing diverter valves.

17. An adequate third source of electric and hydraulic power should have been provided to have been consistent with the triply redundant philosophy employed in

the electronic and sensor portions of the flight control system.

18. Normal aircraft wiring practices are not adequate for interconnecting and interfacing components of "fly-by-wire" systems.

19. The state-of-the-art of the mechanical aspects of electrical and electronic systems (ships wiring, terminals, plugs, connectors, etc.) is far behind that of electronic circuitry design and packaging.

20. Redundant sensors and electronic circuitry for a given function should be physically separated to prevent a single local mechanical failure or environment change affecting all sensors and circuits in the same manner.

21. Environmental qualification of critical system components should be well in excess of the normal design environmental requirements.

22. The mechanical back-up flight control system was marginally acceptable for conventional flight operations and was unacceptable for VTOL operations below approximately 100 knots.

23. Because of high friction forces the rudder pedal system of the airplane was only marginally acceptable.

24. The lift engine collective throttle system that provides only limited authority for individual engine control is considered to be only marginally acceptable. Though not evaluated, the ability to control a single engine malfunction, such as a stall, without affecting the operation of the other three engines is questionable.

25. The functional characteristics of the 80 percent RPM throttle bleed gates were less than desirable. In conventional flight the configuration of the gate interfered with the smooth manipulation of the throttles particularly in landing approaches where power settings less than 80 percent RPM were frequently desired. Though not evaluated, it was also considered that the gates would be a constraint on split throttle operation in the VTOL flight regime.

26. Laterally canted landing gear should be avoided in future designs because of eccentric loading conditions.

27. Separate, large scale, α and β displays should be provided rather than smaller ones incorporated in the ADI.

28. Vertical tape instruments should be used for all engines rather than tapes for the lift engines and dial instruments for the lift/cruise engines.

29. The airplane VHF communications capability was of little value in this program.

APPENDIX

FATIGUE DAMAGE AND INSPECTION PLAN

1. INTRODUCTION

During the course of operational testing some occurrences of fatigue damage were observed and other potential problem areas were identified. Results of these occurrences are summarized in Table V. Inspection procedures developed from the results of operational experiences are presented in Tables VI, VII, VIII and IX. Individual types of structure and equipment are briefly discussed in the following paragraphs.

a. Ducting Systems

Comprehensive design and test programs were accomplished on the complete ducting system. Testing was conducted in both the cyclic test rig and in the aircraft, as described in References 27 and 28. As a result of these programs, procedures were written to cover inspection of the ducting system during the life of the aircraft. These are included in summary form in Table VI for reference.

b. Exhaust Systems

The engine exhaust components such as the diverter valves, tail pipes, and vector nozzles are prone to fatigue failure due to the intense acoustic and temperature environments. Inspection procedures for these components are covered in Table VII.

c. Fuselage Structure

The lower fuselage aluminum skins forward and aft of the lift engine exhaust bay were determined to be marginal in fatigue life during the test program described in Reference 28. The lift engine exit doors were also isolated as a potential problem area due to the high temperature, buffet and acoustic levels present during hover or transition. These areas shall therefore be inspected in accordance with the requirements of Table VII.

d. Empennage and Aft Fuselage

The empennage and aft fuselage tail cone were determined to be particularly responsive to acoustic excitation as evidenced by the relatively severe vibration levels reported in Reference 28. The increase in measured levels over the predicted values were attributed to a more flexible structure than anticipated in the analysis, resulting in superposition of stabilizer vibration and local structural response. Since the XV-4B empennage was modified from the XV-4A configuration (beefed-up to correct the rib sonic fatigue cracks incurred during the XV-4A tests), close attention was given this area during inspections. For these reasons, the inspection requirements of Table VII include a complete X-ray and dye penetrant inspection prior to delivery, as well as every 50 hours thereafter.

e. Equipment Components

Equipment components used in the XV-4B were predominantly off-the-shelf items due to the limited scope of the program. This resulted in the installation of many components which did not meet the original vibration qualification criteria delineated in Reference 26. Measured data obtained during the vibration test program (see Reference 28) were used to define the final qualification status of each major equipment component. The final qualification status of each item is presented in Reference 11. All components were found to be fully qualified from a performance or functional standpoint; however, the fatigue life of many components may be marginal. Therefore, the equipment items shown in Table VIII should be inspected at the indicated intervals to detect possible fatigue problems.

The preflight inspections which were made during the flight test program are summarized in Table IX. A similar set of inspections was conducted before each flight during the controls testing phase.

TABLE V

SUMMARY OF XV-4B No. 102 FATIGUE DAMAGE

ITEM	PART NO.	DAMAGE DESCRIPTION	CORRECTIVE ACTION
Diverter Valve Casing	116020-5/-6	# 1 and # 2 engine diverter valve casings developed 1 1/4 inch longitudinal cracks, one each, at 5 o'clock & 7 o'clock respectively as viewed from rear in .025 casing.	Removed cracked area and spot welded in patch. Added external stiffener to prevent damage during instl.
		2 inch crack in depression on casing approx. 6 inches aft of fwd. engine mate flange.	Straightened depression and welded crack.
		Several cracks around upper and lower mount areas where stiffeners tie to pan and mount fittings. Due to lack of preheating and difference in material thickness.	Ground out cracks and welded.
Diverter Valve Doors	116011 & 116012	Cracks at each end of support tube in cover skins. Due to differential expansion of heavy sections and light skins.	Several modifications made. Final design incorporated slip joint in skin.
Diverter Duct	116027	Expansion bellows cracked on several occasions due to binding, improper alignment, and sonic fatigue.	Improved support and modified bellows design to utilize double wall.
Swivel Nozzles	116024 & 116025	Cracks developed in exit lip and band.	Repaired cracks & defective band joint by removing damage area and rewelding.

TABLE V (Cont'd)

ITEM	PART NO.	DAMAGE DESCRIPTION	CORRECTIVE ACTION
Swivel Nozzles (Cont'd)		Lift and cruise engine seals cracked with several fingers missing.	Redesign seals to provide more seal pressure and improved slit relief radius.
Bleed Air Duct Fus. Mid. Sect.	116004-5	Duct ruptured at intersection of collector ducts due to faulty weld, improper intersection, and inadequate support between collector ducts.	Modified design by enlarging main duct diameter locally, added gussets between collector ducts, increased wall thickness, & subcontracted mfg.
Bleed Air Duct Engine Collector	116035-5	Duct ruptured due to severe joint mismatch in duct wall sections causing high stress concentrations.	Redesigned & subcontracted mfg. to improve wall intersection joints.
Lift Engine Exit Doors	113066	(1) Cracks in inner skin around gusset areas. (2) Also loose rivets.	(1) Removed free edge of gusset which was being excited by exhaust blast. (2) Replaced loose rivets.
		1 1/4 inch crack in inner skin along 4th row of rivets from aft end due to free edge gusset resonance.	Stop drilled and added doubler.
Lift Engine Inlet Vane	116099 & 113065	Support strut separated from fiberglas inlet fairing.	Install new clip with adhesive & rivets.

TABLE V (Cont'd)

ITEM	PART NO.	DAMAGE DESCRIPTION	CORRECTIVE ACTION
Nose Cowling	113006-3	Small cracks around 11 dimples from .10 to .20 inches long.	Stop drill & fill dimples with joggle comp. and add .040 strap across area.
MLG Wheel Well Fairing	113122 & 113149	(1) 1/2 inch crack in skin. (2) Also former at fwd. outboard corner cracked 3.5 inches.	(1) Stop drilled and doubler added. (2) smooth cut crack and add angle reinforcement.
Flap Cove Top Skin	112015	Loose and pulled rivets	Replace loose rivets and add extra rivets as required.
Flap Cover Lower Skin	112015	2 small cracks in skin at rib location. 1/2 inch max. length. LH side. May be results of instl. damage.	OK to fly. Observe for propagation.
Inboard Wing Leading Edge	112003	Cracks around screw holes. Some pulling through. Damage is from frequent screw removal.	Add countersunk washers under screws.
Wing Root Fairing	112030	(1) Leading edge cracked 1 1/4 inch along weld joint. Due to deficient weld. (2) Also several .25 to .45 cracks.	(1) Continuity not required. OK for flight. (2) Weld and smooth.
Aft Fus. Skin	113004	2 inch crack in LH side skin 8 inches fwd of strake. Also 1/2 inch crack in LH side skin below and fwd of strake and parallel to vert. rivet line due to cruise eng. noise.	Stop drilled and double added at both locations.

TABLE V (Cont'd)

ITEM	PART NO.	DAMAGE DESCRIPTION	CORRECTIVE ACTION
Fuel Tank Upper Contour Covers	113031 & 113051	Panel stiffeners cracked due to mechanics using area as walk way.	Stiffeners replaced with beefed-up configuration.
Strake	113128	(1) Inbd fairing cracked. (2) Several loose rivets.	(1) Trim and smooth. (2) Replace loose rivets.
Rudder	114009	Approx. 7 loose rivets between T.E. skin and spar.	Inspect frequently. Looseness does not appreciably effect strength.
Rudder Bal. Wt.	114009	4 loose fasteners in RH wt.	Replace loose fasteners.
Elev. L.E. Skin	114005	Loose rivets common to inbd hinge rib.	Replace loose rivets except 1 under bal. wt. acceptable.
Bullet Fairing	114006	RH side of bullet below horiz. stab. cracked 1 1/2 inches long due to faulty weld.	Remove and weld.
Nacelle Frame	113121-1	1 1/4 inch crack in LH Nacelle Frame Flange in wheel well area due to local damage during testing which developed into a fatigue crack.	Stop drilled & trimmed. Spliced with similar frame segment.

TABLE VI
AIRCRAFT INSPECTION MEMO (AIM)
SUMMARY

ITEM INSPECTED	PROCEDURE	FREQUENCY
AIM #1 Diverter Valve Housing 116010-1, 116010-2	1. Gain access to valve interior. 2. Inspect for wrinkles in interior. 3. Inspect same area for evidence of fatigue cracks, etc. 4. If crack exists repair.	8-12 Operational hours
AIM #4 Lift Elbow 11602701, -2 Cruise Tailpipe 116025-1, -2, -3, -4 Vector Nozzles 116024-65 Lift Tailpipe 116026-3, -4, -5, -6	1. Gain access to all assemblies. 2. Inspect all assemblies for cracks in welds and skins. Seals, bearings, and supports shall be inspected for condition. 3. All discrepancies found shall be dispositioned by M.R.B. action.	50 Operational hours.
AIM #5 Pitch Valve Fwd 116021-1 Roll Valve 116022-1, -2 Pitch & Yaw Valve Aft 116023-1 B_{LO} Bleed Ducts 116004-3, -311, -7 Cruise Engine Bleed Ducts 116035-5, -7, -9, -10, -99, -101	1. Gain access to all assemblies. 2. Inspect all assemblies for cracks in welds and skins. Seals, bearings, actuating arms, push rods, shear pins, and supports shall be inspected for condition. 3. All discrepancies found shall be dispositioned by M.R.B. action.	25 Pressurized hours with engine bleed air

TABLE VI (Cont'd)

ITEM INSPECTED	PROCEDURE	FREQUENCY
AIM # 6 Lift Engine Bleed Ducts 116005-9, -10, -13, -14	1. Gain access to all assemblies. 2. Inspect all assemblies for cracks in welds and skins. Supports shall be inspected for condition. 3. All discrepancies found shall be dispositioned by M.R.B. action.	50 pressurized hours with engine bleed air
AIM # 7 (In process of release at time of accident) Wing Roll Ducts 116030-83, -84, -85, & -87	1. Gain access to all assemblies. 2. Inspect as required in AIM # 5.	25 pressurized hours with engine bleed air.

TABLE VII

STRUCTURAL INSPECTION REQUIREMENTS

DESCRIPTION	PART NO.	TYPE OF INSPECTION (1)	INSPECTION INTERVALS
A. DUCTING SYSTEMS			
1. Center Fuselage Main Bleed Air Duct	116004-311	Visual	25 Hours (2)
2. Lift Engine Bleed Air Manifold Ducts	116005-9, -10 -13, -14	Visual (3)	50 Hours (2), (3)
3. Cruise Engine Bleed Air Manifold Ducts	116035-5, -99	Visual (3)	25 Hours (2), (3)
4. Wing Mounted Roll Ducts	116030-83, -84, -85, -87	Visual	25 Hours (2)
5. Fwd Fuselage Main Bleed Air Duct	116004-7	Visual	25 Hours (2)
6. Aft Fuselage Main Bleed Air Duct	116004-3	Visual	25 Hours (2)
7. Wing to Center Fuselage Interconnect Ducts	116035-7, -9, -10, -101	Visual	25 Hours (2)
8. Wing Duct Transition Section	116030-3	Visual	25 Hours (2)
9. Air Conditioner Bleed Air Ducts	116114-1	Visual	50 Hours (2)
10. Pitch-Yaw Valve Including Shutters & Supports	116023-1	Visual	25 Hours (2)
11. Pitch Valve Including Shutters & Supports	116021-1	Visual	25 Hours (2)
12. Roll Valves Including Shutters & Supports	116022-1, -2	Visual	25 Hours (2)
B. EXHAUST SYSTEMS			
13. Diverter Valve Housing & Doors	116020-1, -2	Visual	10 Hours (4)
14. Lift Elbow Assemblies	116027-1, -2	Visual	50 Hours (4)

TABLE VII (Cont'd)

DESCRIPTION	PART NO.	TYPE OF INSPECTION (1)	INSPECTION INTERVALS
15. Vector Nozzles & Pivot Fittings	116025-65	Visual	50 Hours (4)
16. Cruise Engine Tail Pipes & Support Fittings	116025-1, -2, -3, -4	Visual	50 Hours (4)
17. Lift Engine Tail Pipes & Support Fittings	116026-3, -4, -5, -6	Visual	50 Hours (4)
C. FUSELAGE STRUCTURE			
18. Fwd and Aft Lwr Fuselage Skins	113033-117, -119, -121, -122, -65, -91	Visual	25 Hours (4)
19. Fwd and Aft Lwr Fuselage Skins	113053-17, -78, -79 -80, -85, -93	Visual	25 Hours
20. Lift Engine Exit Doors	113066-3, -4	Visual	25 Hours
D. EMPENNAGE & AFT FUSELAGE			
21. Horizontal Stabilizer	114002-1, -2	Visual, X-Ray	Before Delivery; 50 Hours
22. Vertical Stabilizer	HB4004	Visual, X-Ray	Before Delivery; 50 Hours
23. Aft Fuselage	113004-1	Visual	50 Hours

NOTES:

(1) Dye Penetrant inspect all questionable areas.

(2) Accumulation of time for bleed air ducts is that time during which they are pressurized.

(3) A complete dye penetrant inspection should be performed on all engine bleed air ducts each time an engine is removed.

(4) Accumulation of time for engine exhausts and associated hardware is engine run time for each engine.

TABLE VIII

EQUIPMENT INSPECTION REQUIREMENTS

The following equipment components shall be visually inspected at 50 hour intervals for evidence of wear, fatigue or other damage which could affect performance of the item. Dye penetrant, X-Ray, disassembly, or other means of inspection shall be used on all questionable areas. Functional test shall also be conducted on those items so marked at the indicated intervals to verify continued satisfactory performance. These functional tests shall be in accordance with and meet the requirements of the applicable specification or functional test document.

DESCRIPTION	PART NO.	MANUFACTURER	SUB-SYSTEM
Inverter	MGE 23-400	Leland Airborne Products	AC Generation
Engine Driven Pump	PV3-0222	Vickers	Hydraulic Power Generation
Hydraulic Reservoir	7111	Pneudraulics Inc.	Hydraulic Power Generation
Pressure Snubber	3H90003-101	Operating & Maintenance Specialities	Hydraulic Power Generation
Relief Valve	1703-14	Droitcour Co.	Hydraulic Power Generation
Hydraulic Pressure Shut-off Valve	JH 1026-1	Aircraft Products Co.	Hydraulic Power Generation
Hydraulic Suction Shut-off Valve	JH 1010-1	Whittaker	Hydraulic Power Generation
Check Valve	112-589976	Parker Aircraft Co.	Hydraulic Power Generation
Check Valve	112-589977	Parker Aircraft Co.	Hydraulic Power Generation
Rudder Pedal Force Transducer	231E921P1	General Electric	Primary Flight Control
Gain Change Box	929C844G1	General Electric	Primary Flight Control
Shut-off Valve	853DG29	Marotta Valve Corp.	Primary Flight Control
Hydraulic Cylinder	OMP 3507-1	Ozone Metal Products	Primary Flight Control
Position Transmitter	8TJ39AB02	GE Instrument Div.	Primary Flight Control

TABLE VIII (Cont'd)

DESCRIPTION	PART NO.	MANUFACTURER	SUB-SYSTEM
Position Transmitter	8TJ39ABC2	GE Instrument Div.	Primary Flight Control
Cable Tension Regulator	0501156-1	Pacific Scientific Co.	Primary Flight Control
Hydraulic Cylinder	OMP 3504-1	Ozone Metal Products	Flap
Check Valve	112-589976	Parker Aircraft Co.	Flap
Accumulator	MS 28700-1	Parker Aircraft Co.	Flap
Position Transmitter	8TJ39ABC2	GE Instrument Div.	Flap
Hydraulic Servo	68200-301	Bertea Corp.	Vector Nozzle
Hydraulic Cylinder	OMP 3508-1	Ozone Metal Products	Lift Engine Exit Door
Hydraulic Lock Valve	HP 31100-8	Hydra-Power	Lift Engine Exit Door
Hydraulic Actuator	8-8000-1	Prosser Industries, Inc.	Propulsion/Exhaust
Spin & Drogue Chute System	SK 6690-0040-1	Stencel Aero Engineering Co.	Drogue Chute
Auto Ignition Actuator	10-382390-1	Bendix Corp.	Collective Throttle
Valve	121150	Rovalco	Bleed Air
Valve	121160	Rovalco	Bleed Air
Valve	35870	Sterer	Fuel
Linear Actuator	EDL1020M154	Nash Controls Inc.	Fire Detection
Linear Actuator	EDL1020M154-1	Nash Controls Inc.	Fire Detection
Keyer	695537-1	Seaboard Electric Corp.	Fire Detection
Stall Warning Box[1]	II 8065-1	Lockheed	Stall Warning
Controller-Warning Lights	R7954	Radar Relay	Instrument Panel
Transceiver & Power Supply	522-2593-011	Collins Radio	VHF Communication
VHF Communication Antenna	ARC 35180-0300	Aircraft Radio Corp.	VHF Communication
Slaving Accessory[1]	522-2644-011	Collins Radio	Compass
Directional Gyro[1]	522-3241-000	Collins Radio	Compass
Attitude Director Indicator[1]	4058T	Lear Siegler Instruments	Attitude Director Indicator
Vertical Gyro[1]	14168-1A	Bendix	Attitude Director Indicator

TABLE VIII (Cont'd)

DESCRIPTION	PART NO.	MANUFACTURER	SUB-SYSTEM
Pressure Regulator & Bleed Air Shut-off Valve	392678-2-1	AiResearch	Air Conditioning
Flow Control Valve	106018-2	AiResearch	Air Conditioning
Aft Compartment Blower	M4941C-1B	Dynamic Air Engineering	Air Conditioning
Pressure Reducer & Cylinder Assy	F5038350-17	Aro Firewel Corp.	Oxygen
Selector Valve	893933	Walter Kidde Co.	Landing Gear
Check Valve	112-589976	Parker Aircraft Co.	Landing Gear
Shuttle Valve	3H90059-103	Pneudraulic Inc.	Landing Gear
MLG Uplock Actuator	JH 1035-1	Carl Drescher Co.	Landing Gear
Flow Regulator	37030	Sterer	Landing Gear
Flow Regulator	37040	Sterer	Landing Gear
Accumulator	2670206	Parker Aircraft Co.	Landing Gear
Controllable Check Valve	JH 1007-2	Allen Aircraft	Landing Gear
Brake Valve	HP891100-61	Hydra-Power	Landing Gear
Restrictor	2R14410-6.6-5	Gar Kenyon	Landing Gear
Nose Gear Accumulator	3H90048-107	Arkwin Industries, Inc.	Landing Gear
Control Box Assy	1176L025-3	Loud Products	Landing Gear
Feed Back Pot Assy	1925L350	Loud Products	Landing Gear
Command Pot Assy	1176L099	Loud Products	Landing Gear
Steering Control Valve	1176L001	Loud Products	Landing Gear
Pressure Brake Reducer	411590-13	Bendix	Landing Gear

NOTE:

(1) Functional Test required at each 50 hours interval.

TABLE IX

SUMMARY OF PREFLIGHT INSPECTIONS
(FATIGUE DAMAGE)

AREA OF INSPECTION	ITEMS TO INSPECT	INSPECT FOR
Nose Compartment Top	Elec. System, Wire Bundles, Plugs, Relays, Transducers and Control Actuators.	Condition & Security
	Control Cables, Pulleys, Bellcranks and Support Brackets.	Condition & Security
	Hydraulic Lines.	Condition, Security and Leaks.
	Air Speed Boom & Boom Structure.	Condition & Security
Nose Compartment Lower	Pitch Control Valves, Bleed Air Ducts, Control Arms and Links.	Condition, Cracks & Security
Aft Fuselage Interior	Electric Control Panels, C/B Panel, Inverters, Relays, Wire Bundles, Plugs, Voltage Regulators, Battery, Battery Box & Cooling Fan.	Condition, Security and Operation.
	Instrumentation.	Condition & Security
	S.A.S. Equipment	Condition & Security
	Hydraulic Lines, Vent and Pressure Tubing	Condition, Security and Leaks.
	Control Cables, Pulleys and Brackets.	Condition, Security & Safety

TABLE IX (Cont'd)

AREA OF INSPECTION	ITEMS TO INSPECT	INSPECT FOR
	Bleed Air Ducts	Condition, Security, Leaks and Overheat.
Aft Fuselage Interior	Pitch and Yaw Control Valves, Control Arms and Links.	Conditions, Security, Leaks and Cracks.
	Spin and Drag Chute Installation.	Condition, Security & Safety Pin Installed
Empennage	Horizontal & Vertical Stabilizers, Elevator and Tab, Rudder and Tab	Attachment, Security, Cracks and Damage
Center Fuselage Lower	Exit Doors, Side Walls, Vector Nozzle Assembly	Condition, Security, Leaks and Cracks.
	Hydraulic Actuators and Attach Points	Condition, Security and Leaks
	Vector Nozzle Seals	Condition
Center Fuselage Upper	Fire Extinguisher Bottles	Condition & Security
	E.P.R. Transmitters	Condition & Security
	Electric Wiring in Lift Engine Inlet Area	Condition & Security
	Hydraulic Reservoir, Accumulator, Filters, Lines and Valves.	Condition, Security and Leaks

TABLE IX (Cont'd)

AREA OF INSPECTION	ITEMS OF INSPECT	INSPECT FOR
Wings (with Flaps Extended)	Wings, Flaps, Flap Coves, Ailerons, Wing Tip Pods	Security, Leaks and Damage.
	Roll Control Valves, Control Arms and Links	Security, Cracks and Leaks.
Landing Gear	Landing Gear Structure, Wheels, Support Fittings and Wheel Well Area.	Damage and Leaks
	Nose Landing Gear Door	Condition & Security
	Brakes	Damage and Leaks
Cruise Engines	Tail Pipes, Diverter Valves and Housing	Cracks
All Engines	Fuel Lines, Hydraulic Lines, Vent Lines, Control Rods, Brackets and Wiring.	Condition, Security, Leaks, Cracks and Safety.
	Intake, Inlet Guide Vanes and Turbine Blades.	Cracks and Condition
Cockpit	General Area and Canopy	Condition
General	Removal, DR and Squawk	Check for Open Items.

REFERENCES

1. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-S-8070, Model Specification XV-4B.

2. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-26-696, XV-4B Final Weight and Balance Report, May 1969.

3. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9271, Vol. I and II, XV-4B Configuration Definition - Subsystem Design and Analysis, May 1968.

4. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-26-587, Structural Design Criteria, XV-4B Hummingbird II, December 1966 (Rev. Sep 67 & June 68).

5. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-26-604, XV-4B Structural Design Loads, September 1967 (Revised June 1968).

6. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-23-606, Transient Loads Analysis Report, September 1967.

7. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-26-670, XV-4B Analysis Report, September 1968.

8. Loud Company, Report 1924L, Main Gear Stress Analysis Lockheed Georgia Company Model XV-4B, July 1967.

9. Loud Company, Report 1925L, Nose Gear Stress Analysis Lockheed Georgia Company XV-4B, July 1967.

10. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9194, XV-4B Flight Control System Synthesis and Analysis Report, June 1967.

11. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9271, Vol. II, Appendix IV, Configuration Definition - Subsystem Design and Analysis (Preliminary Submittal of Appendix IV, Volume II), October 1968.

12. AFFDL-TR-67-27, External Visibility Criteria for VTOL Aircraft, E. O. Roberts, March 1967.

13. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9959, XV-4B Escape System Development Program, November 1968.

14. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9052, XV-4B, Hummingbird Aerodynamic Analysis Report, May 1968.

15. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9092, XV-4B Wind Tunnel Test of a 0.16 Scale Model in the 17-Foot Test Section of the NASA-Langley 7 x 10 Foot Wind Tunnel, July 1967.

16. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9061, XV-4B Wind Tunnel Test of a 0.16 Scale XV-4B Hummingbird Model in the NASA-Langley 17-Foot Low Speed Wind Tunnel and the 7 x 10 Foot High Speed Wind Tunnel. September 1968.

17. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Unpublished Results From Wind Tunnel Tests Conducted at the University of Maryland, Tests 488 and 493, June and August 1967.

18. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9047, XV-4B Thermo-Propulsion Analysis Report, March 1968.

19. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9243, XV-4B Inlet Development Test Report, July 1967.

20. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9058, XV-4B Inverted Telescope and Balance System, June 1968.

21. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9282, Hot Cycle Testing of the XV-4B Propulsive and VTOL Control Systems, June 1968.

22. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9731, Performance Tests on Three Reaction Control Valve Configurations for V/STOL Aircraft, February 1968.

23. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-26-669, XV-4B Control Static Test Report, July 1968.

24. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9458, Landing Gear Drop Test Analysis, September 1967.

25. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-23-662-68, XV-4B Flutter Analysis and Ground Vibration Test, September 1968.

26. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-23-580-66, Vibration, Noise and Shock Qualification Requirements, October 1966.

27. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report SRD 72-23-664-68, XV-4B Duct System Dynamic Measurement Program, June 1968.

28. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-10073, XV-4B Acoustic Sonic Fatigue and Vibration Final Report, February 1969.

29. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9799, XV-4B Escape System Program Test Plan, Dual, Dynamic Firing, February 1968.

30. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER-9635, XV-4B Hummingbird II VTOL Research Vehicle Engineering Flight Test Development and Evaluation Tests, June 1969.

31. Lockheed-Georgia Company, A Division of Lockheed Aircraft Corp., Report ER 8785, Engineering Flight Test Detail Test Plan for the Model XV-4B, March 1967 (Revised September 1967).

UNCL

Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
a. Direct Jet VTOL Aircraft						
b. Fly-by-wire Control System						
c. Aircraft Development						
d. V/STOL Aircraft						

INSTRUCTIONS

1. ORIGINATING ACTIVITY: Enter the name and address of the contractor, subcontractor, grantee, Department of Defense activity or other organization *(corporate author)* issuing the report.

2a. REPORT SECURITY CLASSIFICATION: Enter the overall security classification of the report. Indicate whether "Restricted Data" is included. Marking is to be in accordance with appropriate security regulations.

2b. GROUP: Automatic downgrading is specified in DoD Directive 5200.10 and Armed Forces Industrial Manual. Enter the group number. Also, when applicable, show that optional markings have been used for Group 3 and Group 4 as authorized.

3. REPORT TITLE: Enter the complete report title in all capital letters. Titles in all cases should be unclassified. If a meaningful title cannot be selected without classification, show title classification in all capitals in parenthesis immediately following the title.

4. DESCRIPTIVE NOTES: If appropriate, enter the type of report, e.g., interim, progress, summary, annual, or final. Give the inclusive dates when a specific reporting period is covered.

5. AUTHOR(S): Enter the name(s) of author(s) as shown on or in the report. Enter last name, first name, middle initial. If military, show rank and branch of service. The name of the principal author is an absolute minimum requirement.

6. REPORT DATE: Enter the date of the report as day, month, year; or month, year. If more than one date appears on the report, use date of publication.

7a. TOTAL NUMBER OF PAGES: The total page count should follow normal pagination procedures, i.e., enter the number of pages containing information.

7b. NUMBER OF REFERENCES: Enter the total number of references cited in the report.

8a. CONTRACT OR GRANT NUMBER: If appropriate, enter the applicable number of the contract or grant under which the report was written.

8b, 8c, & 8d. PROJECT NUMBER: Enter the appropriate military department identification, such as project number, subproject number, system numbers, task number, etc.

9a. ORIGINATOR'S REPORT NUMBER(S): Enter the official report number by which the document will be identified and controlled by the originating activity. This number must be unique to this report.

9b. OTHER REPORT NUMBER(S): If the report has been assigned any other report numbers *(either by the originator or by the sponsor)*, also enter this number(s).

10. AVAILABILITY/LIMITATION NOTICES: Enter any limitations on further dissemination of the report, other than those imposed by security classification, using standard statements such as:

(1) "Qualified requesters may obtain copies of this report from DDC."

(2) "Foreign announcement and dissemination of this report by DDC is not authorized."

(3) "U. S. Government agencies may obtain copies of this report directly from DDC. Other qualified DDC users shall request through ________________."

(4) "U. S. military agencies may obtain copies of this report directly from DDC. Other qualified users shall request through ________________."

(5) "All distribution of this report is controlled. Qualified DDC users shall request through ________________."

If the report has been furnished to the Office of Technical Services, Department of Commerce, for sale to the public, indicate this fact and enter the price, if known.

11. SUPPLEMENTARY NOTES: Use for additional explanatory notes.

12. SPONSORING MILITARY ACTIVITY: Enter the name of the departmental project office or laboratory sponsoring *(paying for)* the research and development. Include address.

13. ABSTRACT: Enter an abstract giving a brief and factual summary of the document indicative of the report, even though it may also appear elsewhere in the body of the technical report. If additional space is required, a continuation sheet shall be attached.

It is highly desirable that the abstract of classified reports be unclassified. Each paragraph of the abstract shall end with an indication of the military security classification of the information in the paragraph, represented as *(TS)*, *(S)*, *(C)*, or *(U)*.

There is no limitation on the length of the abstract. However, the suggested length is from 150 to 225 words.

14. KEY WORDS: Key words are technically meaningful terms or short phrases that characterize a report and may be used as index entries for cataloging the report. Key words must be selected so that no security classification is required. Identifiers, such as equipment model designation, trade name, military project code name, geographic location, may be used as key words but will be followed by an indication of technical context. The assignment of links, rules, and weights is optional.