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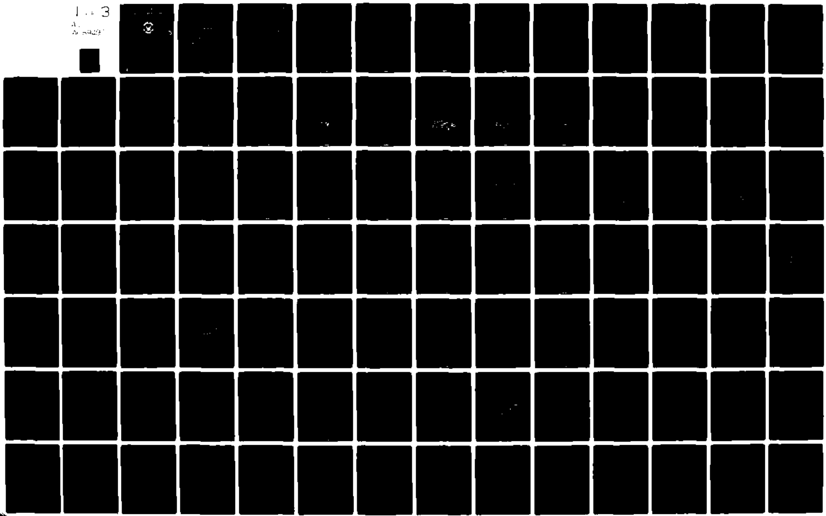

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APPLICATION OF POWERED HIGH LIFT SYSTEMS
TO STOL AIRCRAFT DESIGN

by

Frederick Donald Ameel

~~September~~ 1979

Thesis Advisor: M. F. Platzer

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Application of Powered High Lift Systems to STOL Aircraft Design

by

Frederick Donald Ameel
Lieutenant Commander, United States Navy
B.S., United States Naval Academy, 1966

Submitted in partial fulfillment of the requirements for the degree of

MASTER OF SCIENCE IN AERONAUTICAL ENGINEERING

from the

NAVAL POSTGRADUATE SCHOOL
September 1979

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ABSTRACT

The development of VTOL technology over a thirty year period was reviewed. Various powered-lift concepts were explored and recent application analyzed. The ACSYNT computer program, developed by the NASA Ames Research Center, was used to predict the takeoff performance of a proposed jet STOL transport utilizing the Lockheed AIBF system. A correlation of the performance of the S-3A VIKING aircraft was performed.

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I. INTRODUCTION

The acronym V/STOL stems from the union of two others, VTOL (vertical takeoff and landing) and STOL (short takeoff and landing). The aerodynamics of V/STOL aircraft involves the study of the generation of lifting forces at low forward speeds. However, in producing the necessary lift, the vehicle's cruise performance must not be degraded [Ref. 4-3].

The term VTOL is unambiguous in its meaning: a true vertical ascent/descent capability, transversing zero distance over the ground. But STOL implies "short with relation to what standard?". Since the fall of 1954, when the Bell Air Test Vehicle (ATV) ushered in the jet V/STOL era in the United States, the definition of STOL has been widely interpreted. Table 1-1 displays some of the many proposals offered in attempting to set a standard for short takeoff and landing distance/clearance height combinations.

Definitions of STOL Distances (Ft.)

Ground Distance	Obstacle Ht.	Reference
1000.	50.	1-1, 3-2
1650.	35.	3-1
500.	50.	3-4
1700-2000.	50.	3-6
1250.	50.	4-1
1500-2000.	35.	4-3

TABLE 1-1

A more qualitative approach was taken by the Aerospace Industries Association who proposed:

> "Both VTOL and STOL aircraft be defined as those which are dependent upon propulsive lift and/or powered lift augmentation for performance and/or control in selected flight regimes, with the difference between the two being the VTOL's capability of vertical flight and of hovering over a fixed point in zero wind". [Ref. 1-2]

Although V/STOL research has been actively pursued for some thirty years, numerous uncertainties have hindered the full-scale exploitation in the U.S. of this promising technology. In a special issue on V/STOL technology [Ref. 1-3], <u>Aviation Week and Space Technology</u> pinpointed the dilemma-

> "There are two basic questions which must be answered before any real push can be expected in the development of a total V/STOL system:
>
> Is there a civil market for V/STOL transport operations?
>
> Is there a military mission requiring VTOL or STOL aircraft?"

That assessment was made <u>eleven</u> years ago!

The problems of noise and congestion which have plagued major U.S. airports for the past decade have received much notoriety primarily from environmental and local community interests. Two proposed solutions to airport congestion have been suggested. First was the center-city STOL port concept (1971) which advocated the use of STOL aircraft for short haul passenger service. Ironically, the concern of the U.S. public over noise and pollution has made this concept difficult to sell [Ref. 4-3]. Recently though, a successful short haul passenger operation between Ottawa and Montreal, Canada [Ref. 1-4] has demonstrated that the STOL port idea can work. As a second solution, U.S. commuter airlines have advocated the use of short, separate runways at major airports. This idea employs separate air traffic patterns from the larger commercial carriers to expedite short haul operations. An experiment is about to begin at Washington National Airport in which Ransom Airlines will utilize de Havilland DHC-7 turboprop STOL transports to test the efficiency of the dual-pattern operation [Ref. 4-3].

After the same eleven year period there is still only one operationally deployed tactical jet V/STOL in the free world; that being the Hawker Siddeley HARRIER (AV-8A) which was developed by the British. The U.S. Marine Corps has pioneered the first U.S. military use of a non-rotary wing V/STOL aircraft to meet their close air support needs. Although only the first iteration in the development of capable tactical jet V/STOL vehicles (i.e. payload, range, flying qualities), the HARRIER represents a benchmark from which to assess future designs.

The quest for a U.S. V/STOL military transport was begun in the early 1970's with the creation of the Advanced Medium Short takeoff and landing Transport (AMST) program. The goals set forth by the U.S. Air Force (see Table 4-3) dictated that advanced STOL technology be utilized. Although the AMST program has undergone severe budget reductions, prototype aircraft have been developed and tested. The performance demonstrated by the YC-14 and YC-15 STOL transports exemplify the potential of powered high lift systems.

The purpose of this thesis is twofold. First, the development of VTOL technology over a thirty year period is reviewed. Various powered-lift concepts are then explored and recent applications analyzed. Second, the ACSYNT computer program, developed by NASA Ames for aircraft conceptual design studies, is used to predict the takeoff performance of a proposed jet STOL transport utilizing a powered high lift system.

II. VTOL AIRCRAFT DEVELOPMENT HISTORY, 1949-1979

In 1939 when Russian immigrant Igor Sikorsky built and flew his VS-300, the first completely controllable single-rotor helicopter, it marked the transition point where pioneering work terminated and engineering and production of practical helicopters began. Although the idea of a man-made flying machine capable of vertical take-off and landing was recorded as early as the 15th century by da Vinci, an Englishman named Sir George Cayley first attempted to integrate the features of the helicopter and the airplane. In 1843 his "aerial carriage" design incorporated both a rotor system for vertical take-off/landing and separate propellers for propulsion in conventional airplane flight.

One of the same fundamental problems that hampered early helicopter pioneers also discouraged early VTOL experimenters; that of achieving a satisfactory horsepower to engine weight ratio with available powerplants. As aircraft reciprocating engines improved, the early helicopters fared better due to utilization of relatively large diameter rotors. It was not until the development of the turbine engine in the 1940's that VTOL aircraft development received a much needed shot in the arm. Although early turboprop aircraft engines exceeded unity with their power-to-weight ratios, continuous refinement has resulted in turboprops capable of producing three times the power output of a reciprocating engine of equal weight. [Ref. 2-1]

The brief history of VTOL aircraft which follows covers the major developments in the United States, France, England and the Federal Republic of Germany over a thirty year period.

A. MAJOR DEVELOPMENTS IN THE USA

(I)	(II)	(III)
Concept Feasibility	Tri-Service Operational Feasibility	USN Dispersed Forces Concept

1949 — 1959 — 1969 — 1979

"VTOL DEVELOPMENT PHASES"

1. <u>Phase - I</u>

This period was characterized by the investigation of the performance characteristics of various VTOL concepts such as:

a. Unloaded rotor (McDonnell XV-1)

b. Tilt - rotor (Bell XV-3)

   Tilt - propeller (Curtis-Wright X-100)

   Tilt - ducted fan (Doak VZ-4)

   Tilt - wing (Vertol VZ-2, Hiller X-18)

c. Vertical attitude (Convair XFY-1, Lockheed XFV-1, Ryan X-13)

d. Lift engine (Bell ATV, Bell X-14)

DESIGNATION:	XV-1
MANUFACTURER:	McDonnell Aircraft Company
SPONSOR:	USAF
CONCEPT:	Unloaded Rotor Principle (Compound Helicopter)
MILESTONES:	Go - Ahead 1949 Conversion from helicopter to airplane flight 1955 Project termination 1956
WEIGHT:	Unknown
ENGINES:	(1) Continental R-975 (550 HP) piston engine powered compressors for the rotor tip jets (for vertical flight) and a pusher propeller for cruise flight.
LAYOUT:	See Fig. 1
COMMENTS:	Maximum speed of 200 mph was achieved but detracting features of the design included its piston engine and pusher-prop. configuration.
SOURCE:	Ref. 2-1, p.77

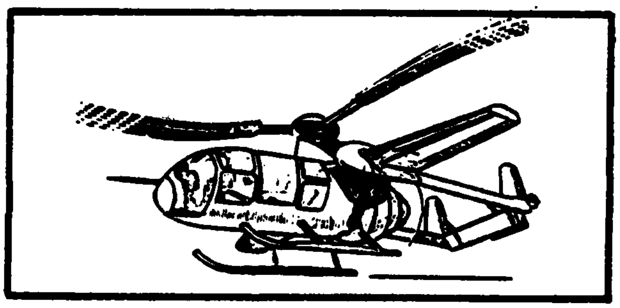


FIGURE 2-1

DESIGNATION:	XV-3 "Convertiplane"
MANUFACTURER:	Bell Helicopter Company
SPONSOR:	USA, USAF, NASA
CONCEPT:	Tilt Rotor Principle
MILESTONES:	Go - Ahead 1951 First VTO Aug 1955 First Conversion Dec 1958 Flight testing terminated 1959
WEIGHT:	4800 LBS
ENGINES:	(1) P & W R-985 reciprocating engine driving two rotors
LAYOUT:	See Fig. 2
COMMENTS:	Two-bladed semirigid rotors were substituted for original three-bladed fully articulated rotors. Poor directional stability during transition required the addition of a ventral fin. During evaluation by the USAF the following deficiencies were noted:

- Lateral instability during in-ground-effect hover
- Requirement for large increase in power as hovering flight was approached
- Excessive blade flapping during conventional flight maneuvering
- Poor longitudinal dynamic stability in high-speed conventional flight
- High drag

SOURCE: Ref. 2-1, p. 73
Ref. 2-2, App. N

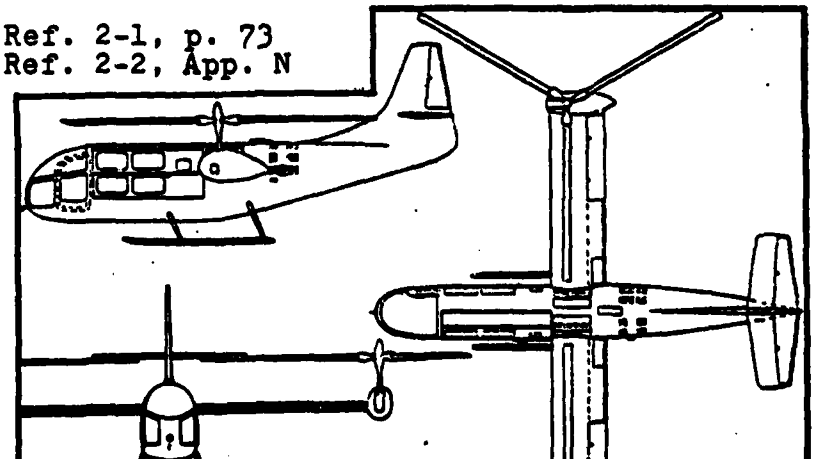


FIGURE 2-2

DESIGNATION:	X-100
MANUFACTURER:	Curtis-Wright Aircraft Company
SPONSOR:	Company venture
CONCEPT:	Tilt Propeller principle (radial lift force)
MILESTONES:	Development begun 1958 First VTO 1959 Transition from VTO to forward flight Mar 1960 NASA evaluation 1960-1961
WEIGHT:	3730 LBS
ENGINES:	(1) T-53 turbine engine (825 SHP) driving two 3-bladed propellers
LAYOUT:	See Fig. 3
COMMENTS:	Concept was less efficient during transition and STOL flight since the wing was not fully immersed in the propeller slipstream. Numerous transitions were made from hover to forward flight. A fairly simple design without serious wing-stall problems.
SOURCE:	Ref. 2-1, p.96

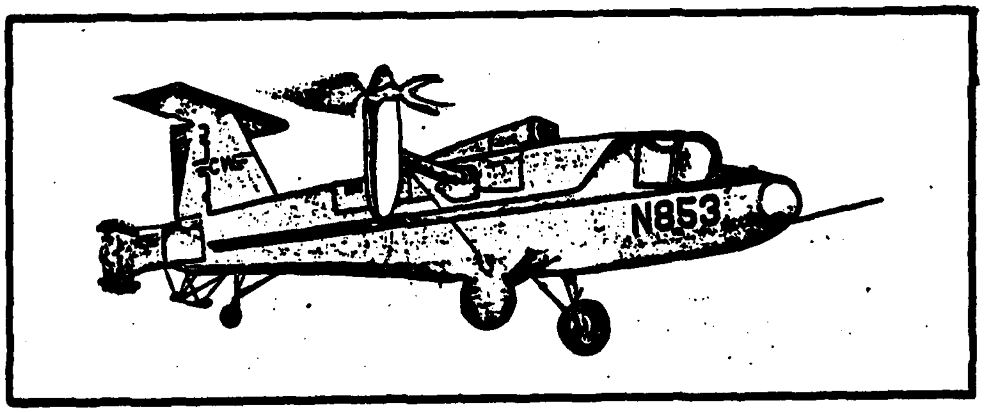


FIGURE 2-3

DESIGNATION: VZ-4

MANUFACTURER: Doak Aircraft Company

SPONSOR: USA, NASA

CONCEPT: Thrust - Tilting Ducted Fan principle

MILESTONES:

First flight	Feb 1958
Transition from VTO to forward flight	1959
Accepted by USA	Sep 1959
All rights sold to Douglas Aircraft Company	1961

WEIGHT: 3200 LBS

ENGINES: (1) Lycoming T-53 turbine engine (840 SHP) driving two 4 ft. diameter ducted fans

LAYOUT: See Fig. 4

COMMENTS: Exhibited a nosing-up tendency caused by the ducts during transition from hovering to forward flight. Less efficient in STOL operations due to non-uniform lift distribution at moderate speeds with partial duct lift; higher power required due to increased drag.

SOURCE: Ref. 2-1, p. 115

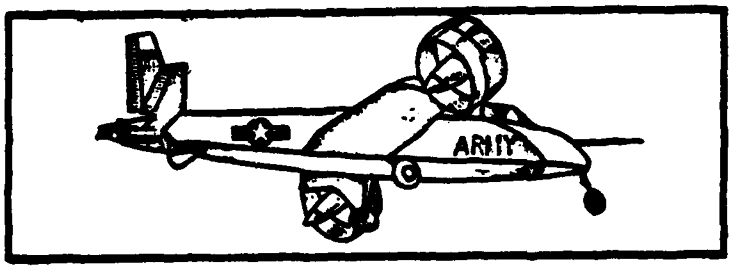


FIGURE 2-4

DESIGNATION: VZ-2

MANUFACTURER: VERTOL Corporation

SPONSOR: USA, ONR, NASA

CONCEPT: Tilt - Wing principle

MILESTONES:

Go - Ahead	1956
First flight	1957
First transition from VTO to forward flight and back	Jul 1958
NASA evaluation	1960

WEIGHT: 3000 LBS

ENGINES: (1) Lycoming T-53 turbine engine driving two 3-bladed propellers (wing mounted) and two tail control fans

LAYOUT: See Fig. 5

COMMENTS: Severe wing stall experienced at high angles of wing incidence during transition maneuvers. NASA added a drooped leading edge to the wing to soften the stall. Random motions experienced in hovering close to the ground.

SOURCE: Ref. 2-1, p. 93

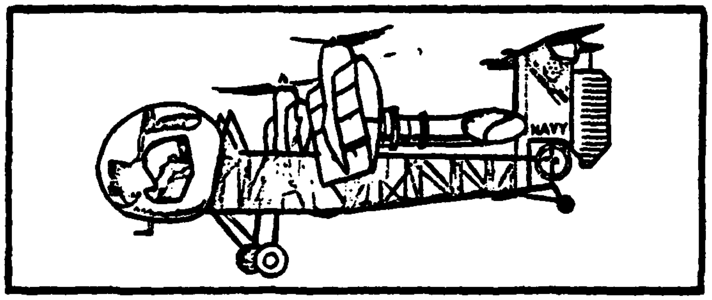


FIGURE 2-5

DESIGNATION: X-18

MANUFACTURER: Hiller Aircraft Company

SPONSOR: USAF

CONCEPT: Tilt - Wing principle

MILESTONES:

Go - Ahead	Feb 1957
First flight (VTO)	
First conventional take off	Nov 1959
Project termination	Jul 1961

WEIGHT: 32,000 LBS

ENGINES: (2) 5000 SHP Allison YT-40A turboprops turning 16 ft. counterrotating coaxial propellers, (1) J-34 turbojet for pitch control

LAYOUT: See Fig. 6

COMMENTS: No complete transitions from hover to cruise flight were made. Drooped wing leading edge reduced severity of wing stall buffet during transition maneuvers. Roll control for hover/low speed flight was unsatisfactory plus lack of interconnection between engines contributed to an unacceptable risk level and flight tests were halted.

SOURCE: Ref. 2-1, p. 94

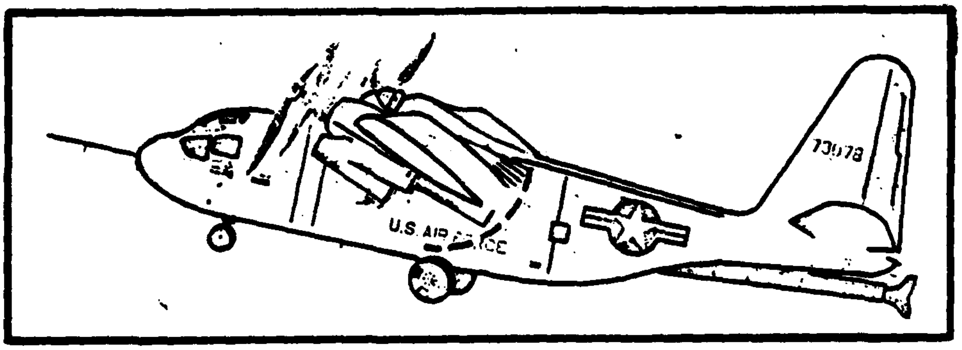


FIGURE 2-6

DESIGNATION: XFY-1 "POGO"

MANUFACTURER: Convair Division, General Dynamics Corporation

SPONSOR: USN

CONCEPT: Vertical Attitude principle

MILESTONES:

Go - Ahead	1951
First flight (VTO)	Aug 1954
First VTO and Transition	Nov 1954
Project termination	1956

WEIGHT: 14,000 LBS

ENGINES: (1) 5000 SHP Allison YT-40A turboprop driving 16 ft. counterrotating, coaxial propellers

LAYOUT: See Fig. 7

COMMENTS: Except for the helicopter, the first VTOL aircraft to accomplish the complete VTOL operation. Vertical attitude posed maintenance and pilot orientation problems. Persistent problems with engine and propellers led to project cancellation.

SOURCE: Ref. 2-1, p. 85

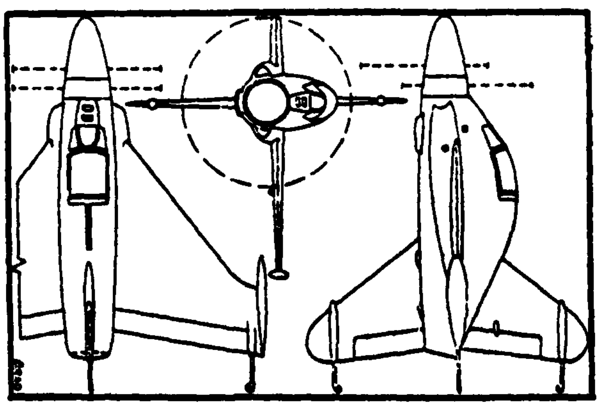


FIGURE 2-7

DESIGNATION: XFV-1 "POGO"

MANUFACTURER: Lockheed Aircraft Corporation

SPONSOR: USN

CONCEPT: Vertical Attitude principle

MILESTONES:

Go - Ahead	1951
First flight (HTO)	1954
*(No vertical takeoffs/ landings attempted)	
Project termination	1956

WEIGHTS: Approximately 14,000 LBS

ENGINES: Same powerplant as XFY-1

LAYOUT: See Fig. 8

COMMENTS: Thirty-two transitions from cruise flight to hover were made at altitude only. Same engine problems which plagued the Convair XFY-1 forced cancellation of this project.

SOURCE: Ref. 2-1, p. 86

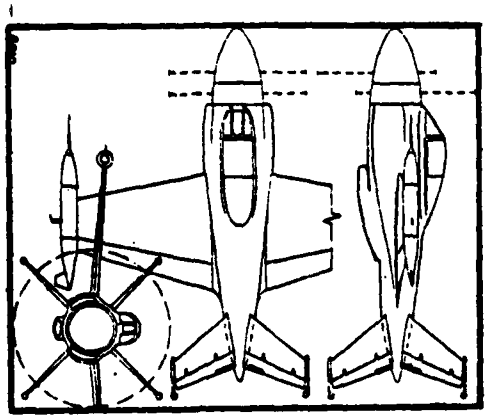


FIGURE 2-8

DESIGNATION: X-13

MANUFACTURER: Ryan Aircraft Company

SPONSOR: USAF

CONCEPT: Vertical Attitude principle

MILESTONES:

Go - Ahead	1953
First horizontal takeoff	Dec 1955
First hover (VTO)	May 1956
First HTO and transition	Nov 1956
First VTO and transition	Apr 1957

WEIGHT: 7500 LBS

ENGINES: (1) Rolls-Royce Avon turbojet of approximately 9500 LBS thrust

LAYOUT: See Fig. 9

COMMENTS: Elaborate SAS required due to (a) large gyroscopic coupling effects produced by the large and neavy engine, (b) during transition the highly swept delta wing was stalled at angles of attack from 30 to 90 degrees. Numerous complete VTOL operations were made from its landing service trailer. Overall, a very successful test project. Development was cancelled in favor of horizontal attitude configurations.

SOURCE: Ref. 2-1, p. 133

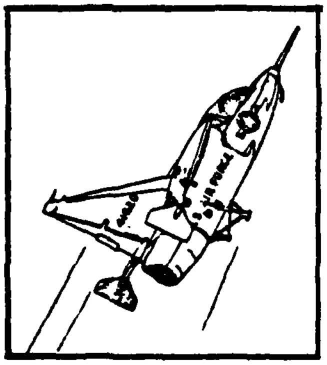


FIGURE 2-9

DESIGNATION: Air Test Vehicle (ATV)

MANUFACTURER: Bell Aircraft Company

SPONSOR: Company venture

CONCEPT: Lift - Engine (jet) principle

MILESTONES:

Go - Ahead	Mar 1953
First hover	Nov 1954
Last flight	Mar 1955

WEIGHT: 2000 LBS

ENGINES: (2) Fairchild J44-R-1 Missile Turbojets

LAYOUT: See Fig. 10

COMMENTS: First jet V/STOLL flown in U.S. No stability augmentation was required. Engines rotated on a common shaft through the fuselage. Each reaction control system (RCS) nozzle was mechanically linked to the corresponding aerodynamic control surface. Partial transitions were accomplished at 30 to 40 ft. altitudes. No complete transitions were performed. Engines were not manrated and limited to five hours.

SOURCE: Ref. 2-1, p. 136
Ref. 2-2, App. A

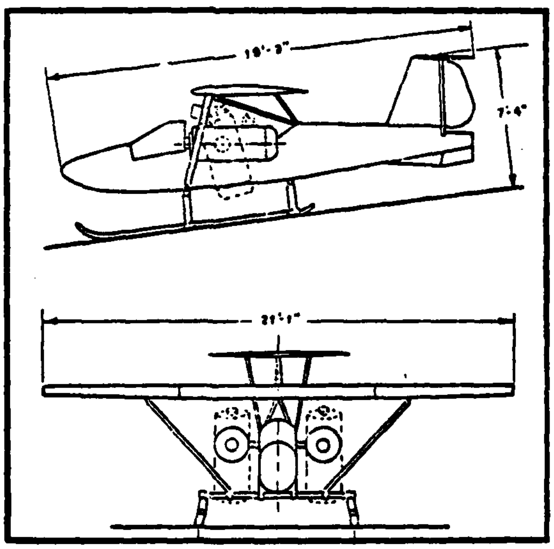


FIGURE 2-10

DESIGNATION:	X-14/X-14A	
MANUFACTURER:	Bell Aerosystems Company	
SPONSOR:	USAF, NASA	
CONCEPT:	Deflected Thrust (jet) principle	
MILESTONES:	Go - Ahead (X-14)	Jul 1955
	First hover	Feb 1957
	First complete VTO cycle	May 1958
	NASA Modif. to X-14A	1960
WEIGHT:	3500 LBS	
ENGINES:	X-14 (2) Armstrong - Siddeley ASV-8 turbojets	
	X-14A (2) G. E. J85-5 turbojets	
LAYOUT:	See Fig. 11	
COMMENTS:	Success of ATV led the USAF to support this design as a horizontal-attitude VTOL (HATOL) demonstrator. Conventional flight mode employed normal flight controls while jet reaction nozzles were used during hover and transition. Basic X-14 had no stability augmentation. Landing gear had to be lengthened and takeoff accomplished from an elevated platform of perforated steel to alleviate ground proximity HGI and suckdown effects. Aircraft had to rise quickly out of ground effect or hot-gas ingestion would result in severe power loss. Installation of J-85 engines provided an improved thrust margin (more bleed air for RCS) although ground effects problems remained. An analog response feedback variable stability and control system was installed in the X-14A.	
SOURCE:	Ref. 2-1, p. 140 Ref. 2-2, App. B	

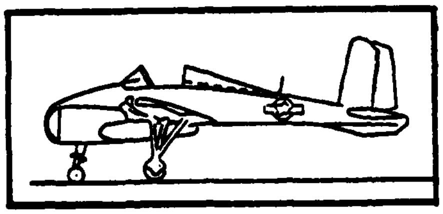


FIGURE 2-11

2. <u>Phase - II</u>

In the fall of 1959 the Perkins Committee, which was formed to assess VTOL technology in the U.S., recommended:

> "The U.S. VTOL research program (test beds) demonstrated the technical feasibility that V/STOL aircraft can be built in a number of configurations which contain the VTOL capability of rotary wing aircraft, yet do not have the limitations of speed, range and complexity of helicopters. However, the operational suitability of V/STOL to meet military requirements must now be domonstrated. Unless a program for operational suitability is initiated, the uncertainty that exists today will continue." [Ref. 2-1]

As a result of this study, the DOD undertook the development of three V/STOL aircraft to prove their suitability. The concepts were:

a. Tilt - wing principle (LTV XC-142A)

b. Tilt - ducted propeller principle (Bell X-22A)

c. Tilt - propeller principle (Curtis-Wright X-19A)

Other major V/STOL development programs during the period 1960-1969 included:

d. Fan - in - wing principle (Ryan XV-5)

e. Thrust agumenting ejector principle (Lockheed - Georgia XV-4A)

f. Lift/Lift cruise engine concept (Lockheed - Georgia XV-4B)

g. Tilt - wing principle (Canadair CL-84)

h. Deflected - slipstream principle (Ryan VZ-3)

DESIGNATION:	XC-142A
MANUFACTURER:	LTV Aerospace Corporation along with Ryan Aeronautical and Fairchild-Hiller
SPONSOR:	Tri-Service Program
CONCEPT:	Tilt - wing principle
MILESTONES:	Go - Ahead — Jan 1962 First VTO/Hover — Dec 1964 First VTO and transition — Jan 1965 First carrier ops — May 1966 First tactical field demo — Oct 1966 Paris Airshow demo — Jun 1967 Last flight (#488) — 09 Oct 1967
WEIGHT:	39,000 LBS
ENGINES:	(4) T64-G.E. - 1 turboshaft engines (2850 SHP each) driving four 15.6 ft. diameter propellers through four inter-connected propeller gear-cases.
LAYOUT:	See Fig. 12

COMMENTS:

This program's objective was to demonstrate the VFR/IFR operational capabilities of the tilt-wing concept. A total of five aircraft were built. The SAS provided rate and attitude damping in roll and pitch and rate damping in yaw and height. It only functioned in the hover and transition flight regimes. For STOL operations the wing angle was limited to 35 degrees maximum due to Lat. - Dir. stability degradation in ground effect. Flight tests showed the span efficiency of the wing to be 28% lower than wind tunnel tests had predicted. Cross-shafting of the four propellers allowed the XC-142A to cruise on only two engines and increase its specific range by 20%. A 1966 flying qualities and performance test evaluation was conducted by the Tri-Service test team and concluded:

(1) Handling qualities in vertical flight were satisfactory with SAS - on, but sufficient longitudinal control power was not available at high wing angles.
(2) Conventional flight flying qualities were unsatisfactory.

(3) Very high noise and vibration levels resulted in operational and structural problems and excessive flight restrictions.
(4) Cockpit temperatures as high as 155 F occurred due to "greenhouse" effect.

<u>SOURCE</u>: Ref. 2-2, App. L

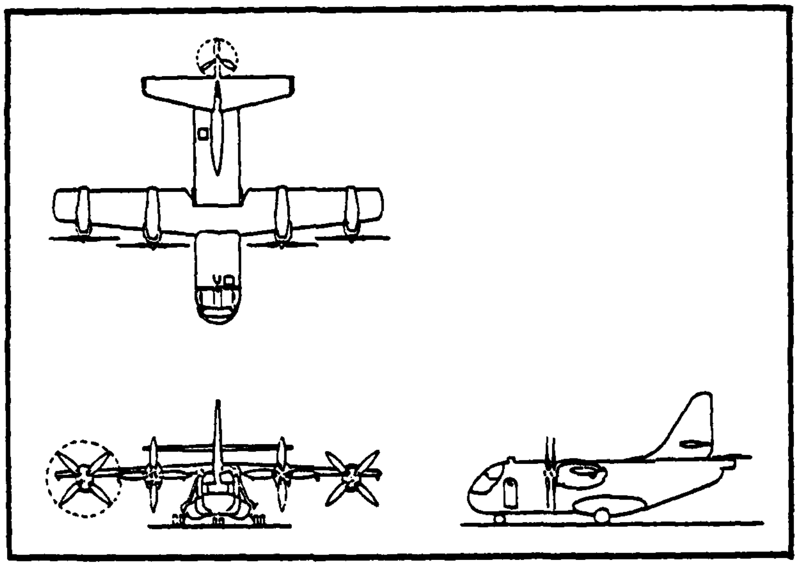


FIGURE 2-12

DESIGNATION:	X-22A
MANUFACTURER:	Bell Aerosystems Company
SPONSOR:	Tri-Service Program
CONCEPT:	Tilt - ducted propeller principle

MILESTONES:

Go - Ahead	Nov 1962
First flight	Mar 1966
Public VTOL transition demo	May 1967
First military pilot evaluation	Jan 1968
Aircraft acceptance	May 1969

WEIGHT: 14,000 LBS

ENGINES: (4) YT58-GE-8D turboshaft engines (1250 SHP each) driving four 7.0 ft. diameter ducted propellers through interconnecting shafts and gearboxes.

LAYOUT: See Fig. 13

COMMENTS: The X-22A was developed as a half-size transport research vehicle. In addition a variable stability system was incorporated in the design. A dual-channel SAS provided rate damping in pitch, roll and yaw. The 3-bladed variable pitch propellers were composite structures consisting of steel spars with fiberglass blade shells. Control moments were obtained from two independent sources: (1) differential deflection of ELEVONS or (2) differential thrust produced by variable blade pitch. Minor problems which occurred during early test flights included: structural resonance of duct skin under the propeller tips, duct vibration during hover and engine overheat. The 1968 military preliminary evaluation listed as a major deficiency the inherent high sideforce characteristics with no means for providing a direct counter-control force that did not significantly introduce controls coupling. Although hot-gas ingestion and suckdown were not evidenced in the X-22A, major hover effects in ground proximity included (1) a net positive thrust cushion effect of 12% during VTO due to fountain effect, (2) significant longitudinal and directional control instabilities were evidenced in ground effect. The X-22A continues to serve as a valuable V/STOL research vehicle.

SOURCE: Ref. 2-2, App. J

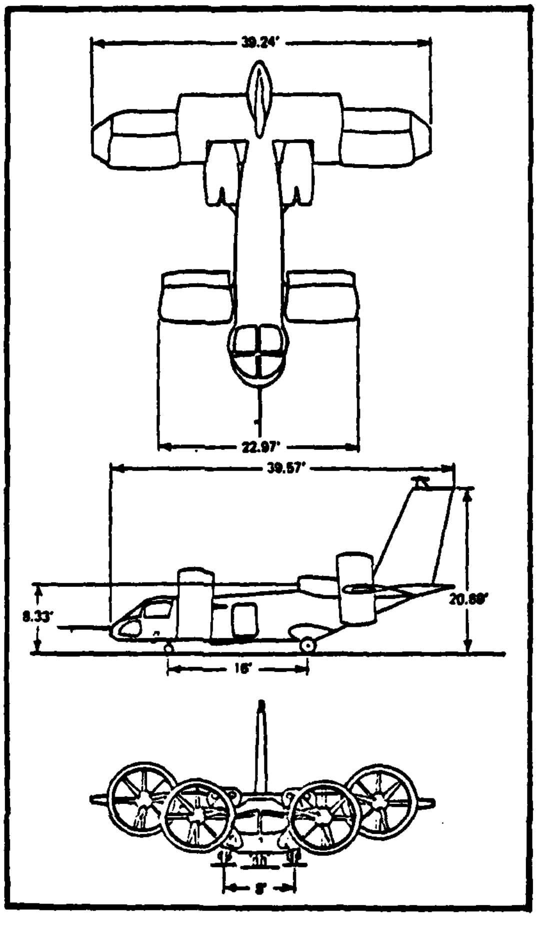


FIGURE 2-13

DESIGNATION:	X-19A	
MANUFACTURER:	Curtis-Wright Corporation	
SPONSOR:	Tri-Service Program	
CONCEPT:	Tilt - propeller principle (Radial Lift Force)	
MILESTONES:	Go - Ahead	Jul 1962
	First VTO flight (hover)	Jun 1964
	#1 aircraft destroyed	Aug 1965
	Project terminated	1966
WEIGHT:	13,600 - 14,700 LBS	
ENGINES:	(2) Lycoming T55-L-5 turboshaft engines (2200 SHP each) driving four 13 ft. diameter propellers through interconnecting shafts and gearboxes	
LAYOUT:	See Fig. 14	
COMMENTS:	This aircraft was initially a company venture which grew out of the development of the X-100 flying test bed. In 1962 the USAF systems Command ordered two prototypes for a tri-service evaluation. The X-19A was a six-seat, twin-engined high-wing aircraft with four propellers mounted on tilting wing-tip nacelles. For VTO operations the nacelles were rotated upward to the ninety degree position and control was maintained by varying the prop blade angle; pitch control through variation of the pitch of fore and aft propellers, roll control by variation of pitch of the starboard and port propellers and yaw control by variation of pitch of diagonally opposite propellers. Normal flight employed conventional control surfaces. Initial flight testing consisted of hover and low-speed transitions. Early lift-offs were performed with the SAS off to analyze longitudinal and lateral control inputs for undesirable characteristics. As flight testing progressed, accurate control of the X-19A was demonstrated in forward, backwards and sidewards flight. Translational flight, to forward speeds of 50 mph, confirmed the positive translational lift which was measured by model tests as well as the control requirements occurring at low forward speeds. The prototype aircraft was lost during a FAA certification check in which a propeller gear box failed in fatigue.	

SOURCE: Ref. 2-2, p. 98
Ref. 2-3, '65-212
Ref. 2-5

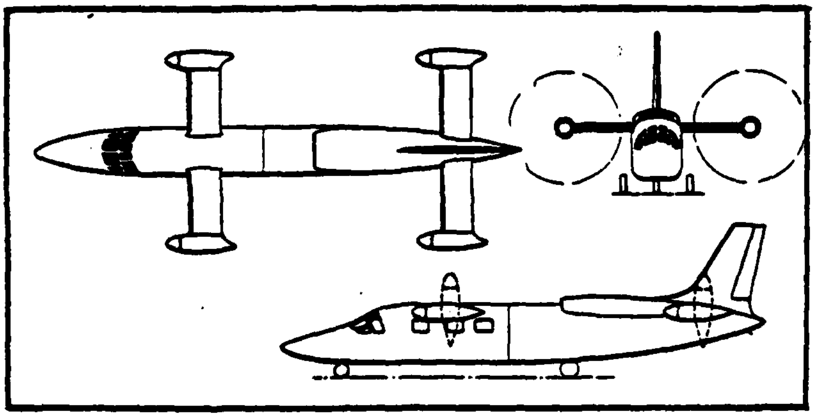


FIGURE 2-14

DESIGNATION:	XV-5A/B	
MANUFACTURER:	Ryan Aeronautical Company	
SPONSOR:	USA, General Electric Company	
CONCEPT:	Fan-in-wing principle	
MILESTONES:	Ryan and G. E. development begun	1959
	Prototypes ordered by USA	Nov 1961
	First CTOL flight	May 1964
	First VTOL flight	Jul 1964
	First complete transition	Nov 1964
	Aircraft accepted by USA	Jan 1965
	Fatal crash (#1 aircraft)	27 Apr 1965
	Resumption of flight tests	Jun 1965
	Modification for NASA as XV-5B	1966
WEIGHT:	12,300 LBS (Max VTO)	
ENGINES:	(2) J85-GE-5 turbojet engines (2568 LB thrust each) driving two wing fans and a single nose fan. Jet exhaust was directed to the tip-turbine driven fans through diverter valves.	
LAYOUT:	See Fig. 15	

COMMENTS:

Construction of the two prototypes was the culmination of several years' work by G.E. in lift-fan technology. The XV-5 had two primary flight control systems: one for powered lift and a separate one for conventional flight. The fan system augmented the turbojet thrust during powered-lift flight. During transition from VTO to horizontal flight, louvers situated on the lower surface of the fans vectored the fan exhaust rearward. Once flying speed (140 mph) was achieved, the diverter valves were moved to the straight-through position and the jet engines operated conventionally. At this time the wing-fan doors and louvers were closed. Speeds of up to 450 mph were achieved during flight tests. A SAS provided rate of attitude stabilization of the aircraft.

300 hours of full scale wind tunnel testing was completed prior to flight. Tethered hovering was not attempted due to misgivings about the artificial loads induced. Ground effects on stability and control in hover were apparent up to wheel heights of six feet and hot gas ingestion was evidenced up to ten feet wheel height. During the transition from conventional

flight to a vertical landing, a strong nose-up pitch was evident with fan start-up. The Army technical evaluation concluded that SAS characteristics were excellent and compatibility between the fan-mode and jet-mode control systems was good, but poor hover flying qualities were experienced in ground effect. After rebuilding the aircraft as the XV-5B, NASA used it to assess precise flight path control during steep decelerating approaches in visual flight conditions.

SOURCE: Ref. 2-2, App. M
Ref. 2-1, p. 127
Ref. 2-3, '65-297

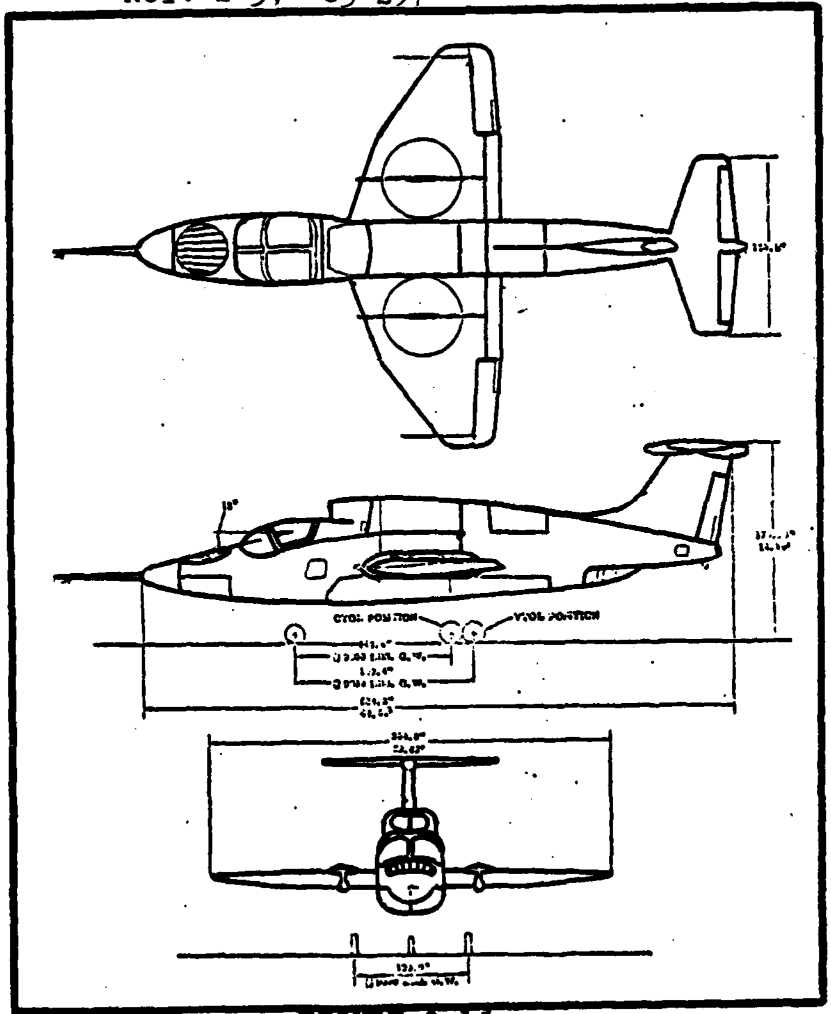


FIGURE 2-15

DESIGNATION:	XV-4A "Hummingbird"
MANUFACTURER:	Lockheed-Georgia Company
SPONSOR:	USA
CONCEPT:	Augmented - jet ejector principle
MILESTONES:	Two prototypes ordered by USA — Sep 1961 First CTOL flight — Jul 1962 Tethered hover tests — Nov 1962-Feb 1963 First VTO and transition — Nov 1963 Fatal crash (#1 aircraft) — 10 Jul 1964 Project termination — 1965
WEIGHT:	7200 LBS (max. VTO); maximum thrust-to-weight ratio achieved was 1.08
ENGINES:	(2) P & W JT-12A-3 turbojets (3000 LBS thrust each). Exhaust gas was directed by a diverter valve to either conventional tailpipes for CTOL operations or into an ejector manifold for VTOL flight.
LAYOUT:	See Fig. 16
COMMENTS:	The XV-4A program was funded as a proof-of-concept venture with minimal guarantees and was an out-growth of some earlier jet ejector studies by Lockheed. For powered-lift flight, the engine exhaust gases were directed into a mixing chamber via a manifold. Thrust augmentation was obtained by inducement of secondary airflow into the ejector system. This "jet pump effect" increased the mass flow by about 5.5 times. Three-axis jet reaction controls were used for hover and were supplemented by conventional control surfaces during the transition. Unique features incorporated to aid longitudinal control during VTOL phases included (1) an automatic elevator drooper to provide a nose-down pitching moment to balance a counter moment (momentum drag moments) that increased with increasing forward velocity, (2) a BLC system for the horizontal stabilizer/elevator to aid in overcoming the inherent positive pitching moment of the ejector concept. A manually adjustable gain feature was incorporated in the SAS to permit evaluation of VTOL handling qualities.

Over 300 hours of scale model wind tunnel testing was completed prior to flight. The work-up during flight testing was slow and methodical. Tethered functional ejector checks were conducted on an elevated platform to eliminate ground effect and it was determined that the design vertical thrust level was not being achieved. A simple teeter-board rig was used to support the aircraft while checking roll control power. As a result, additional wing tip control nozzles were added. The first eight flights were CTOL and their purpose was to evalutate stability and control. The first hover was accomplished with a 300 LB lead weight attached beneath the fuselage and an additional twenty-five hover flights were required to determine usable SAS gain settings. The first transition attempts were made starting from a taxi run and later transition was made from a hover. But a combination of low thrust margin and hot exhaust gas reingestion near the ground severely handicapped the XV-4A's performance. The flight test program concluded that (1) jet ejector augmentation was feasible, (2) reaction control systems provided an excellent means of VTOL aircraft, (3) rate-only SAS was adequate for VTOL operations.

<u>SOURCE</u>: Ref. 2-2, App. I
Ref. 2-3, '64 p. 254

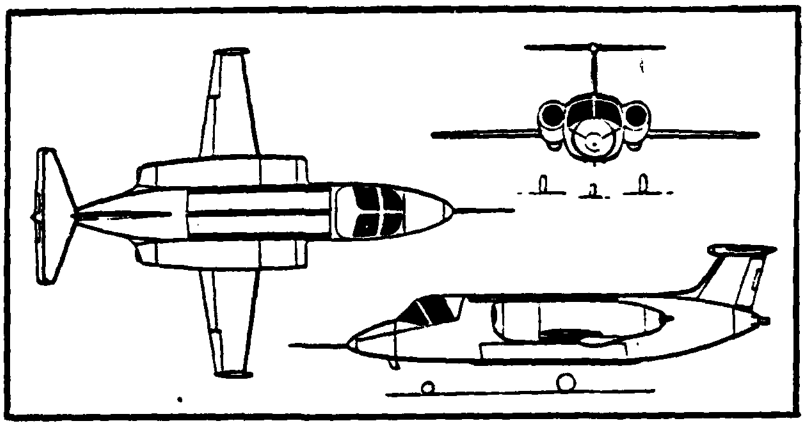


FIGURE 2-16

DESIGNATION: XV-4B "Hummingbird II"

MANUFACTURER: Lockheed-Georgia Company

SPONSOR: USAF

CONCEPT: Lift/lift-cruise engine principle

MILESTONES:

Contract awarded by USAF	Sep 1966
Capitve testing	Jul-Sep 1968
First CTOL flight	28 Sep 1968
Aircraft destroyed in crash	14 Mar 1969
*No VTOL free-flights were accomplished	

WEIGHT: 12,580 LBS (max VTO)

ENGINES: (6) G.E. YK85-19 turbojets (3015 lbs uninstalled each); four mounted vertically in the center fuselage, other two conventionally mounted in fuselage nacelles

LAYOUT: See Fig. 17

COMMENTS: The XV-4B was developed for use in VTOL and transition flight investigations by the USAF. Its close resemblance to the earlier XV-4A (1962-1965) external configuration was retained to minimize unknown configuration effects on conventional flight characteristics. In addition to housing the two lift-cruise engines, the fuselage nacelles contained the diverter valves which directed the exhaust gases to either the horizontal-thrust nozzles or the vertical-lift nozzles beneath the fuselage. The four lift-only engines could be started with bleed air from the two lift-cruise powerplants. The primary flight control system was fly-by-wire and included an integral SAS to provide augmented damping about all three axes. A conventional mechanical system served as a backup. Reaction control jets were provided for VTO flight.

Extensive wind tunnel work was completed using a 1/16 - scale model in which the lift engine system was simulated by using externally supplied compressed air to power six ejector units. An inverted telescope test rig was built which permitted captive VTOL operations. The XV-4B was lost in a conventional flight accident early in the test program. Twenty-three flights had been conducted to explore this flight regime and the high-speed end of transition, down to 95 kts. Reliability of the fly-by-wire control system was good and no in-flight failures were experienced.

SOURCE: Ref. 2-2, App. C
Ref. 2-3, '69 - 369

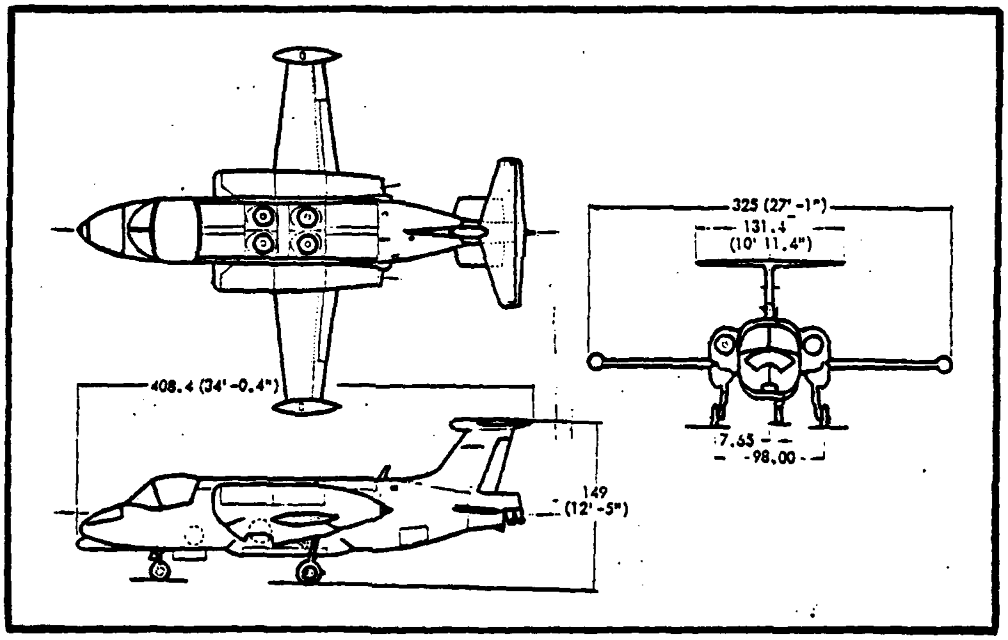


FIGURE 2-17

DESIGNATION:	CL-84
MANUFACTURER:	Canadair Limited
SPONSOR:	Canadair and Canadian Department of Industry
CONCEPT:	Tilt-wing principle

MILESTONES:

Development studies	1956-1963
Prototype construction begun	Nov 1963
First VTO	May 1965
First transition (VTO to conventional flight)	Jan 1966
Tri-service evaluation	1967
CL-84-1 crash during U.S. Navy flight testing	Sep 1967

WEIGHT: 12,600 LBS (VTOL max); 14,500 LBS (STOL max)

ENGINES: (2) Lycoming T53 turboshaft engines (1500 SHP each) driving two 14.0 ft. diameter four-bladed propellers through reduction gearing and a 7.0 ft. diameter counterrotating tail propeller.

LAYOUT: See Fig. 18

COMMENTS: This jointly funded venture was the culmination of seven years of V/STOL technology investigation which concluded that a tilt-wing was best for VTOL and a deflected slipstream was optimum for STOL operations. The CL-84 design, with full-span Kruger and slotted trailing edge flaps, was intended to capitalize on both features. The main propellers were interconnected by a cross-shafting system that was also connected to the tail propeller which provided lift and pitch control in powered-lift flight. In the hover, roll was controlled by varying main propeller thrust and yaw by differential deflection of the flap/ailerons. During transition from VTO to forward flight, the wing was tilted downward to the horizontal position and control was progressively transferred to conventional control surfaces. A SAS provided rate damping about all three axes and attitude hold in pitch.

Developmental testing included both powered and unpowered wind tunnel models and a 36% scale propeller and propeller/nacelle model which was utilized to define engine and propeller inlet flow and losses. The single CL-84 prototype flew 305 flights, executed 151 transitions and 346 VTOL sorties. Both NASA and Tri-Service evaluations concluded that (1) powered-lift handling qualities were good even in gusts to 35 kts., (2) hover thrust was several percentage below the predicted value owing to airframe-propeller interaction, (3) STOL performance was excellent and no ground effect instabilities were present at high wing angles.

<u>SOURCE</u>: Ref. 2-2, App. K
Ref. 2-3, '68 p.16

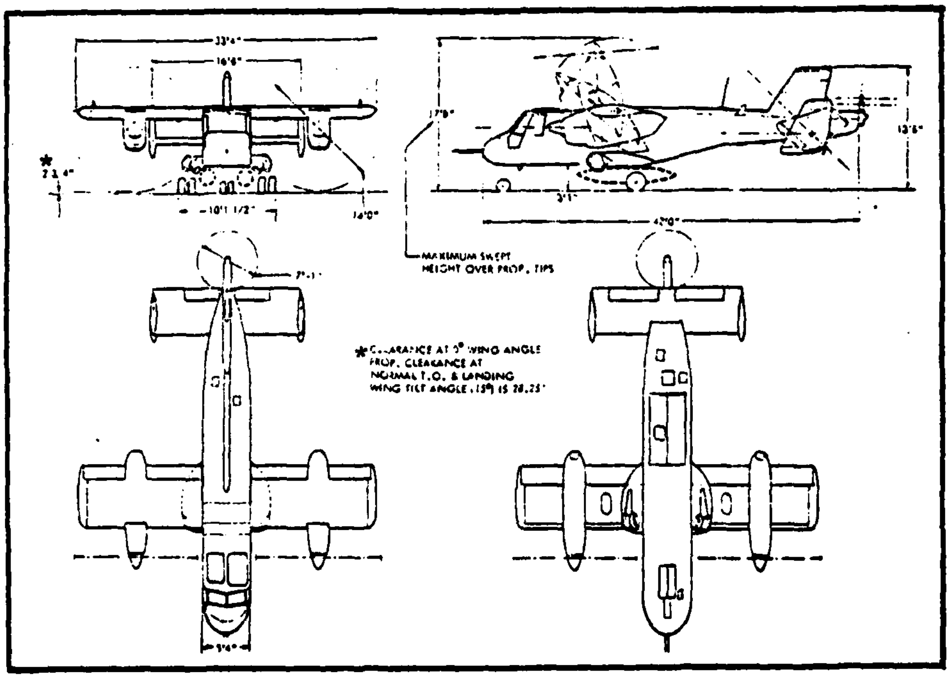


FIGURE 2-18

DESIGNATION:	VZ-3 "Vertiplane"
MANUFACTURER:	Ryan Aeronautical Company
SPONSOR:	USA, ONR
CONCEPT:	Deflected slipstream principle
MILESTONES:	Ground tests begun — Feb 1958 Full-scale wind tunnel tests at NASA Ames — Oct-Dec 1958 First CTOL flight — 1959 Aircraft damaged in crash NASA received rebuilt VZ-3 for further testing — 1961
WEIGHT:	2600 LBS
ENGINES:	(1) Lycoming T53-L-1 turboshaft engine (1000 SHP) driving two 9.0 ft. diameter three-blade wooden propellers
LAYOUT:	See Fig. 19
COMMENTS:	This single-aircraft program was funded by the Army, through an Office of Naval Research contract. Its purpose was to investigate the feasibility of a medium-speed aircraft for liaison and utility operations from rough terrain. Although it was determined that it did not have significant advantage as a VTOL vehicle, the concept was found to exceed requirements for STOL. The VZ-3 could hover OGE but not IGE, due to severe buffet from the deflected slipstream. No vertical landings were made. The aircraft's STOL performance was outstanding. Initially, the VZ-3 could not be operated at high angles of attack due to flow separation on the wing. Revised flap programming and power setting procedures were the solutions. Knowledge from this program contributed heavily to the development of the XC-142A and OV-10 aircraft.
SOURCE:	Ref. 2-2, App. O Ref. 2-1, p. 101 Ref. 2-3, '60 - 385

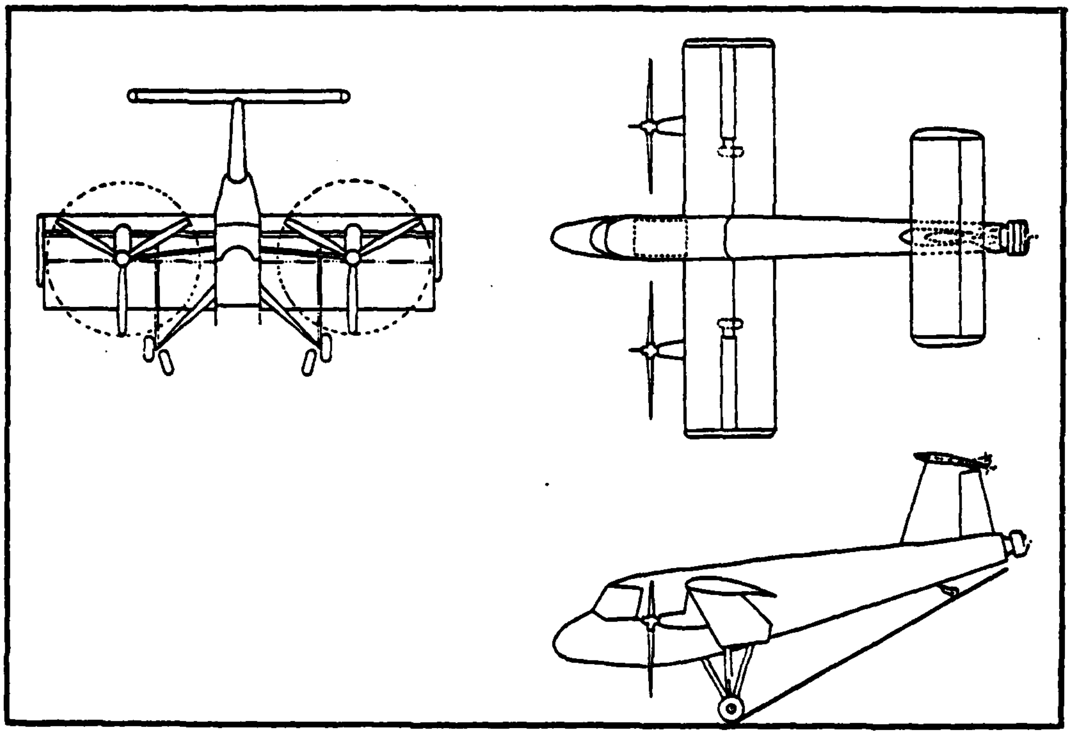


FIGURE 2-19

3. Phase - III

In the late sixties U.S. V/STOL aircraft development was deemphasized. Of significance though was the U.S. Marine Corps' acquisition of the Hawker Siddeley AV-8A "Harrier" which was deployed operationally in 1973. A year earlier Navy planners introduced the concept of the Sea Control Ship (approximately 14,000 ton displacement), essentially an ASW helicopter platform. Out of the need for a V/STOL fighter aircraft for protective cover of this ASW asset, the XFV-12A project was born.

In 1974 however, because of its restricted capabilities, the Sea Control Ship concept was scrapped and in its place the V/STOL Support Ship (VSS) was spawned. This 23,000 ton vessel was envisioned as a mission-flexible helicopter/fighter-attack V/STOL platform [Ref. 2-6].

DESIGNATION: XFV-12A

MANUFACTURER: Rockwell International

SPONSOR: U.S. Navy

CONCEPT: Thrust-augmented-wing concept

MILESTONES:

Contract award	Sep 1972
F401 engine incorporated in rotary test rig	Jan 1974
First prototype complete	Aug 1977
Tethered testing at NASA Langley	1977-1978

WEIGHT: 19,500 LBS (max VTO)
24,250 LBS (max STO)

ENGINES: (1) Pratt and Whitney F401-PW-400 turbofan (with after burning) in the 30,000 lb thrust class

LAYOUT: See Fig. 20

COMMENTS: The XFV-12A was developed to meet the need for a V/STOL fighter/attack aircraft capable of operating, without the aid of neither catapult nor arresting gear, from the flight deck of a "Sea Control Ship" of around 14,000 tons displacement. In an effort to keep development costs to a minimum, major components from existing fleet aircraft were incorporated into the design.

The XFV-12A utilizes the augmentor-wing concept in both the foreward canard and aft semi-delta wing. A diverter valve is utilized to block the F401 turbofan nozzle and the exhaust gases are ducted to augmentor nozzles in the canards and wing for V/STOL operations. A full-span ejector-flap system allows ambient air to be entrained over the flaps, mixed with the turbofan exhaust and ejected downward. A 7:1 ratio of entrained-to-primary exhaust air is required to produce the needed jet-lift. In addition to the jet thrust produced by the augmentor system, circulation about the aerodynamic surfaces due to the large mass flow of entrained air is greatly increased during STOL and outbound transition maneuvers.

No reaction controls are required for VTOL operations. Both lift and attitude control is implemented by the augmentor flap system. Height control is provided through simultaneous operation of canard and wing augmentors. Pitch control is achieved through differential operation of canard and wing ejectors, roll control by modulating right and left wing augmentors and differential deflection of wing ejector-flaps provides control in yaw. Outbound transition is accomplished, in theory, by progressively opening the turbofan nozzle diverter valve and as wing-borne flight is achieved, the ejector-flaps are closed to form a flush wing surface. Conventional elevators/flaps and rudders are employed in high-speed flight.

The forward fuselage section, nose and main landing gear assemblies from a McDonnel Douglas A4 "Skyhawk" along with the wing box and engine intakes from a McDonnel Douglas F4 "Phantom II" were incorporated into the design of this unique aircraft (see Fig. 20).

A rotary test rig, capable of achieving speeds of 150 kts., was constructed to evaluate the performance of the complete wing augmentor system. Ground tests to date have revealed the following deficiencies:

(1) higher-than-estimated duct losses,

(2) flow separation along inboard portions of the wing augmentors resulting in thrust-augmentation ratios significantly less than the target value of 1.55,

(3) difficulties in matching the cockpit control system with thrust augmentors and control surfaces.

The NASA Langley Lunar Landing (Appolo Project) gantry is being utilized for tethered testing of the aircraft's hover performance. Sporadic funding of the XFV-12A program, resulting from the yet-to-be defined role of V/STOL aircraft in the U.S. Navy, has contributed to the slow development of this aircraft.

SOURCE: Ref. 2-3: '73-420, '78-423
Ref. 2-7: pp. 48-51

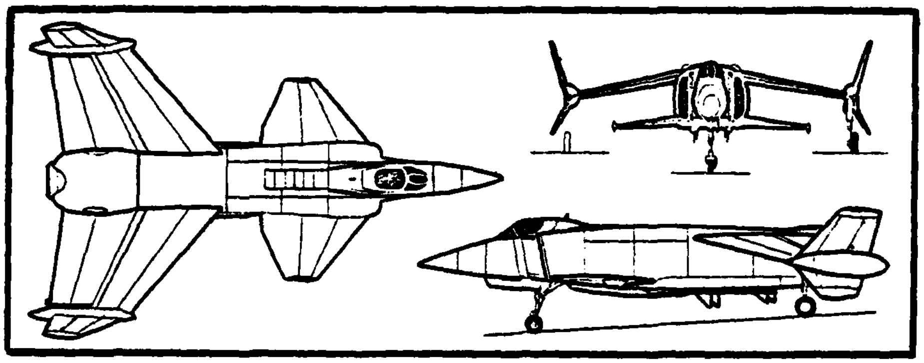


FIGURE 2-20

B. MAJOR DEVELOPMENTS IN FRANCE

Three VTOL aircraft have characterized the progress of the French aviation industry. The lifting concepts include:

1. Vertical attitude principle
   (Turbojet) - (SNECMA Coleoptere)

2. Lift/lift cruise principle
   (Dassault Balzac V-001 and Mirage III-V)

DESIGNATION: COLEOPTERE (C. 450-01)

MANUFACTURER: SNECMA

SPONSOR: Company venture

CONCEPT: Vertical attitude demonstrator

MILESTONES:

Pilotless remote control developmental work	1952-1955
First free vertical flight	May 1959
Aircraft crashed during transition from vertical to horizontal flight	Jun 1959

WEIGHT: 6500 LBS

ENGINES: (1) SNECMA ATAR 101 E.V. Turbojet of 8155 lbs thrust

LAYOUT: See Fig. 21

COMMENTS: SNECMA was engaged from 1954-1958 in testing various pilotless remote controlled and piloted vertically mounted turbojet test rigs. The COLEOPTERE was basically the "ATAR VOLANT" test vehicle fitted with an annular wing to permit transition to horizontal flight. The aircraft was controlled from a tilting ejection seat inside an enlosed cockpit. Directional control during VTOL was provided by pneumatic deflection of the primary jet. Control during conventional flight was by four equally spaced movable fins mounted around the trailing edge of the annular wing. An elaborate SAS was provided and several complete transitions were performed prior to the COLEOPTERE being lost.

SOURCE: Ref. 2-2, p. 136
Ref. 2-3, '59-141

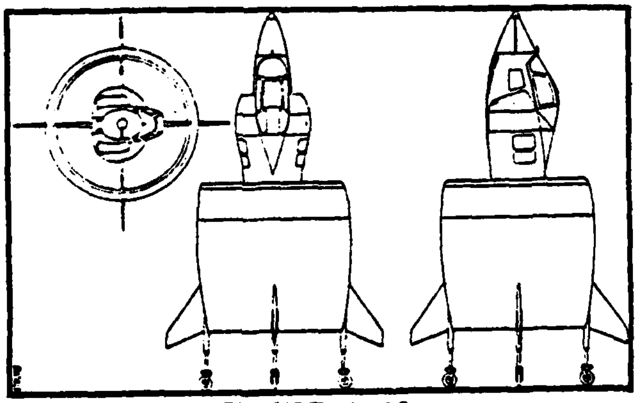


FIGURE 2-21

DESIGNATION: BALZAC V-001

MANUFACTURER: DASSAULT

SPONSOR: French Government

CONCEPT: Lift/lift cruise principle

MILESTONES:

Development begun		1960
First tethered flight	12	Oct 1962
First free VTO	18	Oct 1962
First transition from VTO to forward flight	18	Mar 1963
Crash during test flight		Jan 1964

WEIGHT: 13,500 LBS

LAYOUT: See Fig.22

ENGINES: (1) Bristol Siddeley ORPHEUS Turbojet (4850 lb thrust) plus (08) Rolls-Royce RB. 108 lift engines (2200 lb each)

COMMENTS: Under a government contract, DASSAULT modified the Mirage III prototype airframe to study the problems of vertical flight and develop a control system for a VTOL fighter aircraft. The SNECMA ATAR 9B turbojet was replaced by the smaller Bristol Siddeley cruise engine. In addition, eight vertically-mounted lift engines were installed in pairs fore and aft of each main wheel-bay. Control during VTO operations was accomplished by reaction control jets in the nose, tail and wings of the aircraft. Conventional control surfaces functioned normally following transition to forward flight.

SOURCE: Ref. 2-3: '65, p. 39

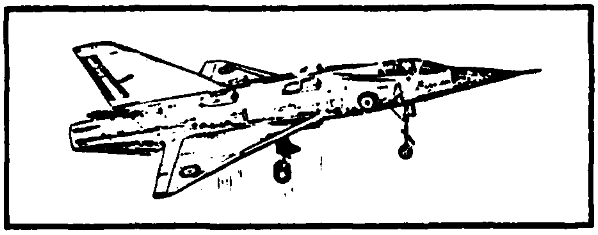


FIGURE 2-22

DESIGNATION:	MIRAGE III-V	
MANUFACTURER:	DASSAULT	
SPONSOR:	French Government	
CONCEPT:	Lift/lift cruise principle	
MILESTONES:	Initial hover tests	Feb 1965
	First transition	24 Mar 1966
	Second prototype attained Mach 2+	12 Sep 1966
	Second prototype lost in crash	28 Nov 1966
WEIGHT:	29,630 LBS	
LAYOUT:	See Fig. 23	
ENGINES:	First prototype - (1) SNECMA TF-104 turbofan (13,890 lbs) later replaced by (1) TF-106 Turbofan (16,755 lbs), plus (8) Rolls-Royce R.B. 162-1 turbojet lift engines (3525 lbs each) Second prototype - (1) P & W TF-30 turbofan (18,500 lbs after burning) plus the eight R.B. 162-1 lift engines.	
COMMENTS:	A follow-on to the BALZAC V-001 begun earlier, this VTOL strike fighter closely resembled the Mirage III-E except for the lengthened fuselage. The uprated TF-106 cruise engine was installed in the first prototype to permit expansion of the flight envelope to MACH 1.35 and transition from horizontal to vertical flight. As with the V-001, a reaction control system was employed in vertical flight. The second prototype was fitted with sideways-opening intake doors over the lift jets. Following the loss of this prototype, production plans were cancelled.	
SOURCE:	Ref. 2-3, '67, p.41	

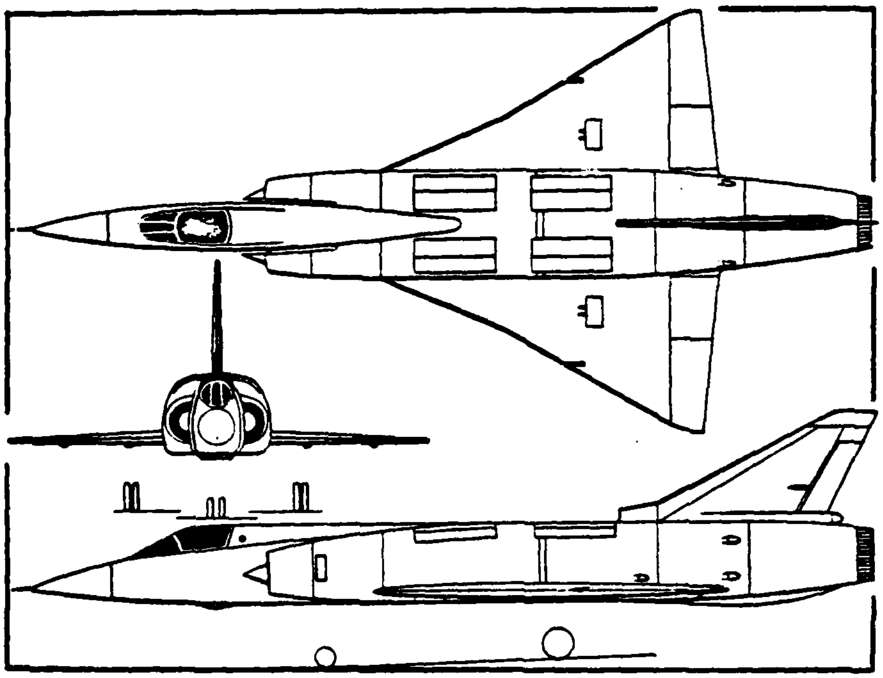


FIGURE 2-23

C. MAJOR DEVELOPMENTS IN ENGLAND

Progress in the VTOL field has been highlighted by the two designs discussed below. Lifting concepts include:

1. Lift/lift cruise principle - (Short SC.1)

2. Vectored-thrust (jet) principle - (Hawker Siddeley p. 1127/Harrier)

DESIGNATION: SC. 1

MANUFACTURER: Short Brothers and Harland Ltd.

SPONSOR: British Government

CONCEPT: Lift/lift cruise principle

MILESTONES:

Construction begun	1954
First conventional flight	Apr 1957
First free VTO	Oct 1958
First complete transition	06 Apr 1960
Crash of second prototype	Oct 1963
Flight tests resumed	Jun 1966

WEIGHT: 8050 LBS (max VTO)

ENGINES: (05) Rolls-Royce RB. 108 turbojet lift engines (2200 lbs each) located in the fuselage; four engines mounted vertically and the remainder horizontal

LAYOUT: See Fig. 24

COMMENTS: The first fixed-wing VTOL airplane built in the UK, the SC. 1 was developed for research in the vertical take-off and landing area. The aircraft was built around the propulsion system, the RB. 108 being selected for its availability and high thrust-to-engine wt. ratio (8.7:1). The four "lift" engines incorporated vectoring nozzles (25 rearward/12 forward) to aid in precise hover and transition flight path control. Eleven percent of the total airflow of the five RB. 108 engines was bled to a common duct to supply wingtip, nose and tail reaction control nozzles (RCS) for control during VTO and transition phases. Conventional aerodynamic surfaces provided control during normal airplane flight.

An elaborate double redundant SAS was developed for the SC. 1. The longitudinal and lateral axes could be operated in one of three different modes: (1) unstabilized - control position directly proportional to control stick position, (2) rate - rate damping was provided and (3) leaky attitude - a quasi-attitude signal was added to give a short-term response similar to attitude control. The yaw axis was unstabilized.

A specifically configured Gloster Meteor aircraft was used as a flying testbed to develop lift engine installation principles. One hundred thirty-five hours of testing was accomplished while investigating hot gas recirculation, intake design features and rolling take-off techniques. After initial conventional flights, the SC. 1 began hover investigations on an elevated platform, tethered in a gantry which confined excursions to a ten foot cube. Free hovers followed and the low-speed transition envelope was expanded. The aircraft was limited to one complete transion circuit (VTO-transition to forward conventional flight - transition and vertical landing) by fuel considerations (16 min.). After extensive flight testing in the hover and transition phases, the following conclusions were drawn: (1) handling of the SC. 1 in the hover was good. Longitudinal and lateral rate damping was adequate but directional control needed improvement, (2) strong dihedral effect in powered lift translations led to roll divergency and (3) pilot workload was excessive during a transition to landing. Lift engine start-up, changes in trim and general deterioration of handling qualities demanded too much of the pilot during this critical flight regime.

Following the crash in 1963, the second prototype was rebuilt with inertial rate damping added in the yaw axis and the sideslip problem during transition was eliminated by incorporation of a lateral accelerometer. Propulsive-induced effects experienced included a nine percent loss in installed thrust during VTO from a runway but only a six percent loss when operating from a gridded, elevated surface. The main problem was engine intake temperature distortion when in ground effect. At about two ft. wheel height, induced flow and ground influences provided for maximum suckdown, estimated to be a loss of ten percent over a solid runway surface. The SC. 1 continues to be used in VTOL research at the RAE Bedford.

<u>SOURCE</u>: Ref. 2-1, p. 147
Ref. 2-2, App. - E
Ref. 2-3, '61-188

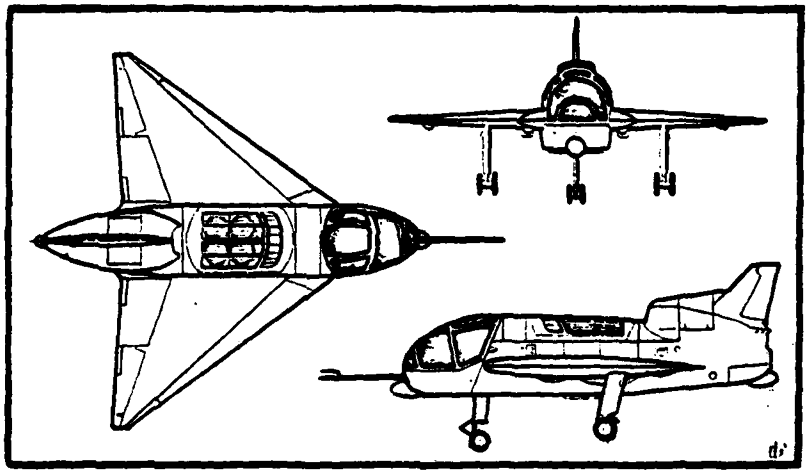


FIGURE 2-24

DESIGNATION:	P. 1127, Kestrel (XV-6A), AV-8A (Harrier)	
MANUFACTURER:	Hawker Siddeley Aviation Limited (formerly Hawker Aircraft Limited)	
SPONSOR:	British Ministry of Aviation	
CONCEPT:	Vectored-thrust (jet) principle	
MILESTONES:	Design and construction of P. 1127 prototype	1957-1960
	British Ministry order of six aircraft (P. 1127)	1960
	Tethered hovering tests begun	Oct 1960
	First free hover	19 Nov 1960
	First complete transitions	12 Sep 1961
	Tripartite order of nine Kestrel's	Jan 1962
	P. 1127 shipboard demo	Feb 1963
	Tripartite flight evaluation (Kestrel)	Apr-Sep 1965
	Harrier GR Mkl production began for RAF	1967
	U.S. Navy NPE of Harrier (AV-8A)	Jan 1969
	Twelve-AV8A's ordered for USMC	Mar 1969
	Harrier operational with RAF	Apr 1969
	First AV-8A delivered to USMC	Jan 1971
	McDonnel-Douglas obtain license to manufacture future Harriers	1971
WEIGHT:	P. 1127 (max VTO) 12,400 LBS Kestrel (max VTO) 15,000 LBS AV-8A (max VTO) 19,000-21,000 LBS	

ENGINES:	P. 1127:	(1) Bristol Siddeley Pegasus 3 turbofan of 13,500 LBS thrust.
	Kestrel:	(1) Pegasus 5 turbofan of 15,200 LBS thrust
	AV-8A:	(1) Pegasus 10 (20,000) or (1) (1) Pegasus 11 (21,500 LBS)
LAYOUT:	See Fig. 25	

COMMENTS:

The P. 1127 was a transonic single-engine V/STOL prototype strike/recon. aircraft. The basic design concept stemmed from a French proposal submitted to the NATO Mutual Weapons Devlopment Program (MWDP) and was discussed with Bristol Aero Engines, Ltd. Hawker Aircraft Limited suggested two major changes to the proposed power plant, the B.E. 53 (later "Pegasus") turbofan: (1) split the jet exhaust and incorporate rotable cascade type nozzles and (2) use countrarotating turbine spools to minimize gyroscopic coupling. Hawker Aircraft began work on the aircraft in 1959 as a company-funded venture while the engine development was funded by the MWDP and Bristol Aero. However, early in the P.1127 development, the British Ministry of Aviation provided funding for two prototypes. The layout of the aircraft is shown in Fig. 2-25.

The Pegasus engine was a straight flow, turbofan with mechanically independent LP and HP compressors which rotated in opposite directions. A portion of the LP fan air was directed to a plenum and exhausted through port and starboard nozzles at the rear of the engine. The four nozzles, being mechanically interconnected by shafts and chains, could be rotated by an air motor to vector the thrust at any intermediate position between horizontally aft (3 deg.) and downward (92 deg.). A single cockpit lever controlled nozzle position, with speed of rotation directly proportional to selector level movement. A unique feature was incorporated to reduce the thrust loss due to reduced inlet efficiency in the low-speed flight regime. Inflattable rubber bags formed variable geometry inlet lips. For hover and low velocity flight a large radius bellmouth lip was created, while for high speed the lips were deflated to give a sharp intake profile.

Except for the nozzle position lever, all cockpit flight controls were similar to a conventional tactical jet airplane and performed the normal function in all flight regimes. In hover/transition modes, jet reaction controls were provided as aerodynamic controls ceased to be effective. When the thrust nozzles were rotated downward, HP bleed air was directed through an interconnected valve to the reaction control ports. With the stick and rudder bar centered, the reaction control valve ports were closed. Longitudinal control was achieved through mechanical linkage to a hydraulically actuated tandem jack unit powering a single-piece unit horizontal tail (UHT). The tail reaction control nozzle (RCN) was connected to the tailplane while the nose RCN was directly connected to the control stick. Lateral control consisted of mechanical linkage to a hydraulically actuated tandem jack unit powering conventional ailerons. A RCN was located in each wingtip and connected to the aileron linkage. The rudder was not power-assisted and the yaw RCN's were connected to it. Hydraulically operated, two-position, trailing edge flaps were fitted and incorporated a blow-up feature which reduced deflection angle as air loads increase at forward speeds. A limited authority single-channel autostabilizer was installed to augment pitch and roll control in powered lift flight.

In January 1962 a tripartite, consisting of the United Kingdom, United States and Federal Republic of Germany, ordered nine developed P. 1127 aircraft (known as Kestrel FGA Mk1 in Britain and the XV-6A in the U.S.) to assess the practicality of jet V/STOL operations. The major engineering changes which differentiated the Kestrel/XV-6A from the earlier P. 1127 included (see Fig. 2-25): (1) revised wing planform and incorporation of upper-surface vortex generators, (2) increased span UHT with marked anhedral, (3) nine inch extension of forward fuselage, (4) fixed metal intakes, (5) increased lateral reaction control power, (6) installation of Pegasus 5 engine, and (7) the autostabilization system was deleted.

The Harrier GR Mk1, designated AV-8A in the U.S., is the production attack aircraft which evolved from the P. 1127/Kestrel development. Even though it employed the same basic concepts of lift, propulsion and control, only five percent of the engineering drawings carried over into the Harrier design. Figure 2-25 shows the layout of the AV-8A.

The Pegasus 11 turbofan incorporates a three-stage LP fan and an eight-stage HP compressor which are driven by independent two-stage turbines. Different engine ratings are provided for conventional and powered-lift flight and a water-injection system allows RPM to be increased for a given turbine inlet temperature to permit higher short lift thrust ratings. The engine is also equipped a jet pipe temperature limiter, a HP RPM limiter and a pressure ratio limiter. The AV-8A flight control system incorporates artificial spring feel units in the lateral and longitudinal axes, with a dynamic pressure Q-feel unit and bob-weight on the latter axis. Rudder feel is primarily aerodynamic with a centering spring fitted for low speed flight. A single-channel three-axis SAS is provided which utilizes rate damping to improve V/STOL handling qualities and the system can only be engaged below 250. KIAS with the landing gear locked down.

The major wind tunnel work was done by RAE Farnborough and the contractor primarily to investigate the conventional characteristics of the aircraft. During the early development period, NASA Langley built and tested a 1/6 scale free-flight model of the P. 1127 to verify the design. Powered models were employed to investigate hot gas ingestion and ground effects. Initial flight test efforts concentrated on the hover phase and a tether test-rig was utilized. The aircraft was operated over a grate-covered pit and tethering cables were attached to the wing-tips and nose gear. A month of tethered tests were carried out prior to the first free hover. Envelope expansion moved from hover to taxi tests to conventional flight. Both roll and yaw control power were increased several times during the course of early flight tests and similarly the control sensitivity of both axes was successively improved. The P. 1127 flight test program concluded that: (1) A thrust-to-weight margin of between five to ten percent was required, depending on the maneuvering task, (2) unstabilized hovering flight was easier than on many contemporary helos, (3) no significant cross-coupling effects existed, (4) STOL operations substantially improved payload capabilities with short distance requirements.

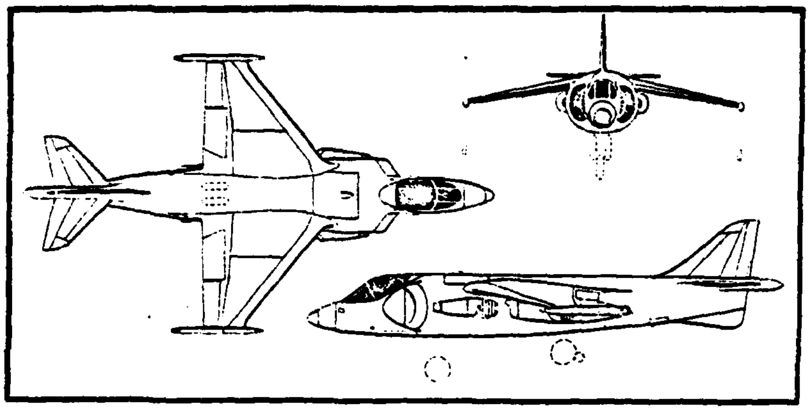
In 1964 the Kestrel became the first jet V/STOL aircraft to be granted a service release by the British military. The service trials concluded that: (1) control in powered-lift flight presented few difficulties (autostabilization not fitted), (2) handling qualities for the conventional ground attack mission where good and (3) engine operation above 15,000 ft. was unsatisfactory. The Tripartite evaluation trials consisted of a tactical assessment which studied the modes of deployment for jet lift V/STOLS. At the completion of the 951 flight evaluation, the conclusion was drawn that a visual mission capability without SAS was possible but that full mission capability could only be recognized with stability augmentation. In late 1965, six Kestrels (XV-6A) came to the U.S. for further operational trials with the Army, Navy and Air Force. These evaluations were followed by investigations of terminal area flight procedures and in-flight thrust-vectoring by NASA Langley and further handling quality work at Edwards Air Force Base.

Initial Harrier (AV-8A) tests stressed optimizing the lifting performance of the wing and then its stores carrying capabilities. Additional development was done on the intake design to achieve a satisfactory compromise between high and low speed flight performance. Blow-in doors and boundary layer bleed ports were incorporated. The Navy's NPE in 1969 listed as major deficiencies: (1) poor operation of the water injection system and (2) lack of directional stability in low-speed powered-lift flight. Once delivered to the USMC, the latter problem was made tolerable by incorporation of a yaw autostabilizer, rudder-pedal shaker and display of sideslip information on the pilot's heads-up display.

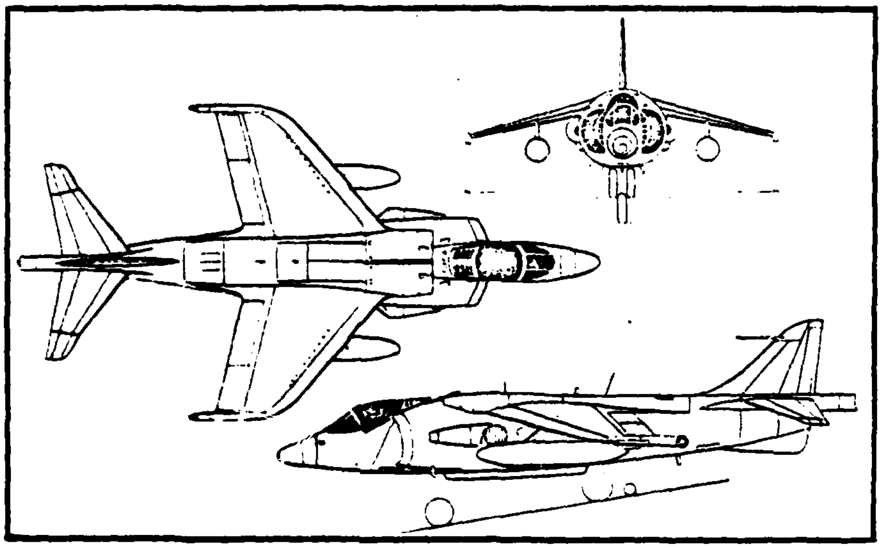
Propulsion-induced effects on the P. 1127/AV-8A design were considered early in the development period. Both test rig and wind tunnel model work was completed and a good correlation was shown with later flight test data. During a VTO, near-field recirculation is the primary cause of hot gas ingestion for wind conditions of under forty knots. During prototype design, Hawker Siddeley predicted a positive ground effect from the four-nozzle arrangement. However, later model tests showed negative pressures in ground proximity. The addition of longitudinal strakes on the lower fuselage reduced the ground height lift loss to zero.

In a decelerating transition to hover, deflection of the jets produces a change in downwash at the UHT. Both static longitudinal and directional stability is reduced. A a result, more attention must be paid to maintaining the desired angle of attack and zero sideslip in the midtransition speed range.

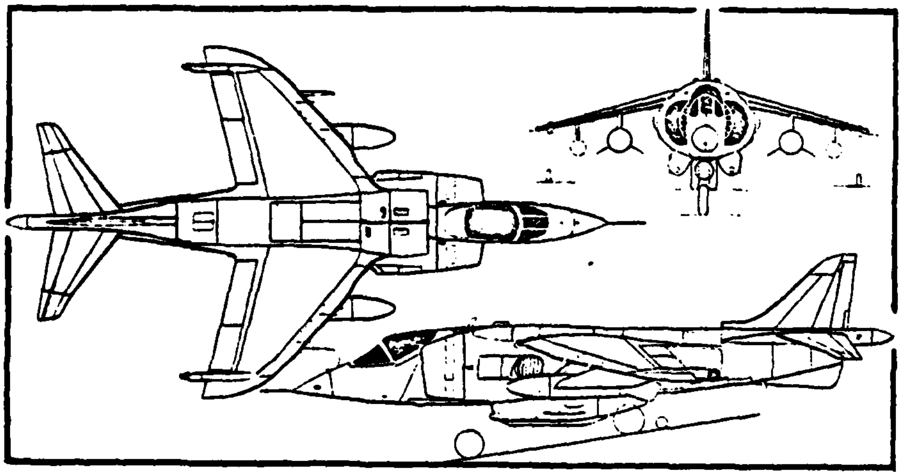
<u>SOURCE</u>: Ref. 2-1, P - 142
Ref. 2-2, App. -H
Ref. 2-3, '61-174, '65-156; '77-186



(P. 1127)



(KESTREL)



(AV-8A)

FIGURE 2-25

D. MAJOR DEVELOPMENTS IN GERMANY

Along with France and England, the Federal Republic of Germany has pursued VTOL development. Three designs, which all utilized the lift/vectored-thrust (jet) principle, will be analyzed.

1. EWR SÜD VJ-101C
2. Dornier DO-31
3. VFW-FOKKER VAK - 191 B

DESIGNATION: VJ-101C

MANUFACTURER: EWR SUD

SPONSOR: Federal German Defense Ministry

CONCEPT: Lift/Vectored-thrust (jet) principle

MILESTONES:

EWR consortium formed	1959
Hovering rig-first free flight	Mar 1962
Partial tethered stabilization tests	Feb 1963
First free hover	10 Apr 1963
First CTOL	Aug 1963
First transition	20 Sep 1963
Crash of prototype #1	Sep 1964
First flight prototype #2	Jun 1965
Program cancelled	1966

WEIGHT: X-1 model - 13,250 LBS (max VTO)
X-2 model - 17,635 LBS (max VTO)

ENGINES: X-1 model - (06) Rolls-Royce RB. 145 turbojets of 2750 LBS thrust each

X-2 model - (06) RB. 145 turbojets; the four wingtip podded engines with after burning (3650 LBS each)

LAYOUT: See Fig. 26

COMMENTS: In 1959 Bölkow, Heinkel and Messerschmitt Companies combined design teams to undertake the task of designing a Mach 2 VTOL jet interceptor. Five years later Heinkel withdrew from the consortium.

The VJ-101C was powered by six RB. 145 lift engines, two of which were mounted vertically (lift only) behind the cockpit and two each in swiveling wing-tip pods (lift/lift cruise). The unique swiveling pod design was selected to eliminate deflection losses and improve performance during transition by utilizing a rotating thrust vector. Figure 26 shows the aircraft's layout. The pods were rotated hydraulically and could be swivelled through an arc of ninety-four degrees. In order to achieve a supersonic pod inlet that would still give full thrust for VTO operations, a sliding shroud auxilary inlet was selectable by translating the entire pod intake forward. The slit formed supplied up to sixty percent of the required engine airflow during transiton and hover. The Rolls-Royce RB. 145 was developed from the RB. 108 lift engine expressly for the VJ-101C project. Besides providing precise and rapid thrust response, a fully modulating after-burning control system was developed for the RB. 145 to allow full operation over almost the entire engine speed range.

Since the three pairs of engines were set in a triangular pattern (see Fig. 26), attitude control during hover was implemented through thrust modulation. Pitch control consisted of differential thrust variation between fuselage and pod-mounted engines. Roll control was obtained by differential thrust variation between podded engines only. Directional control was achieved by swiveling the podded engine pairs in opposite directions. Through a common throttle lever,height control in a hover was controlled by simultaneously modulating the thrust of all six engines. A conventional control stick and rudder pedals, in addition to the single throttle lever, were used throughout the flight envelope and conventional aerodynamic control surfaces operated continuously in the vertical flight mode. A three-axis SAS provided attitude control in pitch and roll while rate control was provided in yaw.

Wind tunnel testing consisted of sixty-six hours of full-scale inlet investigation and powered models were utilized to study the influence of wind speed and height-above-ground on inlet temperature rise. Test rigs were used extensively during the VJ-101 development. A single engine "Wippe" rig demonstrated that the thrust modulation concept could be integrated into an aircraft control system. A hover rig consisting of three RB. 108 lift engines in a triangular configuration was constructed to continue the thrust modulation study and it was also utilized for pilot familiarization.

Prior to the commencement of flight testing, the VJ-101C was tested on a telescope test stand. As a result, once free hover testing began only minor modifications to the aircraft and SAS were required. Significant conslusions drawn early in the testing included (1) exhaust damage to the runway was a major problem and the X-2 model (with after-burning) was restricted from a true vertical take-off. Lift/cruise engines were started with the pods horizontal and then rotated to the 75° position before brake release/ after-burner selection. This resulted in a take-off distance of approximately ten to thirteen feet, (2) due to engine positioning and large available control power, no significant rolling moments resulted from a sideslip condition during transition, (3) improper scheduling of the auxillary pod inlets caused rapid engine surges, (4) unsynchronized accelerations of two opposite engines induced high instantaneous moments which precluded take-off.

Propulsion-induced effects were headed by a mean inlet temperature rise of 10° to 15°C. A "rolling" VTO (RVTO) technique was developed to avoid high exhaust temperature ingestion from far field. For the X-2 model the previously mentioned inlet temperature conditions resulted in a RVTO roll distance of from one hundred to two hundred fifty feet. The suckdown of the VJ-101C was about two percent at ground height, decreasing rapidly with height until, at h/d = 8, a maximum net buoyancy force of four percent was experienced.

SOURCE: Ref. 2-2, App. - G
Ref. 2-3, '65-73

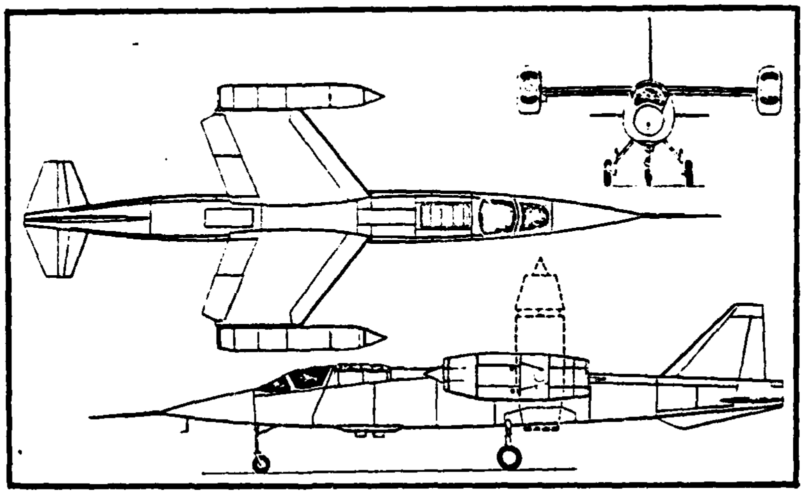


FIGURE 2-26

DESIGNATION: DO-31

MANUFACTURER: Dornier-Werke GMBH

SPONSOR: Federal German Defense Ministry

CONCEPT: Lift/Vectored-thrust (jet) principle

MILESTONES:

	Go - Ahead	Feb 1962
	First flight, small hover rig	Apr 1964
	First flight, large hover rig	Feb 1967
	First flight model - E1	Feb 1967
	First complete transition	Dec 1967
	Paris Airshow display	Jun 1969
	NASA evaluation	Apr 1970
	Program termination	Apr 1970

WEIGHT: 60,500 LBS (max t/o)

ENGINES: (2) Rolls-Royce Bristol Pegasus 5-2 vectored-thrust turbofans (15,500 LBS each) in pods mounted underwing plus (8) Rolls-Royce RB. 162-4D turbojet lift engines (4400 LBS each) mounted vertically four-per-pod on each wing tip.

LAYOUT: See Fig. 27

COMMENTS: The DO-31 was developed to meet a Defense Ministry requirement for a VTOL transport aircraft. Both the Pegasus 5 lift-cruise turbofans and the RB. 162 lift engines were chosen on the basis of their proven performance in previous VTOL programs; the former with the Hawker Siddeley P. 1127/Kestrel and the latter in the Dassault Mirage III-V. Each Pegasus engine was equipped with four swiveling exhaust nozzles that could be rotated from 30 degrees forward to 80 degrees aft of vertical. A maximum of eighteen percent of the HP compressor flow could be bled to meet reaction control system (RCS) demands. The RB. 162 turbojets were fitted with spherical, swiveling nozzles which could be deflected $\pm$ 15 degrees fore and aft of the engine axis. All eight lift engines could be started simultaneously by turbine air impingement from either Pegasus bleed or an external ground source. During airborne starting, positive inlet pressure conditions reduced bleed air demands and stable idle RPM could be achieved within 15 secs. of cycle initiation. A considerable effort was expended on lift engine inlet design, with several types of flow-turning devices being tried prior to the final selection of a cascade type of inlet system. As a result of wind tunnel investigations, the RB. 162 engines were mounted with a 15 degree forward incline, resulting in improved inlet performance.

Control in conventional flight was provided by a rudder, elevators and ailerons. The control stick and rudder pedals were used in all phases of flight as the RCS and aerodynamic control surfaces were integrated. In the powered-lift flight phase pitch attitude was controlled by varying bleed air to a tail mounted RCS nozzle, roll control was implemented by differential thrust variation of the lift engines and directional control resulted from swiveling of the lift engine nozzles. All aerodynamic control surfaces were irreversible and a variable Q-feel system was installed in pitch to prevent overstress at the high speed end speed of the envelope. Altitude control during hover could be effected by varying either main or lift engine thrust. An automatic flight control system (AFCS) was employed in the VTOL regime <u>only</u> and consisted of the following modes: pitch and roll attitude commands, roll rate damping, preselected pitch attitude trim and sink rate command at touchdown.

Approximately 4500 hours of wind tunnel work was performed using both complete aircraft and component models. Two major hover rigs were developed, the first consisting of a cruciform shaped tubular framework on which four RB. 108 lift engines were mounted. This device was tested on the ground, on a test pedestal and in free hover. It provided significant design input for the RCS and SAS systems and accumulated over 255 hours of engine time. A follow-on larger rig consisted of a full-scale DO-31 wing and engine nacelles, however using only three lift engines per wing tip pod. A tubular fabric-covered forward fuselage section was utilized along with the actual hover control, undercarriage and fuel systems of the full-scale aircraft. Over 345 ground, pedestal and free hover tests were completed to validate the hovering stability and control system, powerplant installations and to investigate hot gas ingestion and ground erosion problems.

Following flight testing in the CTOL region (lift engines uninstalled), exploration of the powered-lift region proceeded from hover to outground transiton. Hot gas reingestion was encountered in no-wind VTO's but a rolling take-off technique, during which main engine nozzles were held in the aft position until approximately 60 feet of forward roll was completed and then the nozzles were aligned with the lift engine axis, resulted in smooth main engine operation and take-off velocities of 15 to 25 kts. Approximately eighty percent of the maximum pitch control power was required to trim the DO-31 at midtransition due to jet interference effects and greater-than-estimated trim movements. However, no significant rolling moments were present in sideslip conditions, due in part to the action of the lift engine pods as wing end-plates. The inbound transition (conventional flight-to-vertical landing) maneuvers received the major emphasis during the flight test program. The resulting technique began with lift engine starting while in stable horizontal flight, descent initiation by a combination of changes in pitch attitude/lift engine thrust/main engine nozzle position, a flare executed with pitch attitude and finally vertical speed control by lift engine thrust modulation. NASA evaluated the DO-31 to investigate its performance and handling qualities in relation to the operation of a commercial VTOL transport in the terminal area. During simulated IFR approaches on glideslopes as steep as twelve degrees, although stability and control of the aircraft was acceptable, recirculation effects in a vertical landing were significant below an altitude of 50 feet.

Contractor testing revealed that hot gas ingestion to the Pegasus engines was a major concern. During the landing approach (zero-wind conditions), reingestion resulted in loss of a wave-off capability below a 15 feet wheel height. The situation improved in a headwind (critical altitude being reduced to 3 feet in a thirty knot wind), degraded for tailwinds and was unaffected by crosswinds. The rolling take-off technique noted earlier, along with stopping the lift engines upon touchdown and keeping the Pegasus nozzles aft for vertical landings, minimized such problems during Dornier's flight test program. Propulsive induced forces on the aircraft were significant in both hover and transition modes. During the former, jet-induced lift loss increased from three percent OGE to approximately eight percent at touchdown. Although a nose-down pitching moment change and negative lateral static stability were present below a 3 feet wheel height, the attitude SAS was able to adequately compensate. During OGE transitions, the lift loss increased to 11 percent at 80 kts. after which it gradually decreased with increasing velocity. The jet-induced pitching moment also increased with velocity, achieving a maximum at about 115 kts. Dornier claimed though that aerodynamic lift compensated for any such lift loss encountered during transition to forward flight.

SOURCE: Ref. 2-2, App. D
Ref. 2-3, '65-70, '70-96

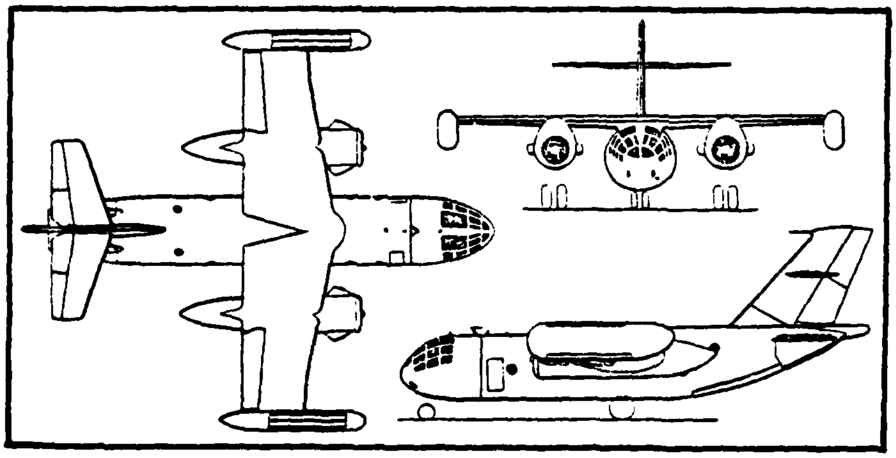


FIGURE 2-27

DESIGNATION: VAK-191B

MANUFACTURER: VFW-Fokker

SPONSOR: Federal German Defense Ministry

CONCEPT: Lift/Vectored-thrust (jet) principle

MILESTONES:

German-Italian development initiated	1964
First flight hover rig	1966
Italian Government withdrawal	1968
Tethered hover tests	1970
First flight (CTOL)	10 Sep 1971
First transition	26 Oct 1972
U.S. Navy-FRG Joint Flight Test Program	1974

WEIGHT: 19,840 LBS (max)

ENGINES: (1) Rolls-Royce RB. 193-12 vectored-thrust turbofan (10,150 lbs thrust) and (2) Rolls-Royce RB. 162-81 turbojet lift engines (5577 lbs each) mounted vertically in the fuselage

LAYOUT: See Fig. 28

COMMENTS: In 1964 the German Defense Ministry issued a requirement for a subsonic VTOL tactical reconnaissance fighter to replace the Fiat G. 91. A design study was submitted by the former Focke-Wulf company and development of this project was initiated jointly by VFW and Fiat of Italy. Four years later the Italian Government withdrew financial support but the project was continued by VFW and Fokker of the Netherlands.

The VAK-191B was powered by three fuselage-mounted engines. The Rolls-Royce RB. 193-12 turbofan, fitted with four swiveling nozzles, was developed specifically for this project. The resulting powerplant never met the predicted compressor/turbine efficiencies though and an overall thrust deficiency resulted.

The RB. 162-81 lift engines were growth-engines which Rolls-Royce developed from their RB. 162 experience in the Dassault Mirage III-V and Dornier DO-31 VTOL programs. These two engines were arranged symmetrically fore and aft of the main lift/cruise powerplant and canted forward 12 1/2 degrees from the vertical to improve intake performance. The lift engine intakes were tailored to the hover/transition modes and the hydraulic powered inlet doors opened laterally. Both two and three-dimensional inlet configurations were tested in the wind tunnel and flow effects from the wing were found to have a negligible effect on inlet performance.

A pseudo fly-by-wire flight control system with mechanical backup was provided. Conventional aerodynamic control surfaces were employed for CTOL operations while a reaction control system (RCS) was installed for use in powered lift flight regimes. Reaction nozzles were located in the nose, tail and outer wing panels and bleed air was supplied by all three engines. The triplex fly-by-wire control system switched automatically from an attitude stabilization/command mode in hover to an attitude stabilization/rate command mode at about 50 kts. and then reverted to rate damping at about 140 kts. The loss of one lift engine resulted in automatic shutdown of the other engine to preclude extreme attitudes outside the ejection seat envelope. A roll control limiter was also incorporated to avoid excess sideslip angles during transition.

Because of the high level of advanced technology present in the VAK-191B design, a comprehensive test and research program was undertaken which included: 7850. hrs. of sub/transonic wind tunnel testing, 2350 hrs. of functional and reliability testing of subsystems, over 900 hours of fixed-based simulator work with hybrid computers and static structural testing of the complete airframe. A hover rig, consisting of an open tubular framework structure on which five RB. 108 lift engines installed, was used as a testbed for ground, pedestal and free hover tests. Valuable data was provided to aid in the development of the RCS and automatic flight control (AFC) systems and to study the ground flow footprint and ground effect characteristics of the full-scale aircraft.

Flight testing began with the aircraft tethered and then proceeded to free hover and low speed transitions to 80 kts. Initially, a 12 percent reduction in forward lift engine thrust was used to correct for a nose-up pitch (due to increased downwash at the horizontal tail caused by propulsive flow induced effects) experienced when accelerating to 140 kts. However, this loss of available thrust seriously hindered the VAK-191B's acceleration capability, so an alternate solution of changing the tail incidence angle by four degrees was implemented. Early in the flight testing it was determined that longitudinal and lateral response and damping characteristics were unsatisfactory for the maneuvering precision required in low-speed powered lift flight. The AFC system was reoptimized and subsequent testing demonstrated that the noted deficient handling qualities were corrected and that the aircraft could be positioned very accurately. Hover precision was only degraded, by upsets in roll, during prolonged hovering at a wheel height of less than 3 ft. caused by jet efflux. For VTO, a thrust-to-weight ratio of 1.07 was determined to be the minimum for a safe lift off and transit through ground effect within a reasonable time. Conventional take-off without the use of lift engine thrust proved to be impossible for VAK-191B. An angle of attack limitation of 16 degrees was established to preclude striking the runway with the tail during rotation. Although sufficient elevator power existed to provide and control the required pitching moment, the additional considerations of inadequate thrust to overcome drag (main engine power only), poor Dutch Roll damping at high angles of attack, rapid deterioration of the directional stability of the basic airframe and the aft location of the rear bicycle-type main gear (dictated by a nuclear-weapons capability) restricted the aircraft to a three engine operation for CTOL.

During VTO tests, hot gas ingestion (HGI) into engine inlets was shown to be a function of main engine nozzle angles. On short take-off runs, between 30 and 40 kts., lift engine exhausts were ingested by the RB. 193-12 if its nozzles were rotated more than 70 degrees from the horizontal. Exhaust gas fountain impingement on the fuselage produced a noticeable cushion effect starting at a gear height of 10 feet during vertical descents. Out of ground effect, an additional 2 percent excess thrust was required to maintain hover due to flow-induced suckdown forces.

In December 1972, the Federal Republic of Germany cut funding to the project and further development of the VAK-191B was terminated.

SOURCE: Ref. 2-2, App. F
Ref. 2-3, '65-77, '73-97

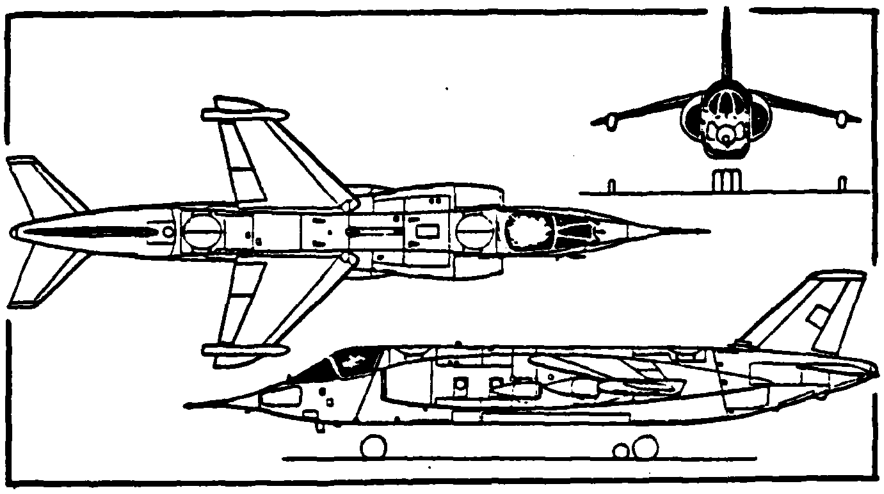


FIGURE 2-28

III. POWERED HIGH LIFT SYSTEMS

A. INTRODUCTION

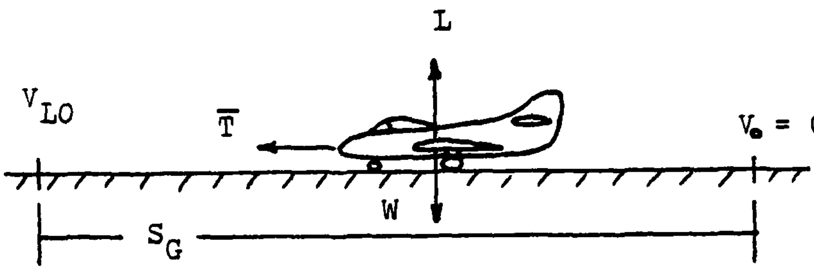


Figure 3-1

In order to analyze the take-off roll of an aircraft, shown in Figure 3-1, from simple first-order considerations, the terminal energy of the vehicle at lift-off can be expressed as:

$$\overline{T} \cdot S_G \cong 1/2 \left(\frac{W}{g}\right) V_{LO}^2 ,$$

where $\overline{T}$ is the mean accelerating thrust (approximately 20% of propulsive energy available is dissipated by runway friction and aerodynamic drag rise) during the ground-run distance S_G [Ref. 3-1].

The lift-off velocity can be written

$$V_{LO} = \sqrt{\frac{2(W/S)}{C_{L(LO)} \cdot \rho}}$$

Substituting this result into the terminal energy expression and solving for S_G yields:

$$S_G \cong \left[\left(\frac{W}{S}\right)\left(\frac{W}{T}\right) \frac{1}{C_{L_{(LO)}}} \left(\frac{1}{\rho \cdot g}\right) \right]$$

For an aircraft of a fixed weight and size operating at a specific altitude, the ground-run can be minimized by achieving a high lift coefficient at lift-off.

Using a definition of $V_{LO} = 1.20\ V_{Stall}$, Table 3-1 emphasizes the reduction of stall speed by increasing the coefficient of lift. These results are from a Boeing test program utilizing a modified 707 prototype (367-80) fitted with various high lift devices (HLD) of increasing sophistication [Ref. 3-2]. A thirty-five percent reduction in stall speed resulted from an increase by a factor of about two in maximum lift coefficient.

High Lift System	V_S (kts)	$C_{L(max)Trim}$
Basic double-slotted T.E. flap	104.	$\approx$ 1.50
Additional KRUEGER L.E. flap (unblown)	95.	
Triple-slotted T.E. flap	88.	
Additional "blown" KRUEGER L.W. flap	83.	
Experimental "blown" T.E. flap	68.	$\approx$ 3.00

TABLE 3-1

B. HIGH LIFT DEVICES

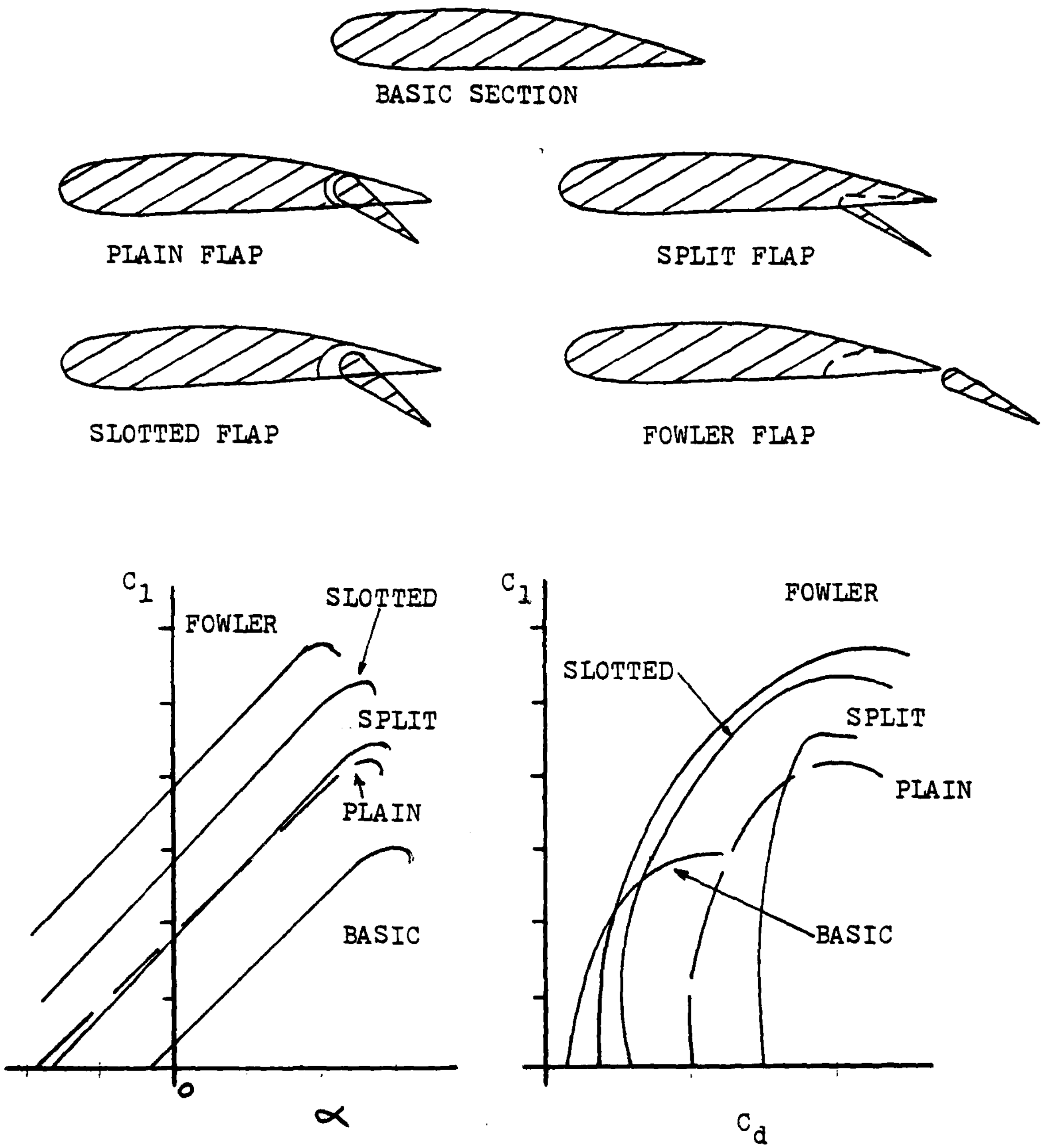
To increase the maximum coefficient of lift of a particular wing configuration two factors must be considered. First the circulation must be increased and secondly airflow separation, due to the formation of an adverse pressure gradient over the upper airfoil surface, must be delayed as long as possible.

High lift devices can be either unpowered (mechanical) or powered. Examples of the former include trailing edge (TE) flaps which enhance the circulation about the airfoil section and devices which delay boundary layer separation such as slots, slats, leading edge (LE) flaps and boundary layer control (BLC). Specific Powered High Lift Systems (PHLS) will be covered in a separate section.

1. Unpowered High Lift Devices

Trailing edge flaps, when lowered, effectively increase the camber of the airfoil section, resulting in an increase in the maximum lift coefficient. As is shown in Figure 3-2, this positive increase in camber results in a negative shift in the angle of zero lift [Ref. 3-3]. An additional feature depicted in this figure is that $\alpha_{\text{Stall (Flapped)}}$ is less than $\alpha_{\text{Stall (Basic)}}$ or that flow separation occurs sooner because of the increased circulation due to the presence of the TE flap.

Flow separation devices aid in maintaining a turbulent boundary layer which prolongs attached flow as long as possible



Trailing Edge Flap Characteristics

FIGURE 3-2 (Ref. 1-1)

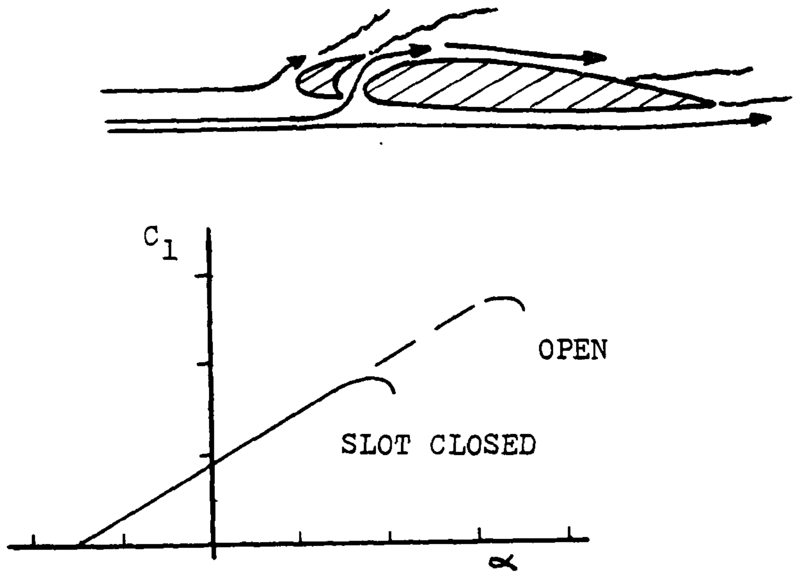
in the presence of an adverse pressure gradient. Three mechanical devices are currently employed:

a. A slot or slat (movable slot) allows the airstream to pass through and acclerate, thereby maintaining a high kinetic energy level in the airflow along the upper airfoil surface. Slots/slats are employed on moderate to thick airfoil sections and are most effective in reducing wing tip stall.

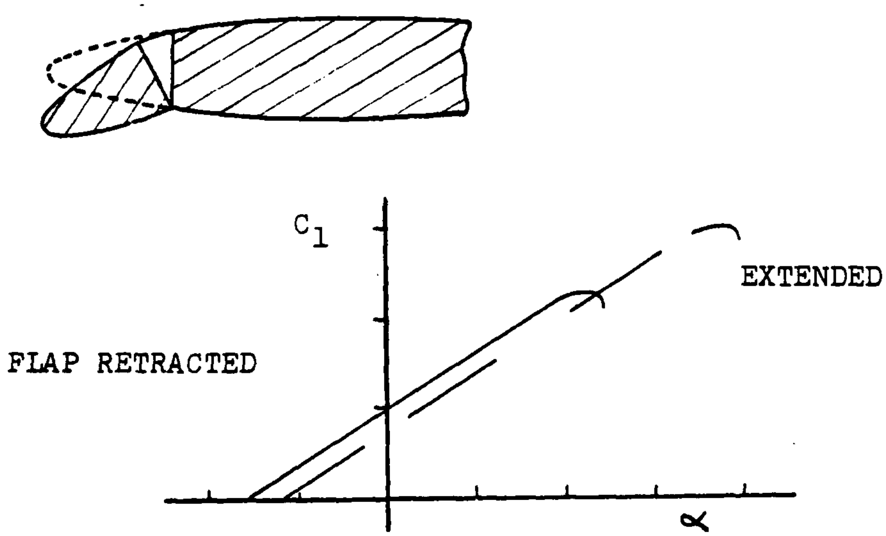
b. LE flaps, when extended, effectively increase the camber of the airfoil section. At high angles of attack the airflow can more readily align itself to the airfoil shape. Due to structural constraints, LE flaps are more practical for use on thin/sharp LE airfoils than the slot. Figure 3-3 shows the effects on $C_{L(max)}$ of slots/slats and LE flaps. Note that when both types of devices are employed, the angle of attack corresponding to $C_{L(max)}$ is increased. For a finite wing with TE flaps <u>only</u>, the maximum achievable coefficient of lift is about 2.5 - 2.7 (full-span flaps) and about 2.2 for a partial span configuration [Ref. 3-4].

c. Boundary Layer Control is mechanized either by activating the low energy boundary layer through blowing high velocity air along the upper airfoil surface or drawing off the boundary layer air through means of suction. Although BLC is dependent upon the use of air or suction pumps, it is not categorized as a PHLS due to the relatively small amount of power required [Ref. 3-3]. Figure 3-4 depicts the trend of the maximum lift capacity of unpowered high lift systems.

Characteristics of Mechanical LE Devices

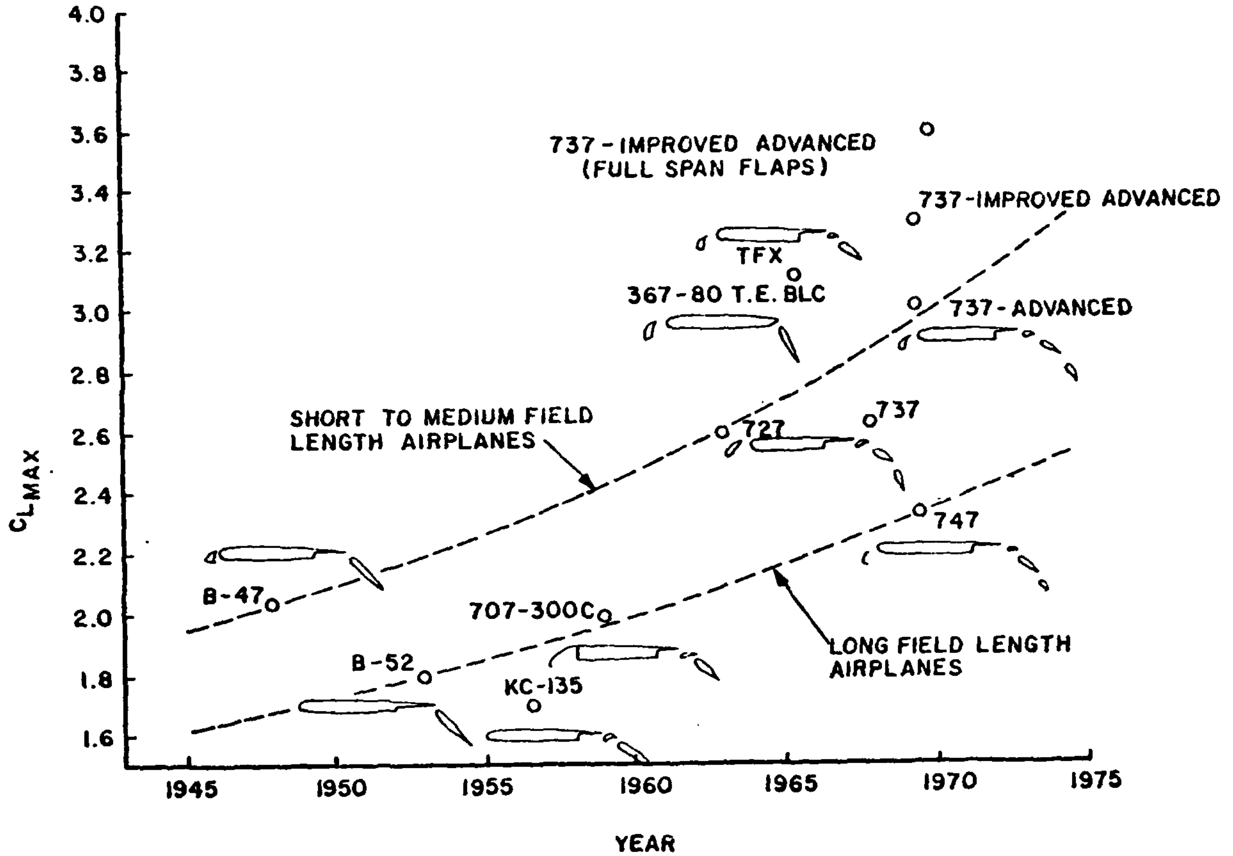


SLOTS/SLATS



LE FLAP

FIGURE 3-3 (Ref. 1-1)



Maximum Lift Capacity of Unpowered High Lift Systems

FIGURE 3-4 (Ref. 1-1)

2. Jet Flap Theory

The most fundamental of the powered high lift devices, the pure jet flap, will be analyzed as an introduction to PHLS. The approach developed in Ref. 3-4 will be followed except where otherwise noted.

As depicted in Figure 3-5, a jet of air is continuously ejected from the trailing edge of the airfoil, deflected downward at some angle (δ) relative to the zero lift line. The jet forms a sheet of high momentum air which is able to sustain a pressure difference across it. An increase in lift on the airfoil results (over that of the no blowing case) along with a corresponding aft movement of the center of pressure

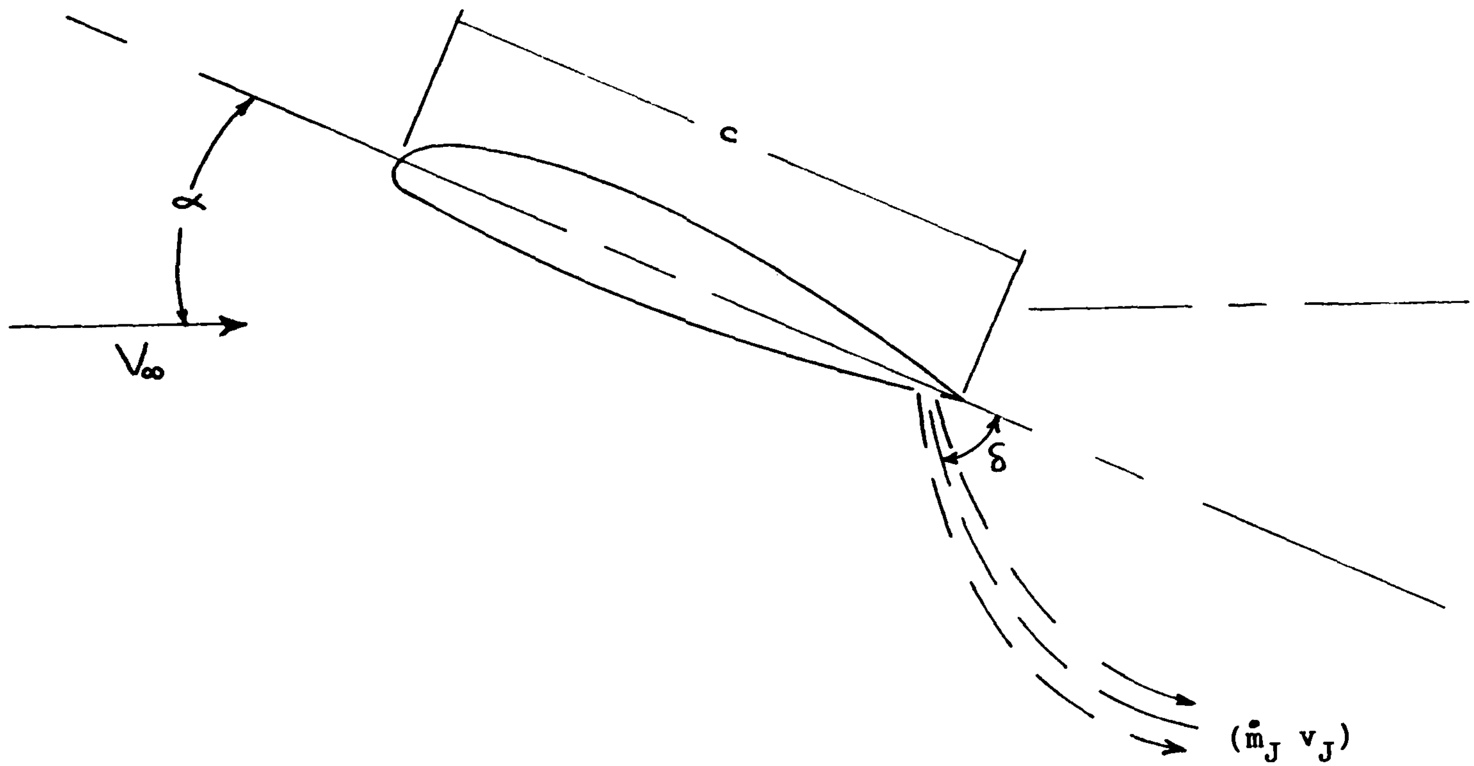
From Thin Airfoil Theory the two-dimensional lift coefficient of the jet-flapped airfoil can be expressed as the sum of the contributions due to just the angle of attack (α) plus that resulting from the jet deflection (δ).

$$c_l = \left[\frac{\partial c_l}{\partial \alpha} \cdot \alpha + \frac{\partial c_l}{\partial \delta} \cdot \delta \right]$$

The derivatives $\left(\frac{\partial c_l}{\partial \alpha} \right)$ and $\left(\frac{\partial c_l}{\partial \delta} \right)$ are functions of C_μ or <u>momentum (blowing) coefficient</u> which is defined:

$$C_\mu = \dot{m}_J V_J \Big/ (q)\,(c) \qquad \text{(Per Unit Span)}$$

A dimensional check of C_μ reveals the ratio of dynamic pressure



Pure Jet Flap

FIGURE 3-5

due to jet momentum to free stream dynamic pressure. Note also that $(\dot{m}_J V_J) \equiv J$ is the momentum flux of the jet and represents the jet force/reaction force.

The total lift on the jet-flapped airfoil can be expressed as that resulting from the circulation around the airfoil plus the vertical component of the jet reaction force.

$$L_T = \left[\rho V_\infty \Gamma + (\dot{m}_J V_J) \sin(\alpha + \delta) \right]$$

If the jet is modeled by a running vortex strength (γ_J) expressed as $\gamma_J = (\dot{m}_J V_J / \; R V_\infty)$, where R is the radius of curvature of the jet, and since in theory the jet aligns itself with the free stream eventually, the circulation induced by the jet can be written as

$$\Gamma_J = \int_0^\infty \gamma \, dx = \left[\frac{\dot{m}_J V_J}{\rho V_\infty} \cdot \sin(\alpha + \delta) \right]$$

Then the total lift on the jet-flapped system is:

$$L_T = \rho V_\infty (\Gamma + \Gamma_J)$$

and the lift coefficient can be expressed as:

$$C_l = \left(\frac{L_T}{q_\infty c} \right) = \left[\frac{2\Gamma}{V_\infty c} + C_\mu \sin(\alpha + \delta) \right]$$

a. Supercirculation

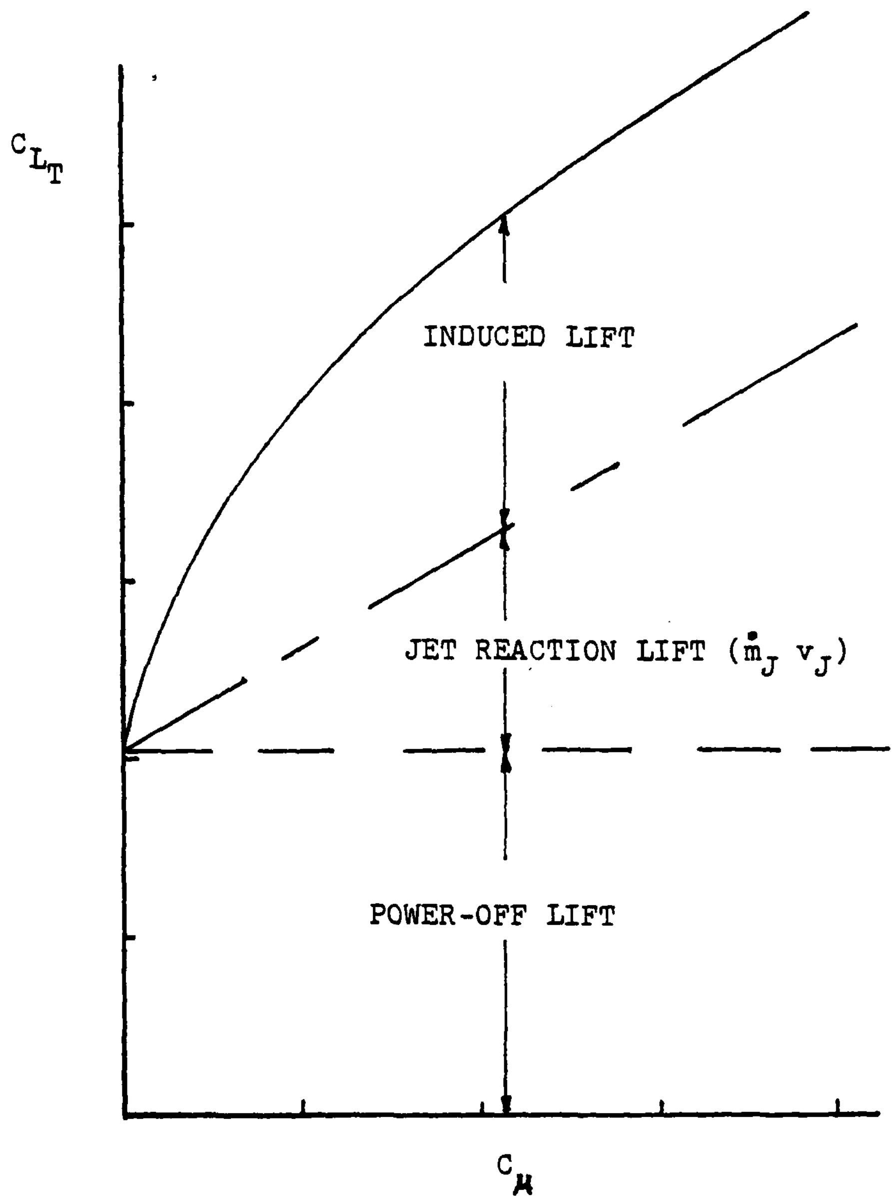
The boundary layer control (BLC) devices mentioned earlier prevented flow separation at the expense of a relatively small amount of power required. Now if the amount of "blowing" (C_μ) is increased beyond that necessary to just keep the boundary layer (BL) attached, the circulation around the wing-flap system will be significantly influenced. The resulting circulation will be <u>greater</u> than that which was present for BLC operations and it is termed "supercirculation" [Ref. 3-5]. Figure 3-6 schematically shows this concept.

b. Factors affecting $C_{L(max)}$

The influence of the jet-induced effects on the surrounding flow field is dependent on the magnitude of the momentum coefficient. For low values of C_μ, flow separation from the airfoil's leading edge reduces the maximum C_L attainable. The use of drooped/highly cambered leading edges or LE blowing has been shown to resolve this problem though. For C_μ greater than 2.0, the jet provides effective BLC thus preventing LE separation. However, additional lift-limiting factors come into play.

(1) <u>Downwash</u>. The jet flap produces a much greater amount of downwash relative to unpowered high lift devices. Experimental results have shown downwash angles of up to forty degrees to exist aft of the airfoil. Placement of the horizontal tail becomes a critical design factor as a high level of control authority is required to trim the aircraft.

(2) <u>Pitching Moment</u>. The nose-down pitch of jet-flapped airfoil is significantly increased over an unpowered HLD. Two contributing factors are first the vertical component of the jet reaction force which acts at the trailing edge of



"SUPERCIRCULATION" EFFECT

FIGURE 3-6

the airfoil, and secondly as C_μ is increased the pressure distribution over the airfoil alters, causing the center of pressure to move aft.

(3) Ground Effect. As C_μ is increased, the jet strikes the ground and obstructs the flow under the airfoil surface. This action limits the attainable maximum C_L. As C_μ or the jet deflection angle (δ) is further increased, vortex formation beneath the airfoil reduces the local pressure. If C_μ is again increased, the center of pressure moves rapidly forward. For jet flapped aircraft operating in ground effect (IGE), large control surface movements are required to trim due to the significant changes in pitching moments and downwash angles.

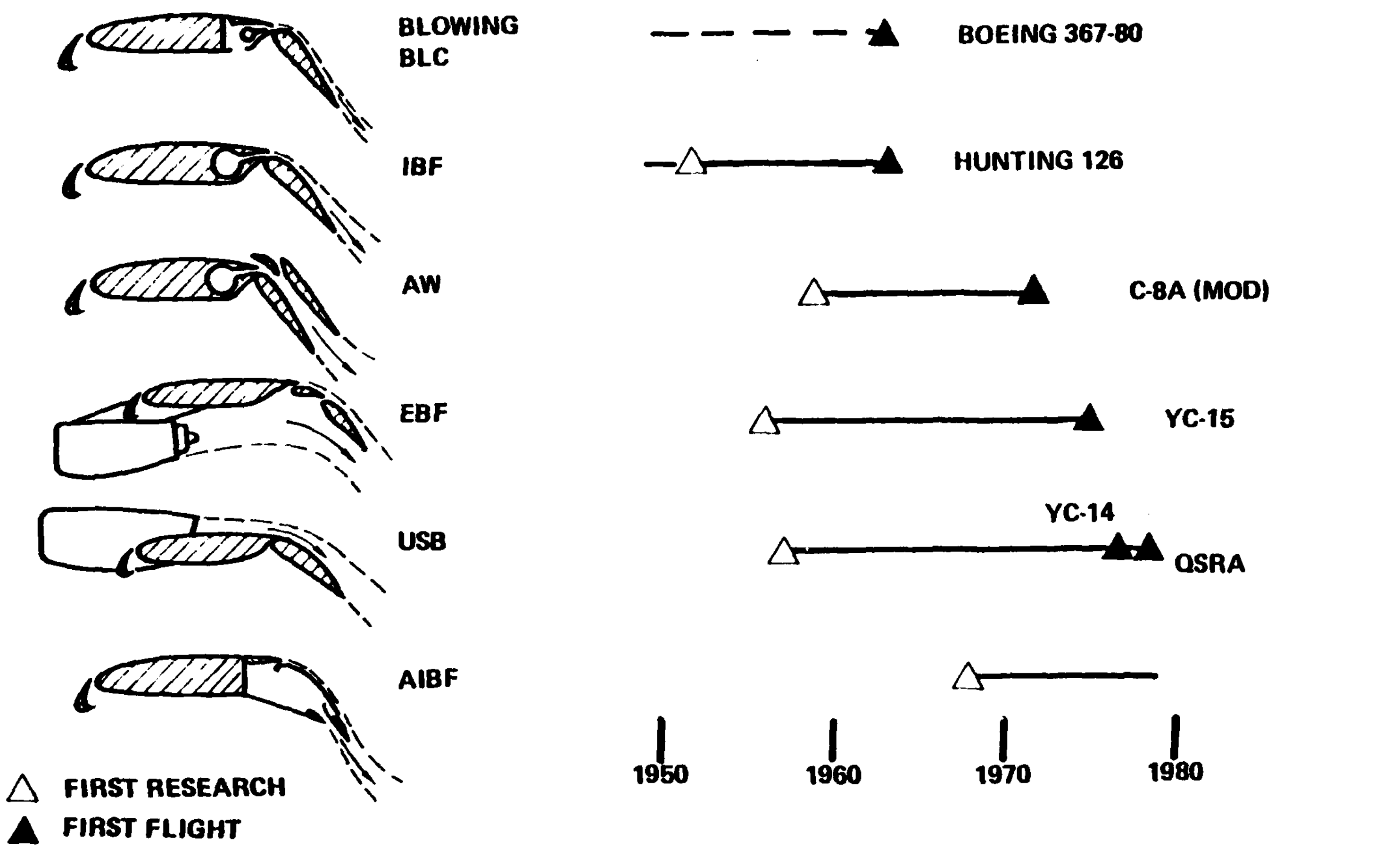
Extensive experimental data pertaining to the three factors detailed above can be found in chapter seven of Ref. 3-4 and throughout Ref. 3-5.

3. Specific Powered High Lift Systems

The following survey of PHLS, except for the AIBF concept, was taken from Ref. 3-6. Figure 3-7 depicts the various systems discussed and Table 3-2 summarizes their pertinent performance characteristics.

a. Internally Blown Flap (IBF or "BLC" Flap)

This concept utilizes tangential slots along the upper surface of the airfoil from which blowing air flows to re-energize the boundary layer of a mechanical flap. Early applications of the IBF with turbojet engines used high pressure (HP) compressor bleed air and required steel or titanium ducting with associated heavy weight penalties. Losses in



Powered High Lift Systems

FIGURE 3-7

(Ref. 5-6)

POWERED HIGH LIFT SYSTEM PERFORMANCE

Type of PHLS	Range of C_u	$C_{L(max)}$
Internally Blown Flap ("BLC" Flap)	0.02 - 0.08	4.5 - 5.0
Jet Flap (JF)	0.40 - 0.70	6.0 - 7.0
Augmentor-Wing (A-W)	0.50 - 0.90	5.0 - 6.0
Externally Blown Flap (EBF)	0.50 - 1.40	5.0 - 6.0
Upper Surface Blowing (USB)	0.70 - 1.20	5.0 - 6.0
Advanced Internally Blown Flap (AIBF) - Lockheed	0.50 - 1.60	5.0 - 7.5

TABLE 3-2

take-off thrust were also large. With high bypass turbofans, the thrust loss was even greater as the bleed air required constituted a greater percentage of engine core flow. The amount of blowing required (see Table 3-2 for C_μ values) is slightly greater than needed for flow attachment to the TE flap. However, lift performance is thereby less sensitive to small air-speed/attitude perturbations.

Examples of turbojet powered military aircraft utilizing the IBF concept include the McDonnell F4 PHANTOM series which employed both LE and TE flap blowing and the Vought F8J CRUSADER which was configured with a blown TE flap/aileron. During the Advanced Medium STOL Transport (AMST) study in 1973, low bypass (2.5:1) turbofan engines were found to have fan pressure ratios sufficient to supply low temperature bleed air directly to IBF systems.

b. Jet Flap (JF)

Compressor/jet exhaust air is used to blow the jet flap along the wing trailing edge (see Fig. 3-5). Early investigators [Fottinger, 1917], [Schubauer, 1933] did not realize the potential of the supercirculation principle. It took the development of the turbojet engine, with its abundant air supply, to find a practical application though. In 1952 H. Constant, director of the National Gas Turbine Establishment (N.G.T.E.) of England, suggested that the exhaust of a gas turbine powerplant be injected over a mechanical flap. Later the mechanical flap was removed. As described in Ref. 3-6, the jet flap concept was: "the complete integration of the propulsive system of an aircraft with its lifting system".

A few years later a jet flap aircraft, the Hunting Percival H.126 (Figure 3-8) was developed and flight tested. Results of wind tunnel tests of the H.126 at NASA Ames are contained in Ref. 3-8. As is evident in Table 3-2, the jet flap operates at a much higher momentum coefficient (C_μ) than the IBF.

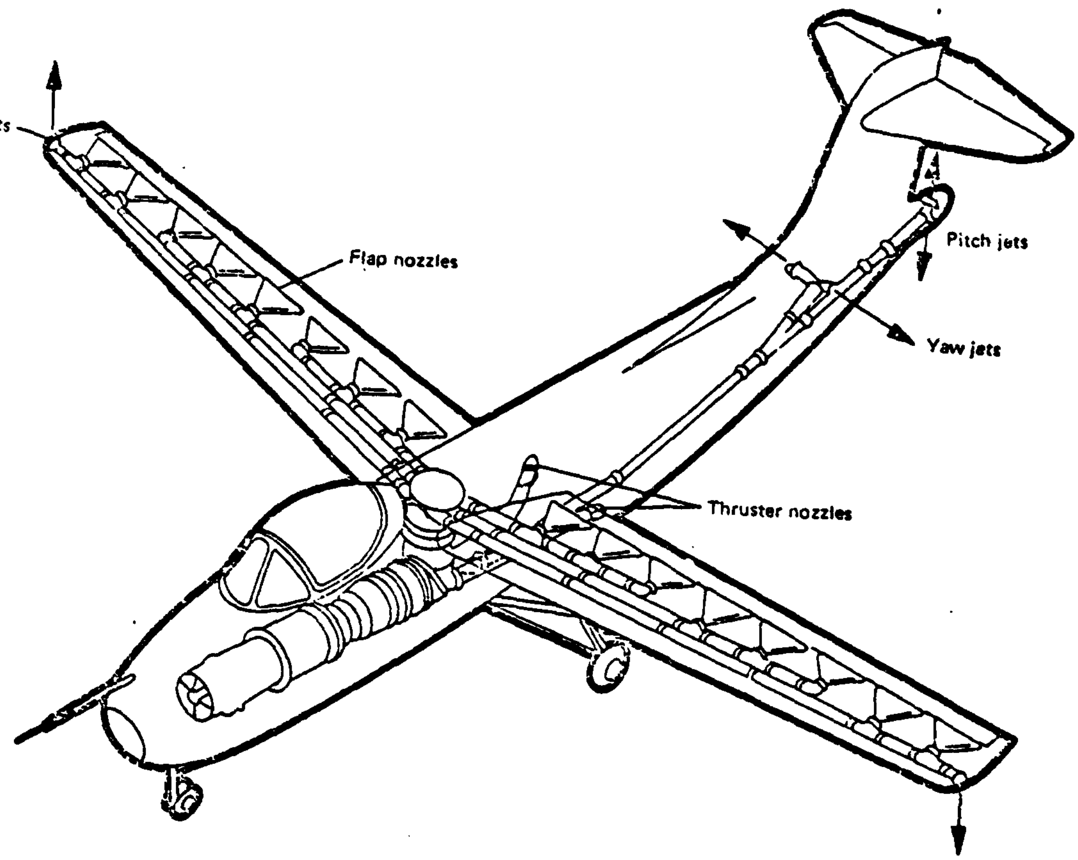
c. Augmentor-Wing (A-W)

This PHLS is a jet flap derivative in which bleed/exhaust air is directed into a spanwise channel formed between an upper and lower flap element. The inlet to this channel along the upper wing surface allows entrainment of ambient air and mixing with the primary jet flow. Ejector action augments the jet thrust. The ejector principle is analyzed in detail in chapter eleven of Ref. 3-4. The ranges of C_μ and $C_{L(max)}$ are listed in Table 3-2.

This concept has been flight tested on the XC-8A de Havilland Buffalo/Spey research aircraft which is covered in Section IV. The A-W offers a greater noise reduction potential than other PHLS. However, the required air ducting and double-flap mechanization is much more complex and highly integrated than other systems.

d. Externally Blown Flap (EBF)

EBF uses the direct impingement of fan/core exhaust on a multislotted mechanical flap system. The increased lift coefficient results from jet deflection and supercirculation. Changes in lift can be controlled by the amount of flap deflection, thrust modulation and/or actuation of spoilers.



Hunting Percival H. 126

FIGURE 3-8 (Ref. 3-7)

NASA investigated this concept in the 1950's, at a time when high temperature turbojet exhaust dictated that the flap system be constructed of steel. Development of high bypass turbofans with cooler and lower velocity fan exhaust have since made the EBF a practical installation. The McDonnell YC-15 (AMST) is such an example. See Section IV for additional details.

Relatively high values of C_{μ} (see Table 3-2) and jet impingement on the flaps cause the noise levels of the EBF installation to be very high. Possible solutions are the use of mixing nozzles on low bypass engines or the utilization of higher bypass turbofans (17:1 minimum). However, trade-offs in optimum cruise flight conditions will most likely result due to cruise thrust losses, lower nozzle efficiencies and high cowl drag.

e. Upper Surface Blown Flap (USB)

With upper surface blowing the jet exhaust flow is directed over the upper surface of the wing through flattened nozzles. The jet is thereby spread out into a flat sheet and follows the wing/flap contour by means of the "Coanda Effect" [Ref. 3-3]. Again, jet deflection and supercirculation increase attainable values of $C_{L(max)}$.

Also investigated by NASA in the 1950's, this concept was limited in practicality due to high temperature turbojet exhaust. USB is compatible with two-engine configurations. With the powerplants mounted well inboard, asymmetric thrust effects can be reduced to minimize the lift loss due to an

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engine failure. Although Table 3-2 shows C_μ values approximately equal to those for EBF, noise levels associated with upper surface blowing are much lower than with the EBF. The wing acts as a shield to the jet flow from below.

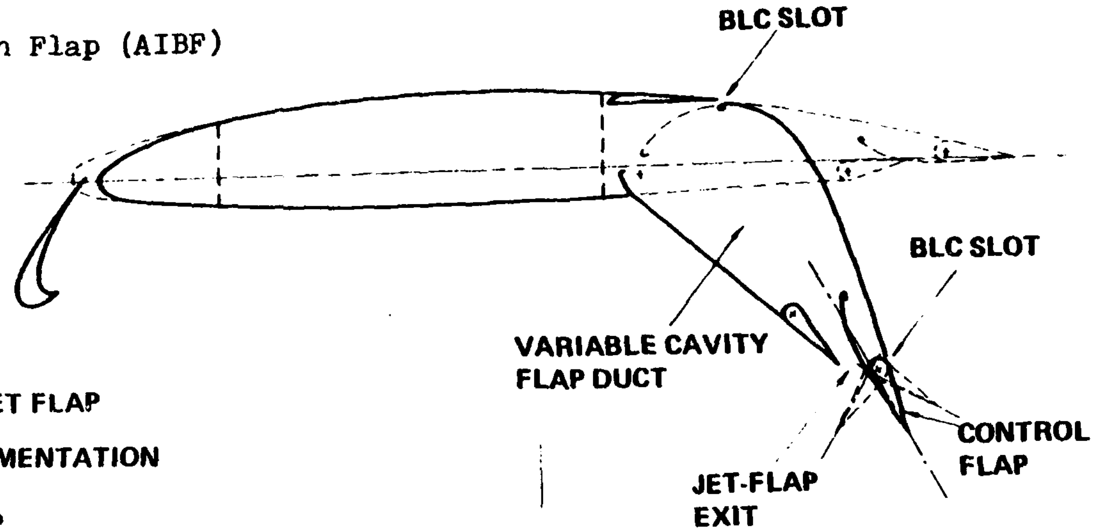
Location of the horizontal tail is a critical design factor for the USB concept as well as for the EBF configuration. Stability and control requirements dictate a large tail volume coefficient and placement of the control surface in a region of favorable downwash. The Boeing YC-14 (AMST) and NASA's Quiet Short-Haul Research Aircraft (QSRA) are applications of the USB concept (see Section IV).

f. Advanced Internally Blown Jet Flap (AIBF)

A hybrid concept combining features of a mechanical flap, the pure JF and the IBF, the AIBF was developed jointly by W. F. Jacobs and C. H. Hurkamp of the Lockheed-Georgia Company. Figure 3-9 depicts the concept. Upon deflection of a trailing edge flap, its cross section expands to form a large spanwise duct. The large volume of this duct permits low pressure fan air to be used for blowing. A control flap located at the TE of the main flap makes it possible to vector the jet formed by the blowing air. BLC is employed over main and control flap surfaces. Changes in lift can be controlled by varying main flap deflection, modulation of the control flap angle and/or the value of C_μ.

Further details of the AIBF concept and the results of large scale model tests in the NASA Ames 40 X 80 ft. wind tunnel can be found in Ref. 3-8. To date the AIBF configuration has not been flight tested.

Advanced Internally Blown Flap (AIBF)



- MECHANICAL FLAP + BLC + JET FLAP
  - o HIGH OVERALL LIFT AUGMENTATION
- FAST-ACTING CONTROL FLAP
  - o POWERFUL MEANS FOR FLIGHT-PATH AND LATERAL CONTROLS
  - o ONE MAIN-FLAP DEFLECTION FOR TAKEOFF/LANDING A POSSIBILITY
- LARGE FLAP DUCT
  - o NO FIXED INTERNAL FLOW DUCT
  - o LOW DUCT LOSSES AND NOISE
- PROPULSION/LIFT SYSTEM INTEGRATION FLEXIBILITY
- APPLICABLE TO STOL AND ULTRA-STOL

FIGURE 3-9

(Ref. 5-6)

IV. CURRENT APPLICATION OF POWERED HIGH LIFT SYSTEMS

Within the past decade requirements for transport aircraft having STOL capabilities have been generated by both civil and military aviation interests in the United States. As was depicted by Figure 3-4, the limitations of unpowered high lift devices have led to an increased interest in further development of propulsive (powered) lift systems which assist the aerodynamic surfaces, thereby lowering engine power requirements [Ref. 4-1]. Concurrently, the steady improvements in turbofan engine performance have produced a much more quiet, fuel-efficient powerplant capable of satisfying the bleed air requirements of various powered-lift concepts. In this section, current research/prototype aircraft developments employing PHLS are reviewed.

A. NASA RESEARCH VEHICLES

In the civil operations area, government funded research by the National Aeronautics and Space Administration (NASA) has provided a strong technology base applicable to the systems discussed in Section III. The problems of airport noise and congestion, spawned during the 1960's, led to the creation of the Quiet Propulsive-Lift Technology (QPLT) program in the early 1970's [Ref. 3-6]. The objective was to gather inflight data which would support innovative design methods and aid in the formulation of certification criteria for quiet, powered-lift aircraft.

Two jet STOL research airplanes have been developed by NASA. These vehicles are considered to be flight research facilities, just as a flight simulator or a wind tunnel is a test facility. Their primary mission is to gather data [Ref. 4-2].

1. XC-8A (Buffalo/Spey): Augmentor - wing principle.
2. QSRA: Upper surface blowing concept.

DESIGNATION:	XC-8A (Buffalo/Spey)
MANUFACTURER:	De Havilland Aircraft of Canada/Boeing Company
SPONSOR:	NASA, Department of Industry, Trade and Commerce of Canada (DITC)
CONCEPT:	Augmentor - wing

MILESTONES:

NASA/Canadian Government research program begun	1965
Large-scale model wind tunnel testing	1967-1970
C-8A modification begun	1971
XC-8A rollout	Feb 1972
First flight	01 May 1972

WEIGHT:	45,000 LBS (max T/O)
ENGINES:	(2) Rolls-Royce Spey Mk 801 SF turbofans of 9000 lbs thrust each.
LAYOUT:	See Fig. 4-1

COMMENTS:

This joint venture was initiated to develop a a powered-lift jet STOL transport utilizing the augmentor-wing concept. De Havilland of Canada had begun augmentor-wing research in 1960 and by 1970 had completed large-scale model wind tunnel tests at the NASA Ames Research Center. The success of their work warranted the development of a flight research vehicle and the U.S. and Canadian governments agreed to the modification of a de Havilland C-8A Buffalo turboprop transport. NASA contracted with the Boeing Company to modify the airframe, install the augmentor system and conduct initial flight testing. De Havilland and Rolls-Royce of Canada were contracted by the DITC to supply the powerplants and modify the nacelles for the installation.

Major airframe modifications included: (1) an eighteen percent reduction in wing span, (2) replacement of the wing structure aft of the rear spar by the augmentor flap system, (3) installation of drooped ailerons with BLC, fixed full-span L.E. slats and a redesigned spoiler system, (4) an air distribution system to duct fan air to the augmentor-wing system, aileron and fuselage BLC, (5) an increased capacity hydraulic system and (6) a fixed landing gear system.

The four-section augmentor flaps have a constant chord of 3.5 ft. and a maximum deflection of 75 degrees. When extended these upper/lower slotted segments deflect the primary jet flow downward, mixing it with the freestream flow coming over the upper wing surface. Simultaneously, air is pulled through both the upper/lower flap slots. The combined effect is that of mixing four different airflows between the two flap segments which increases both lift and thrust and provides suction BLC to delay boundary layer separation.

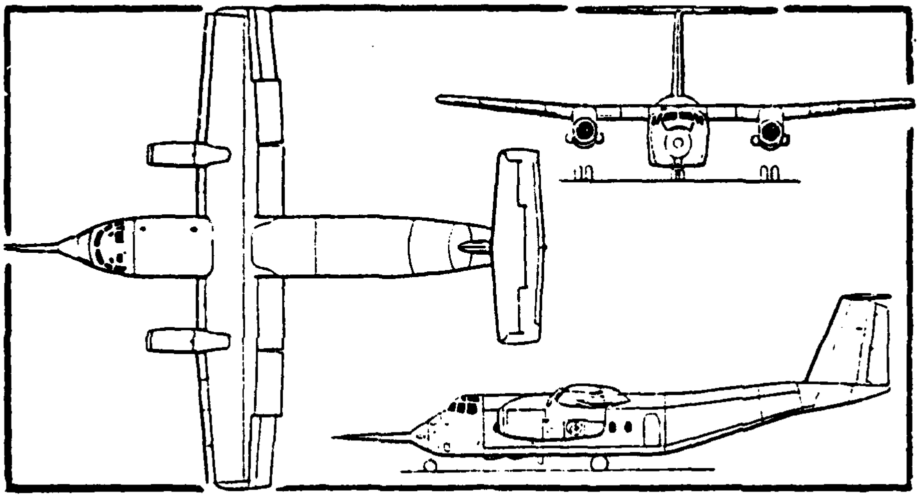
The lateral control system utilized three surfaces to provide the required rolling moments: (1) a mechanically drooped aileron (30 degrees maximum deflection), (2) a spoiler located forward of the aileron and (3) an augmentor choke in the TE flap system. The purpose of the chokes was to restrict the fan air outflow area of the flap system and thereby control the lift produced. Upon touchdown all four augmentor chokes were closed to aid in lift spoiling.

Powerplant selection was driven by two primary factors: (1) minimum cost and (2) the requirement to retain as much of the original Buffalo airframe configuration as possible. Earlier studies by de Havilland indicated that between a 35 to 45 percent blowing thrust to total thrust ratio was optimum to balance the take-off/landing distances. After considering a dual propulsion system consisting of separate cruise and blowing engines, the Rolls-Royce Spey 511-8 turbofan was selected. This commercial production engine was extensively modified by: (1) separation of the low pressure (LP) compressor/fan airflow from the turbine exhaust by an annular bypass duct which routed the fan air to the augmentor ducting, (2) replacement of the LP compressor with a more durable version and (3) incorporation of a split flow tail pipe with vectoring nozzles, an off-the-shelf item directly from the Rolls-Royce PEGASUS program. Following sixty-three hours of qualification testing, the engines were redesignated as the Mk 801 SF Spey.

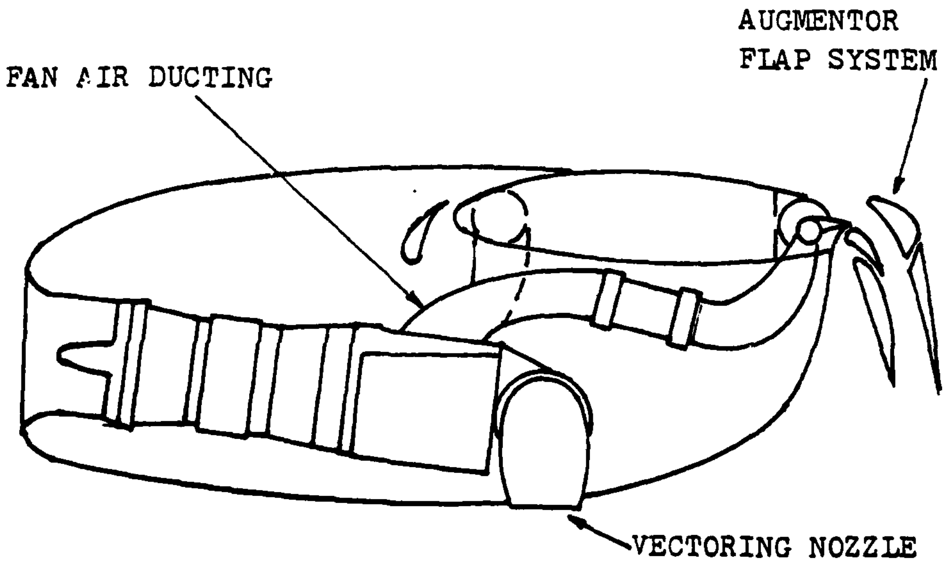
The fan air ducting systems from each engine provided air to the complete flap system, fuselage BLC and the <u>opposite</u> blown aileron. This distribution system was designed so that in case of an engine failure, a higher proportion of augmentor flow was available to the engine-out side together with aileron blowing on that side. Overall duct losses were relatively low. A maximum of an eleven percent loss was measured between the fan shroud exit and the aileron duct entrance.

The contractor's flight test program was designed to establish the basic airworthiness of the aircraft. While verification of the structural design and an evaluation of the aircraft's systems were performed, no STOL landings were attempted. Although takeoff power was restricted to eighty-five percent of maximum thrust available, takeoff distances of less than 2000 feet were achieved. At the end of eighteen hours of testing Boeing concluded: (1) inflight engine performance matched that determined during ground tests, (2) except for slow decelerations below ninety percent N_H, engine response was satisfactory irregardless of flight condition, and (3) vectoring nozzles should be rotated forward or idle thrust selected at 60 kts. after landing to preclude hot gas reingestion. Although stall test results were not available, during wave-off evaluations by Boeing, lift coefficients of up to 7.0 were developed. Approximately 1.5 was attributed to thrust deflection.

<u>SOURCES</u>: Ref. 2-3: '77-358
Ref. 4-1



XC-8A (Buffalo/Spey)



Mk 801 SF Spey Nacelle

FIGURE 4-1

DESIGNATION:	QSRA
MANUFACTURER:	Boeing Commercial Airplane Company
SPONSOR:	NASA
CONCEPT:	Upper Surface Blowing (USB)

MILESTONES:

QUESTOL design studies	1971
QSRA preliminary study by NASA	Jan 1974
RFP issued	Nov 1974
Contract award to Boeing	Mar 1976
QSRA rollout	31 Mar 1978
First flight	06 Jul 1978
Delivery to NASA Ames	03 Aug 1978

WEIGHT: 50,000 LBS

ENGINES: (4) Lycoming YF-102 turbofans (bypass ratio 6.5:1) of 7500 LBS thrust each

LAYOUT: See Fig. 4-2

COMMENTS: The first jet STOL research vehicle developed by NASA, the XC-8A Buffalo/Spey, represented first generation propulsive-lift technology with approach lift coefficients of 3.5 to 4.0. Its major shortcoming though was the high sideline noise level from the thrust vectoring nozzles. A follow-on program, the Quiet Experimental STOL airplane (QUESTOL), was initiated but funding cuts in early 1973 forced its cancellation.

In January 1974, NASA embarked on the development of a second generation jet STOL research aircraft (see Table 4-1) which would feature very low sideline noise levels and achieve approach lift coefficients of 4.5 to 5.5 (see Figure 4-3). Preliminary design studies were completed by Boeing and the Lockheed-Georgia Company, with the former being awarded the hardware contract for their USB-flapped configuration.

As was the case in the Buffalo/Spey program, budget limitations dictated a management approach which included the use of an existing airframe where possible, "off-the-shelf" hardware, use of goals vice rigid requirements and in-house participation where possible. Two significant factors were responsible for

QSRA INITIAL GOALS

Approach lift coefficient = 4.6 (steep approach with margins)

Approach path of -7-1/2° with margin for gusts, wind, etc.

90 EPNdB combined takeoff and landing footprint area, when scaled to 150,000 lb of 1 mile^2

Minimum duration of test mission - 50 minutes

Minimum wing loading at gross weight = 65 lb/ft^2

Maximum cruise speed 160 knots

Wing/nacelle configuration representative of cruise at M = 0.74.

TABLE 4-1

minimizing early program costs: (1) the availability of a de Havilland C-8A transport airframe through an inter-agency transfer, and (2) the use of six YF-102 high bypass turbofan engines (obtained from the U.S. Air Force's AX fly-off competition).

Extensive wind tunnel testing was completed using a 0.55 scale model in the Ames 40 by 80 ft. tunnel. Powered by five turbofan engines, the model was utilized to check the design analysis and to study the effects of configuration changes.

The fuselage of the C-8A Buffalo was essentially unmodified except for some structural reinforcement aft and incorporation of a fairing at the wing-body junction. SAS actuators were added to both the rudder and elevator and elevator control was converted to a powered system. Boeing designed and built a new wing (Fig. 4-2) employing supercritical airfoil technology and containing two integral fuel cells of ten thousand pounds capacity. High lift devices consist of fixed LE flaps with BLC, two trailing edge USB flaps on either side of centerline, two double-slotted TE flaps further outboard and drooped ailerons with BLC.

The four turbofan engines are cantilever mounted from the forward wing spar. The nacelles incorporate sound absorbing honeycomb liners to attenuate powerplant noise. Engine fan/core exhaust flow is mixed in a nozzle then proceeds along the contour of the USB flaps. Both the LE flaps and the ailerons are blown by a mixed-flow BLC system which uses both compressor and/or fan bleed air, depending on the power setting. When the fan pressure ratio is high (high power settings), compressor bleed is zero. At low power settings about ten percent core airflow is bled, thereby maintaining a constant pressure BLC system. Thrust loss is approximately three percent at the full power setting and port side engines supply the starboard side BLC system and vice versa. In the event of the failure of either outboard engine some degree of roll compensation is thereby built-in.

The Ames Flight Simulator for Advanced Aircraft (FSAA) was used during the QSRA development to study various levels of control power and rates that would ensure satisfactory handling qualities under normal and emergency flight conditions. Design trade-offs were conducted for both pitch and roll control augmentation systems. The details of FSAA studies performed can be found in Ref. 4-2. A single channel, three-axis limited authority series type SAS evolved in which the yaw and roll axes are stabilized by an analog system and the pitch axis is rate-command, attitude-hold system. Both the longitudinal SAS and the direct lift control (DLC) system are integrated with a digital computer.

The USB flaps, the spoilers and the double-slotted TE flaps are all individually actuated by a digital control system. The pilot can modulate USB flap position between 30 degrees and 66 degrees deflection through a throttle-mounted thumb switch. By varying the flap setting during a landing approach, an additional means of glideslope control is provided. The pilot also has the option to command asymmetric deployment of the double-slotted flaps to trim in case of an engine failure.

A fourteen-flight contractor test program was completed in just twenty-seven days. The QSRA demonstrated a minimum airspeed of 50 kts. (all engines operating) and a maximum lift coefficient of 9.06. Other highlights included: (1) a 52 kt. minimum control speed with an outboard engine out, (2) maximum airspeed of 190 kts. at 15,000 ft. and (3) a takeoff distance of 820. ft. No STOL landings were attempted.

Table 4-2 lists the performance values demonstrated by the QSRA through the first forty NASA research flights. Low noise characteristics was one of the goals of the QSRA endeavor. Figure 4-4 shows a comparison of the 90 EPNdB footprint area of a conventional medium jet transport with that of the QSRA when scaled to the same gross weight.

At the completion of the initial flight testing program, NASA will conduct a comprehensive flight experiment program which will include the following objectives: (1) establish design criteria to assess the best compromise with respect to wing loading, propulsive lift and flight control systems, (2) provide the

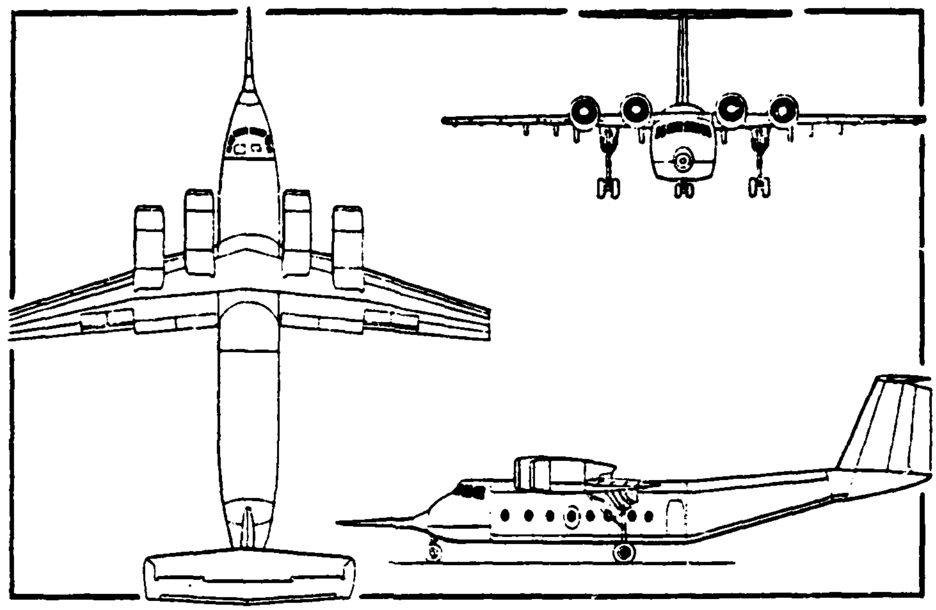
DEMONSTRATED QSRA PERFORMANCE

$C_{L_{max}}$ (All Engines Operating)	8.9
$C_{L_{approach}}$	5.5
V_{min}	50 kt.
Approach Flight Path	7.5°
FAR Field Length	1500 ft.
Turn Radius	600 ft.
Ground Roll (zero wind)	
Takeoff	650 ft.
Landing	550 ft.
90 EPNdB Footprint	1.0 sq. mi.
500 ft. Sideline Noise	
Takeoff	93 EPNdB
Landing	89 "

TABLE 4-2

on-going NASA/FAA study of STOL certification criteria and proposed airworthiness standards with test case, relevent flight experience for high propulsive-lift performance levels and for advanced flight control concepts and displays and (3) investigate ground proximity effects on aerodynamics and stability and control at very high lift coefficients.

SOURCES: Ref. 2-3: '77-359
Ref. 4-2, 4-3, 4-4

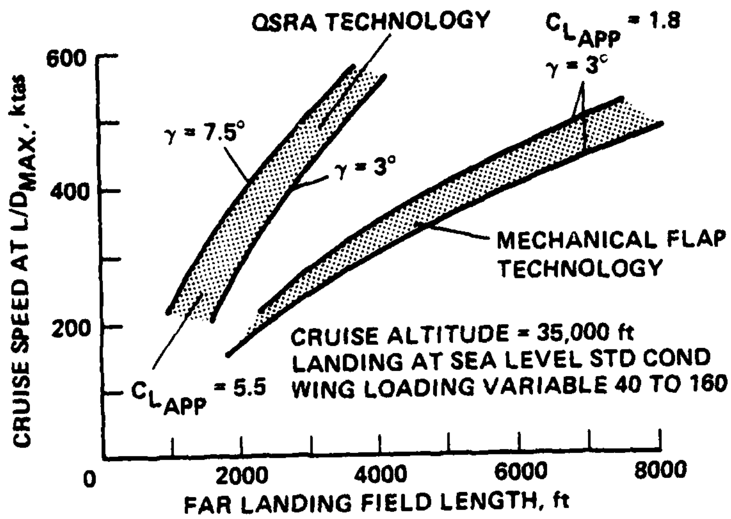


QSRA

Principal Dimensions:

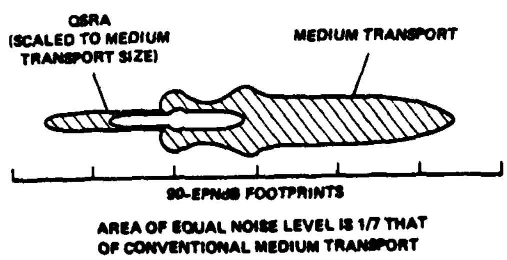
Aspect ratio	9.0
Length overall	93.25 ft.
Height overall	28.7 ft.
Wing area	600. sq. ft.
Max. wing loading -	
Normal	83.3 lb/sq. ft.
Overload	100.0 lb/sq. ft.

FIGURE 4-2



QSRA Technology Comparison (Ref. 4-3)

FIGURE 4-3



QSRA Noise Level Comparison (Ref. 4-2)

FIGURE 4-4

B. MILITARY PROTOTYPES

Military funded research in propulsive lift technology began to increase when a turbofan STOL transport replacement was considered for the Lockheed C-130 Hercules [Ref. 3-6]. The Advanced Medium Short Takeoff and Landing Transport (AMST) program was created and in early 1972 requests for proposals (RFP) were issued by the Air Force Systems Command to nine U.S. contractors. Table 4-3 lists primary program goals. Responses were received from Fairchild Industries, McDonnell-Douglas, Boeing, a Lockheed-Georgia/North American Rockwell team and Bell Aerospace. In November, Boeing and McDonnell-Douglas were awarded contracts to design, build and flight test two prototypes each which would participate in a fly-off competition. Initial program funding allocated 105.9 million dollars to Boeing and 85.9 million dollars to McDonnell-Douglas. Once the program was well underway though, Congress set a limit of 25 million dollars on the AMST program in the fiscal year 1974 budget. Although the Air Force had requested more than twice that amount of money, the prototypes were developed and initial testing was completed [Ref. 2-3: '77-259].

1. YC-14: Upper surface blown flap concept.
2. YC-15: Externally blown flap concept.

Even though the fiscal year 1979 budget contained no funding for the AMST program, an official cancellation has not been announced. Both contractors are exploring the market, foreign and domestic, for commercial derivatives of the YC-14/15 designs. As recent as April 1979, the Air Force directed both McDonnell-Douglas and Boeing to conduct studies of the AMST with an air-launch capability for the MX advanced intercontinental ballistic missile [Refs. 4-11, 4-12].

AMST PROGRAM GOALS (Partial List)

Aircraft must be capable of operating from 2000 ft. semi-prepared strips with acceptable safety margins, carrying a 27,000. pound payload on a 400 nm. mission with a landing and unrefueled takeoff at the midpoint.

Aircraft cargo compartment approximate measurements: 47 ft. by 12 ft. by 12 ft.

Limit costs of the 300th production unit to five million (1972 dollars).

TABLE 4-3

DESIGNATION:	YC-14 (AMST)
MANUFACTURER:	The Boeing Company
SPONSOR:	USAF Systems Command
CONCEPT:	Upper Surface Blown (USB) Flap

MILESTONES:

RFP issue to industry	1972
Boeing and McDonnell-Douglas selected for prototype development	Nov 1972
AMST budget cut	Dec 1973
Prototype roll-out	May 1976
First flight (No. 1)	09 Aug 1976
First flight (No.2)	21 Oct 1976

WEIGHT: 170,000 LBS (max STOL takeoff)
160,000 LBS (STOL landing)

ENGINES: (2) G.E. CF6-50D twin-shaft turbofans (bypass ratio 4.3:1) of 51,000 LBS thrust each

LAYOUT: See Figure 4-5

COMMENTS: The most prominent feature of the YC-14 is its relatively small, unswept shoulder-mounted supercritical wing from which the engine nacelles are cantilevered ahead and above the leading edge. Engine location was dictated by Boeing's choice of powered-lift concept, the USB flap (see Section III). Other advantages of the unique engine location include: (1) the type of thrust reverser employed eliminates foreign object damage (FOD) and visibility restrictions during reversing and allows engines to remain running during cargo loading/unloading, (2) powerplant noise levels are diminished, (3) infared signature from below is reduced and (4) low speed rolling/yawing moments are minimized during an engine failure.

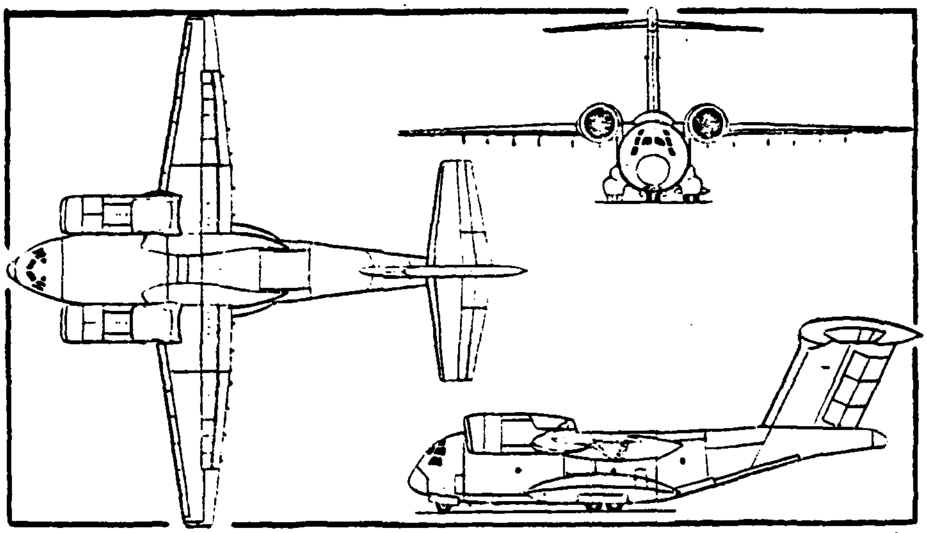
Principal dimensions of the aircraft are listed in Table 4-4. The wide-body fuselage design will accomodate one hundred fifty combat-equipped troops or 27,000 lbs of cargo in STOL operations. The empennage consists of a high T-tail assembly consisting of a vertical fin of constant section using three identical double-hinged rudder panels and a large tail volume coefficient horizontal stabilizer/elevator combination.

Extensive wind tunnel work went into tailoring the USB system to provide satisfactory performance in both STOL and cruise flight regimes. In the case of the jet nozzles, STOL operations require a 2-D high aspect ratio nozzle (a thin spread-out jet) to optimize the flow turning (Coanda Effect) along the USB surface. On the other hand cruise flight efficiency dictates a more circular exit geometry. A variable geometry nozzle along with flow director vanes was developed to satisfy these mission requirements (Figure 4-6).

The flap system arrangement consists of: (1) full-span, variable camber LE flaps supplemented with BLC (used for takeoff and landing), (2) outboard TE two-segment double-slotted flaps (used for takeoff and landing) and (3) the inboard USB flaps, directly aft of the engines, which are modulated automatically by the stability augmentation system (SAS). The USB flaps are used for landings only and are not controlled by the pilot. They are employed for vectoring thrust as well as varying the drag for airspeed control. Lateral control is accomplished through conventional ailerons and spoilers, the latter which double a speed brakes in flight. During a STOL landing, the spoilers are used for Direct Lift Control (DLC).

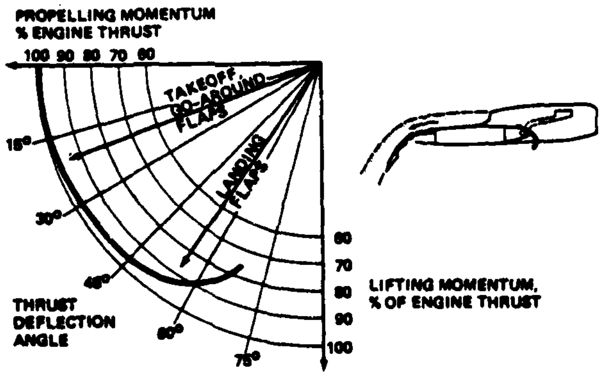
The flight control system on the YC-14 consists of integrated mechanical/electrical systems. The SAS features an "airspeed hold" mode which automatically modulates USB flap position and throttle setting to vary drag, thrust vector and angle of attack in order to maintain a selected target airspeed. During a single-engine approach, both the USB flap and the throttle on the good engine side continue to modulate to maintain the approach airspeed. NASA Ames' FSAA simulator was utilized extensively during the development of the flight control system.

After three months of flight testing by the contractor, both prototypes were evaluated by the USAF Joint Test Force (JTF) at Edwards Air Force Base. Expansion of the cruise configuration speed envelope was taken to 0.78 mach and an equivalent airspeed of 362 kts. Structural damping was good throughout the speed range. The G.E. CF6-50D turbofans proved to be highly reliable and the upward vectoring thrust reversers proved highly effective during ground operations. During minimum controllable airspeed tests, an angle of attack limit of 32 degrees and maximum rate of descent of



YC-14

FIGURE 4-5



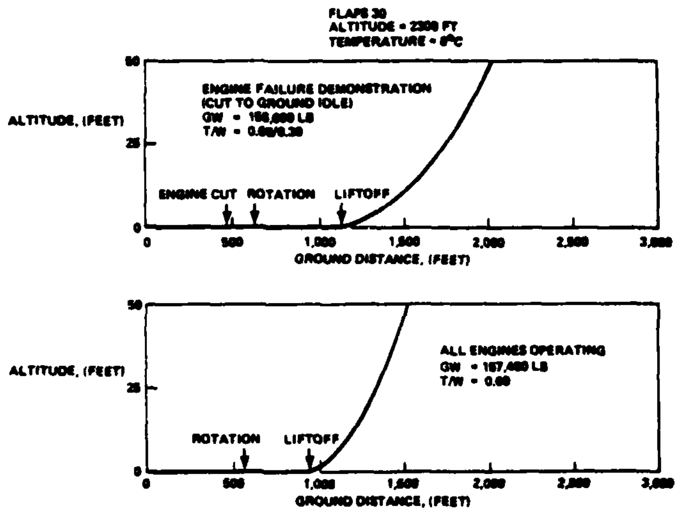
Nozzle Turning Efficiency (Ref. 4-5)

FIGURE 4-6

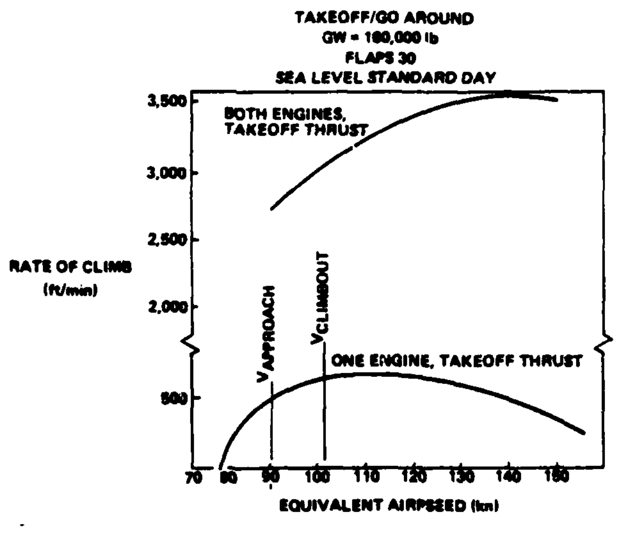
2500 ft. per minute was established since normal stall characteristics were not present. The maximum lift coefficient obtained in the STOL configuration was 7.0.

Figure 4-7 shows demonstrated takeoff and climb performance. STOL landing performance surpassed predictions. Utilizing thrust reversers at reverse idle, a 160,000 lb aircraft was brought to rest on 800 ft. of runway. STOL approaches were flown on a six degree glide slope using little or no flare prior to touchdown. The maximum rate of sink encountered was 14.5 feet per second. A positive ground effect in the STOL landing configuration brought a twenty-five percent reduction in touchdown sink rates. Engine-out tests in the STOL configuration resulted in such low lateral/directional control forces that retrimming was not neccessary.

<u>SOURCES</u>: Ref. 2-3: '77-259,260
Refs. 4-5, 4-6, 4-7.



YC-14 Takeoff Performance



YC-14 Climb Performance

(Ref. 4-7)

FIGURE 4-7

PRINCIPAL DIMENSIONS - YC-14/YC-15

Dimensions	YC-14	YC-15
External (Ft.)		
Wing span	129.0	132.6 (No. 1) 110.3 (No. 2)
Length overall	131.7	124.3
Height overall	48.3	43.3
Fuselage width	17.8	18.0
Tailplane span	54.7	56.0
Internal (Ft.)		
Cargo Compartment:		
Length	47.3	47.0
Width	11.6	11.7
Height	11.6	11.3
Areas ($Ft.^2$)		
Wing (Gross)	1762.	2107. (No. 1) 1740. (No. 2)

TABLE 4-4

DESIGNATION: YC-15 (AMST)

MANUFACTURER: McDonnell Douglas Corporation

SPONSOR: USAF Systems Command

CONCEPT: Externally Blown Flap (EBF)

MILESTONES:

*Same as YC-14 through December 1973	
Prototype No. 1 roll-out	05 Aug 1975
First flight	26 Aug 1975
First flight No. 2	05 Dec 1975

WEIGHT: 180,000 LBS (max STOL takeoff)
150,000 LBS (Design landing weight)

ENGINES: (4) P & W JT8D-17 turbofan engines (bypass ratio 1.03:1) of 16,000 LBS thrust each.

LAYOUT: See Figure 4-8

COMMENTS: The YC-15 represents a different aerodynamic approach to achievement of the AMST program goals (Table 4-3). McDonnell Douglas selected the EBF powered-lift concept (see Section III). A fairly thick, straight wing using a supercritical airfoil section forms the backbone of the EBF system. The four P & W JT8D-17 turbofans are mounted well ahead of the wing leading edge. Engine core and fan exhaust impinge directly on the two-segment titanium slotted flap system, providing powered-lift both directly as vectored thrust and indirectly due to flow entrainment through the flap slots. The turbofan engines are fitted with special nozzles that mix ambient air with the hot core gases, reducing the outflow temperature to prevent damage to the wing structure.

Table 4-4 lists the principal dimensions of the aircraft. The wide-body fuselage features a rear undersurface loading ramp and a maximum weight-limited payload capacity of 62,000 lbs. The conventional tail surfaces are large with a trimmable horizontal stabilizer and the high mounted elevator is sized for 80 kt. rotation speeds. The two-segment double-hinged rudder provides adequate directional ground control with an engine out on takeoff. The wing is fitted with LE flaps inboard of the nacelles and slats outboard. Three panels of fly-by-wire spoilers are fitted directly forward of the TE flaps. The two outboard spoilers work with the aileron to augment lateral control.

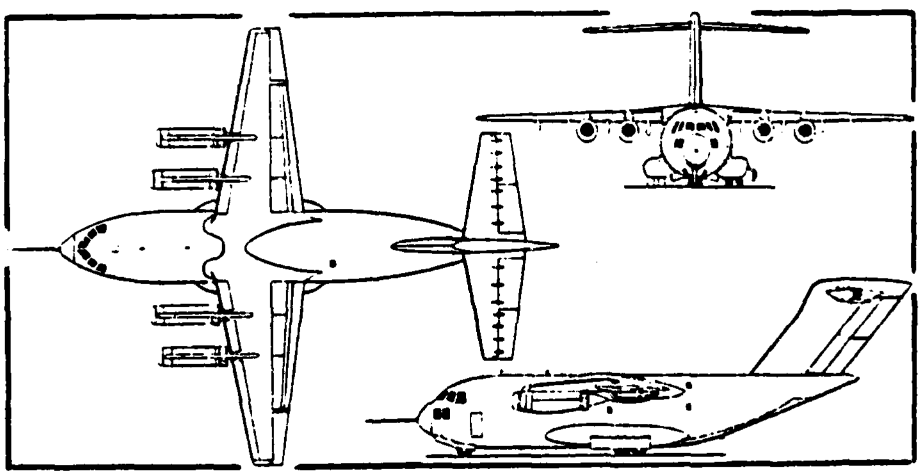
A Stability and Control Augmentation System (SCAS) was designed to enhance the YC-15's handling qualities. The heart of this system consists of three, dual-channel digital computers. To make the aircraft more responsive suring STOL approaches, Direct Lift Control (DLC) spoilers are used. The DLC is a zero-bias system where the spoilers extend only for a down correction; thrust must be added for all up corrections. Thus, both up and down corrections to glidepath are conservative with respect to thrust.

A three phase flight test program was conducted from August 1975 to August 1977. Phase I consisted of 473 flight hours during which a flutter clearance, SCAS on and off, resulted in operational flight limits of 0.76 mach and 350 kts. equivalent airspeed. Minimum controllable airspeed tests were conducted to investigate the aircraft's high angle of attack characteristics and to ensure safe operating margins for STOL operations. About 370 stalls were performed in all normal slat/flap settings, SCAS on and off, idle to maximum continuous thrust (MCT), symmetric and engine-out configurations and straight and turning entries. The YC-15 exhibited almost classic stall characteristics, g-break and nose drop with no tendency to roll-off. However, virtually no natural prestall buffet/warning was exhibited. A warning horn was installed to define an operational angle of attack limit (32 degrees) in the STOL landing configurations. In the approach configuration, minimum controllable speeds varied from 89 kts. at idle to 66 kts. at MCT. The thrust reversers (Figure 4-9), designed to permit the use of reverse thrust without ground flow impingement/hot gas reingestion, were cleared to MCT at zero airspeed. Tests on a powdery dirt runway resulted in no FOD to the engines and backing maneuvers employing the reversers were routinely used when parking the aircraft. Also, the thrust reversers were regularly used for penetration descents from altitude.

During Phase II a thrust management system (TMS) was incorporated to reduce the pilot workload and improve engine-out STOL performance. The TMS limits engine thrust to the value indicated on a thrust rating indicator at full throttle. At reduced power settings it maintains engine trim. Initial predictions of sink rate on a six degree flight path produced a fifteen feet per second rate of descent and dictated the need for a flare maneuver. However, analysis of early landings showed the ground effect to be strongly positive so the flare maneuver was abandoned. For the twelve STOL landings performed during this phase, the touchdown dispersion was +25 ft. with an average touchdown sink rate of nine feet per second (no flare).

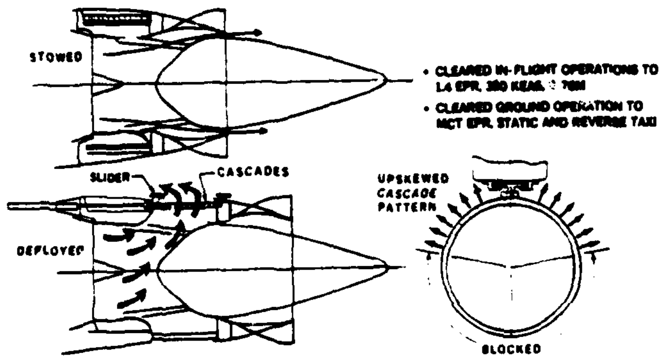
For Phase III testing a 2000 X 60 ft. dirt strip was utilized for STOL approaches/landings. Fifty STOL approaches (six degree glideslope) were made to landings with thirty-eight percent being one engine out. Figure 4-10 shows the test results. Automatic ground spoiler deployment with main wheel spinup permitted immediate anti-skid braking which resulted in maximum effort stops in 600-900 feet of ground roll. STOL approaches were conducted in up to forty-one knots of crosswind, with a thirty knot limit being established. The engine-out approaches/landings were conducted in the same manner as the all engine approaches and configuration changes were not required. No significant handling quality problems were discovered.

SOURCES: Ref. 2-3: '77-351,352
Refs. 4-8, 4-9, 4-10



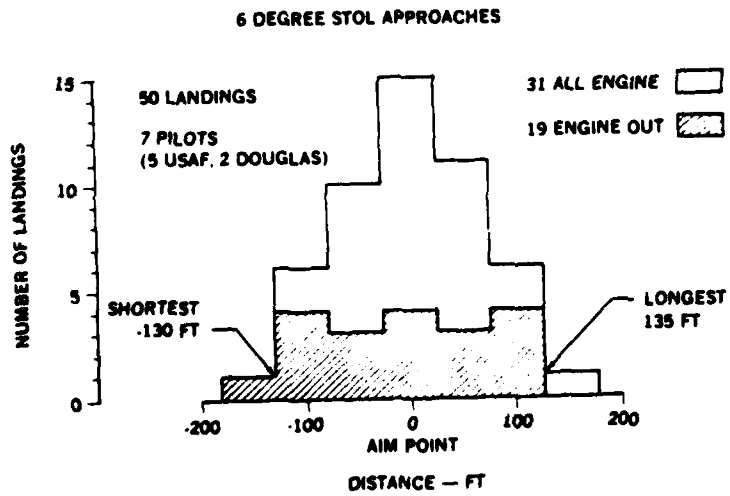
YC-15

FIGURE 4-8



YC-15 Thrust Reversers (Ref. 4-9)

FIGURE 4-9



STOL Landing Accuracy (Ref. 4-10)

FIGURE 4-10

V. UTILIZATION OF THE ACSYNT COMPUTER PROGRAM FOR A STOL MODIFICATION OF THE LOCKHEED S-3A "VIKING"

A. BRIEF OVERVIEW OF ACSYNT PROGRAM

The AirCraft SYNThesis (ACSYNT) computer program was developed by the NASA Ames Research Center, Moffet Field, California as a first generation approach to computerized aircraft design. In 1976 ACSYNT was installed on the Naval Postgraduate School's IBM 360/67 computer system by Dr. Garret Vanderplaats, one of the program's developers. The program was quite large and complex (requiring approximately 600 K bytes of memory) and was intended for conceptual design studies of various aircraft ranging from unmanned reconnaissance vehicles to large transports.

ACSYNT is modular in organization, as shown in Figure 5-1. As presently installed at this institution the program consists of the CONTROL (0), GEOMETRY (1), TRAJECTORY (2), AERODYNAMICS (3), PROPULSION (4), WEIGHTS (6), NAVY (10) and SUMMARY (11) modules.

ACSYNT employs two general purpose programs: COPES (COntrol Program for Engineering Synthesis) and CONMIN (CONstrained function MINimization). The former aids in the optimization, sensitivity analysis and two-variable function space modes while the latter is a subroutine utilized for design optimization. Further details of these two programs can be found in Ref. 5-1. Section II of Ref. 5-2 explains ACSYNT's five modes of operation.

ACSYNT'S MODULAR ORGANIZATION

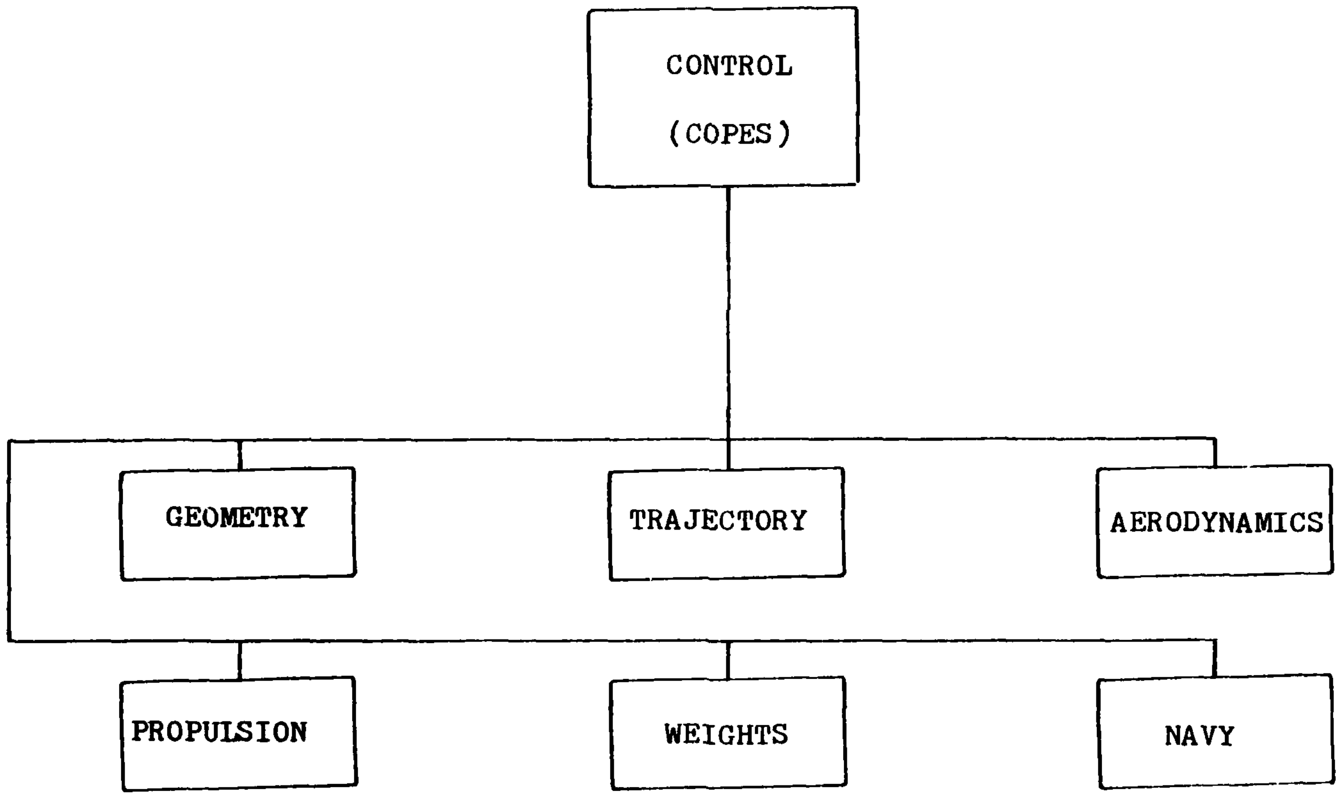


FIGURE 5-1

1. <u>Summary of ACSYNT Modules</u>

a. CONTROL:

This module coordinates the various design and analysis functions and controls the transfer of information between the various modules by means of a single labeled common block ("GLOBAL").

b. GEOMETRY:

Aircraft surface areas and volumes are calculated by this module. In addition, both plan view and profile view plots of the computer model are generated.

c. TRAJECTORY:

The function of this module is to determine the flight path of the aircraft, given a specific mission profile. Changes in aircraft weight are computed as the vehicle's flight conditions change and fuel is consumed. Both the takeoff balanced field length and landing distance are computed.

d. AERODYNAMICS:

The AERODYNAMICS module is based on compressible wing theory. It computes the coefficients of lift, minimum and induced drag, and pitching moment for wings and wing-body combinations, with or without a horizontal tail.

e. PROPULSION:

This module utilizes the performance data from one of five current jet engines (one turbojet and four turbofans), selected by the user, to size the aircraft's powerplant and then calculate its performance.

f. WEIGHTS:

The purpose of this module is to provide initial values of weights of major aircraft components. Known component/subsystem weights can be specified.

g. NAVY:

The NAVY module ensures compatibility between the proposed aircraft design and CV-59 class aircraft carriers.

h. SUMMARY:

This module provides a one-page compilation of pertinent output data from the aforementioned modules.

2. ACSYNT User's Manual

Reference 5-1 was utilized as the guide for all input/output data while working with the program. Reference 5-2 contained recommended changes/corrections to the User's Manual which this author incorporated into Ref. 5-1.

B. CORRELATION OF THE S-3A

1. Purpose

A correlation study was undertaken to compare the actual performance of the S-3A with ACSYNT predictions. Once the correlation was achieved, the resulting "baseline" aircraft was modified to a STOL configuration utilizing the Lockheed AIBF system. ACSYNT's developers have claimed an accuracy of within ten percent in predicting the gross weight of an aircraft.

2. Correlation Procedure

The S-3A "Viking" is a high-wing, twin turbofan powered, carrier based antisubmarine warfare (ASW) aircraft which carries a crew of four. Aircraft layout is shown in Figure 5-2. Both surface and subsurface search equipment is installed and an integrated target acquisition and sensor coordinating system collects, processes, interprets and stores ASW sensor data.

a. Input Data

Reference 5-2 contained correlations of the VAK-191B and the AV-8A Harrier. These examples proved extremely helpful in determining initially the minimum input data required and whether or not to override various default values within ACSYNT.

(1) <u>ACSYNT Control</u>. Since just the convergence mode of the program was utilized, only COPES data blocks A and B were required; NCALC = 1. Additional control data were input through ACSYNT data blocks A through F with WGMAX = 60,000 lbs (S-3A maximum takeoff weight = 52,500 lbs).

(2) <u>Geometry</u>. Aircraft dimensions and systems information were obtained from Ref. 5-3 and engineering drawings obtained from the Lockheed-California Company. Appendix A summarizes this input data.

(3) <u>Trajectory</u>. A high-low-high ASW mission was selected and is depicted in Figure 5-3. The engines were sized for a specific thrust level under static, sea level conditions; IPSIZE = -3.

(4) <u>Aerodynamics</u>. S-3A airfoil data was obtained from engineering drawings and Ref. 5-3 provided the flap settings for takeoff and landing maneuvers. Appendix A lists the input data.

S-3A VIKING

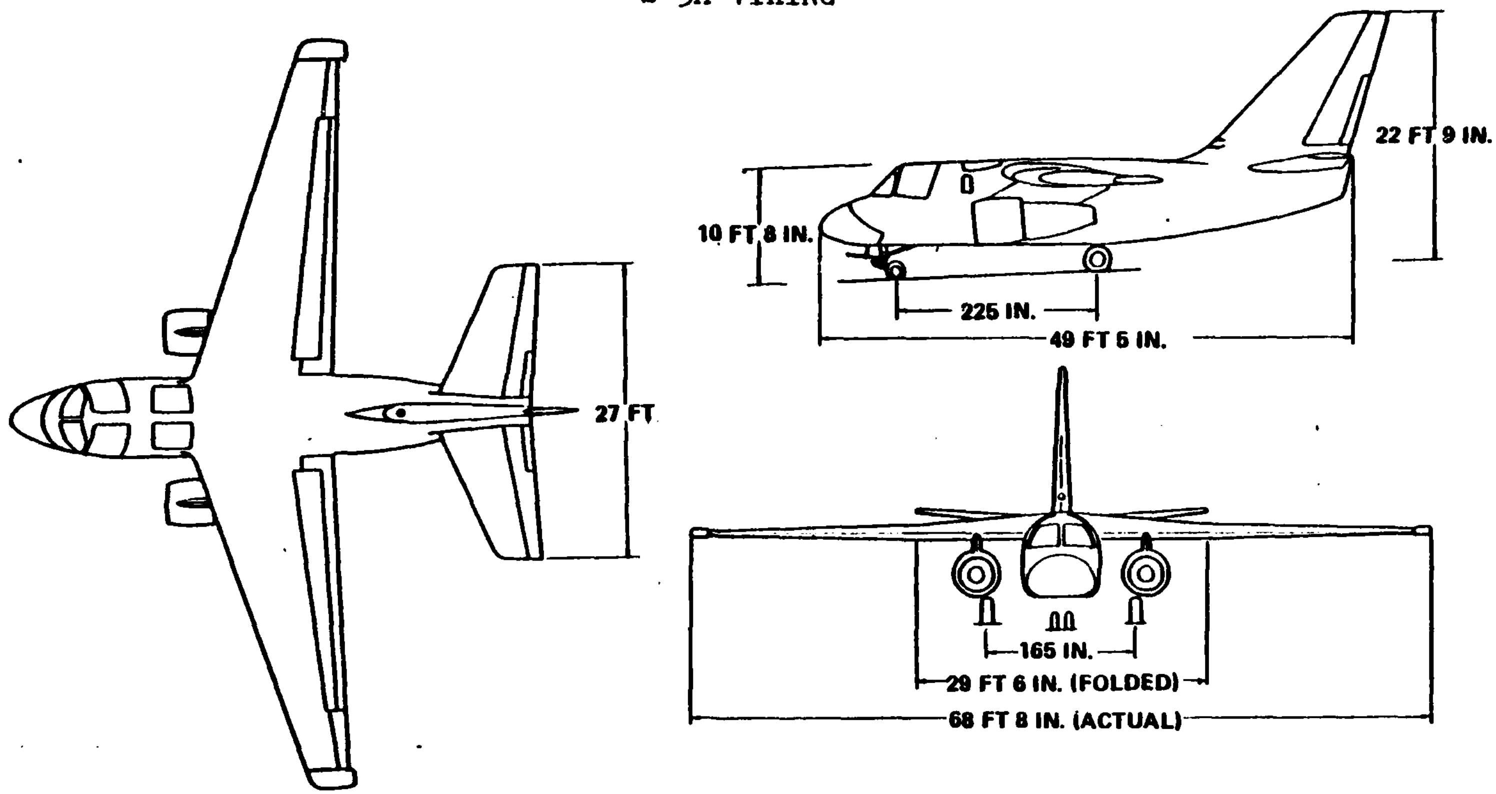


FIGURE 5-2 (Ref. 5-6)

S-3A Correlation Mission: High-Low-High Profile

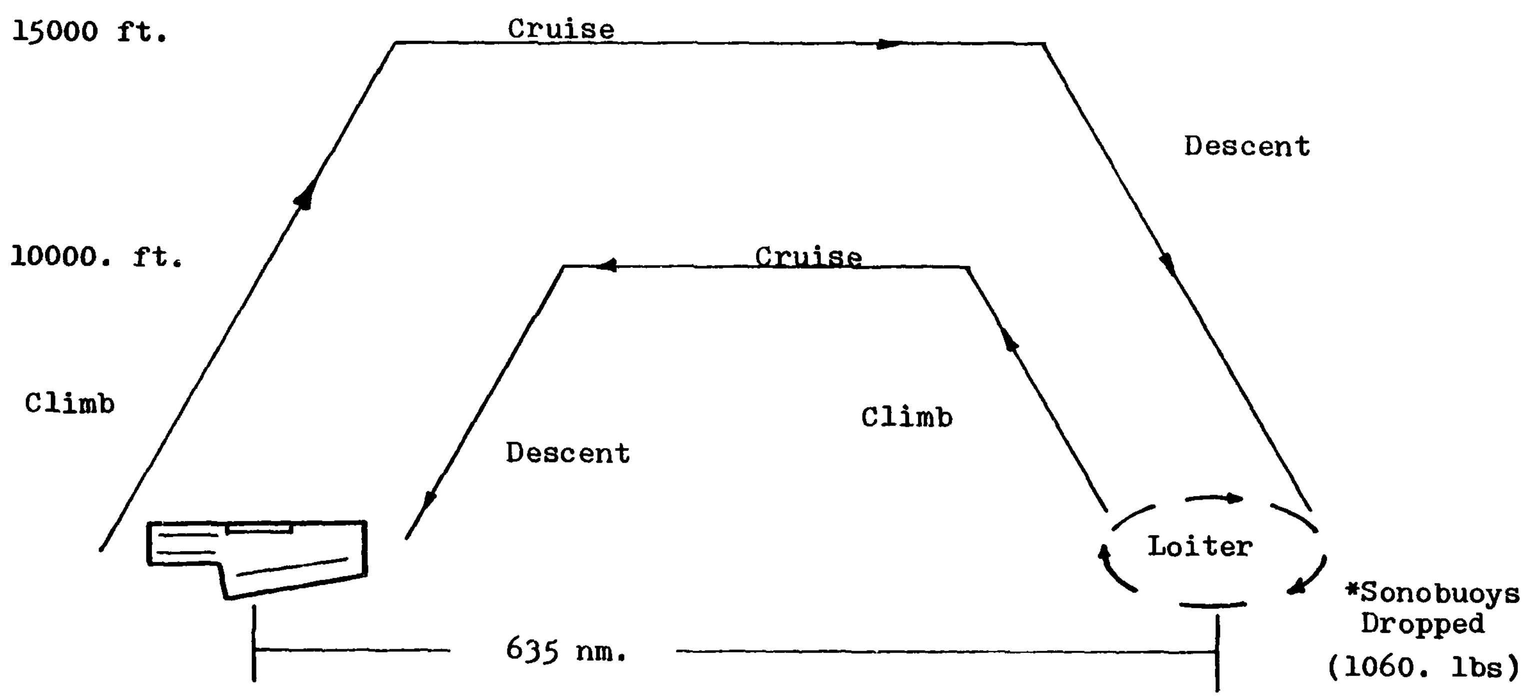


FIGURE 5-3

(5) Propulsion. Since the engine that powers the S-3A (TF34-GE-2 turbofan) was not one of the five contained in ACSYNT, the TF-30-P-100 (without afterburner) was "rubberized" to TF-34 specifications [Ref. 5-4] and sized to fly the specified flight profile. Specific fuel consumption (SFC) multiplying factors, SFSFC1 and SFSFC2, were employed to more closely model TF-34 performance (see Fig. 5-4). Appendix A lists this input data.

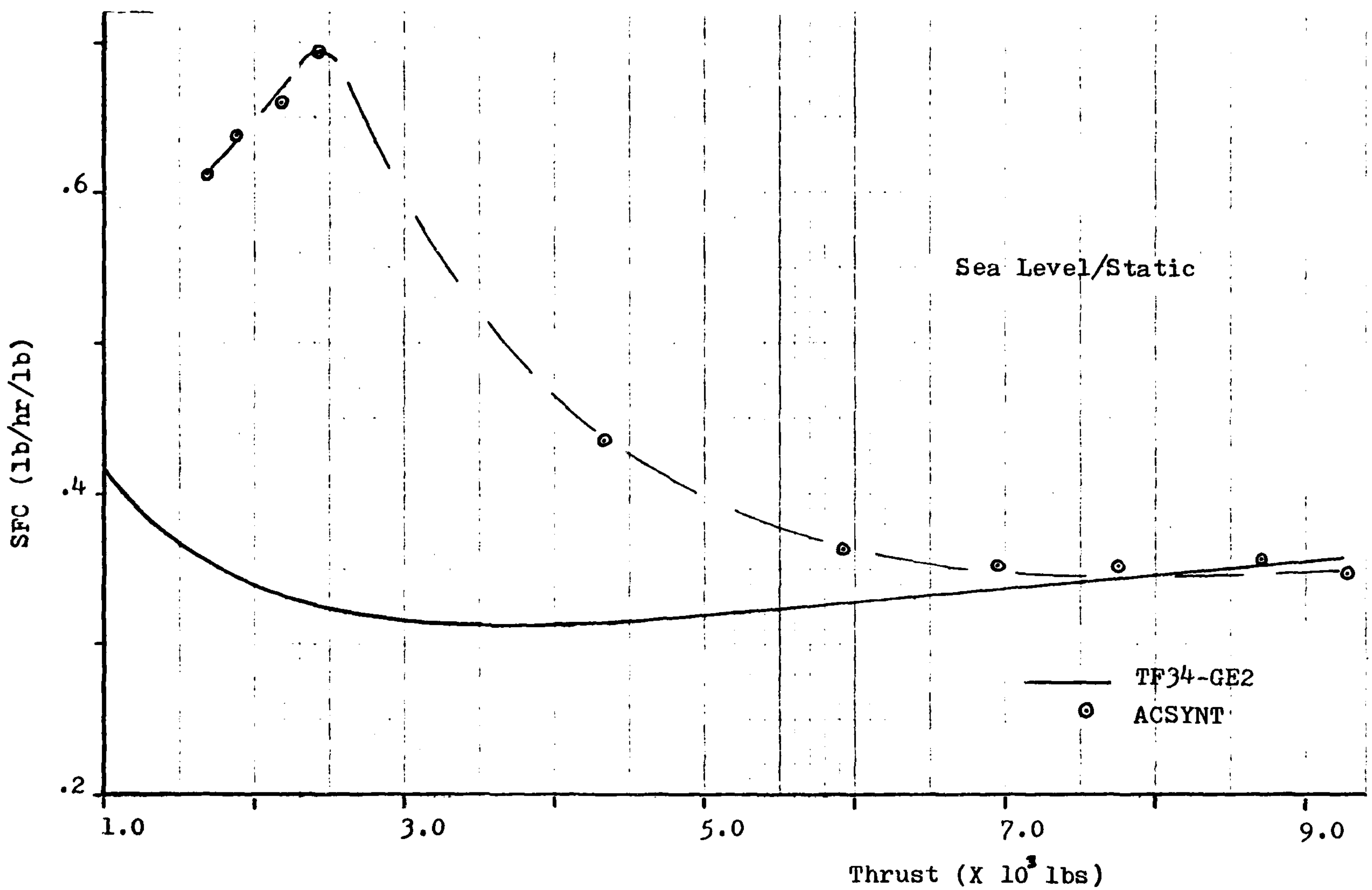
(6) Weights. S-3A subassembly/component weights were taken from Ref. 5-5. For the aircraft type requirement of namelist FIXW, "bomber" was selected. All FIXW variables were specified with the exception of WAF, WFEQ, WPL AND WTSUM. For the mission profile specified in TRAJECTORY, WAMMUN = 1,060 lbs and WGTO = 43,000 lbs. Appendix A lists the WEIGHTS data.

(7) The NAVY module was not utilized since the S-3A is a current operational carrier aircraft.

Appendix A contains the correlation study computer printout.

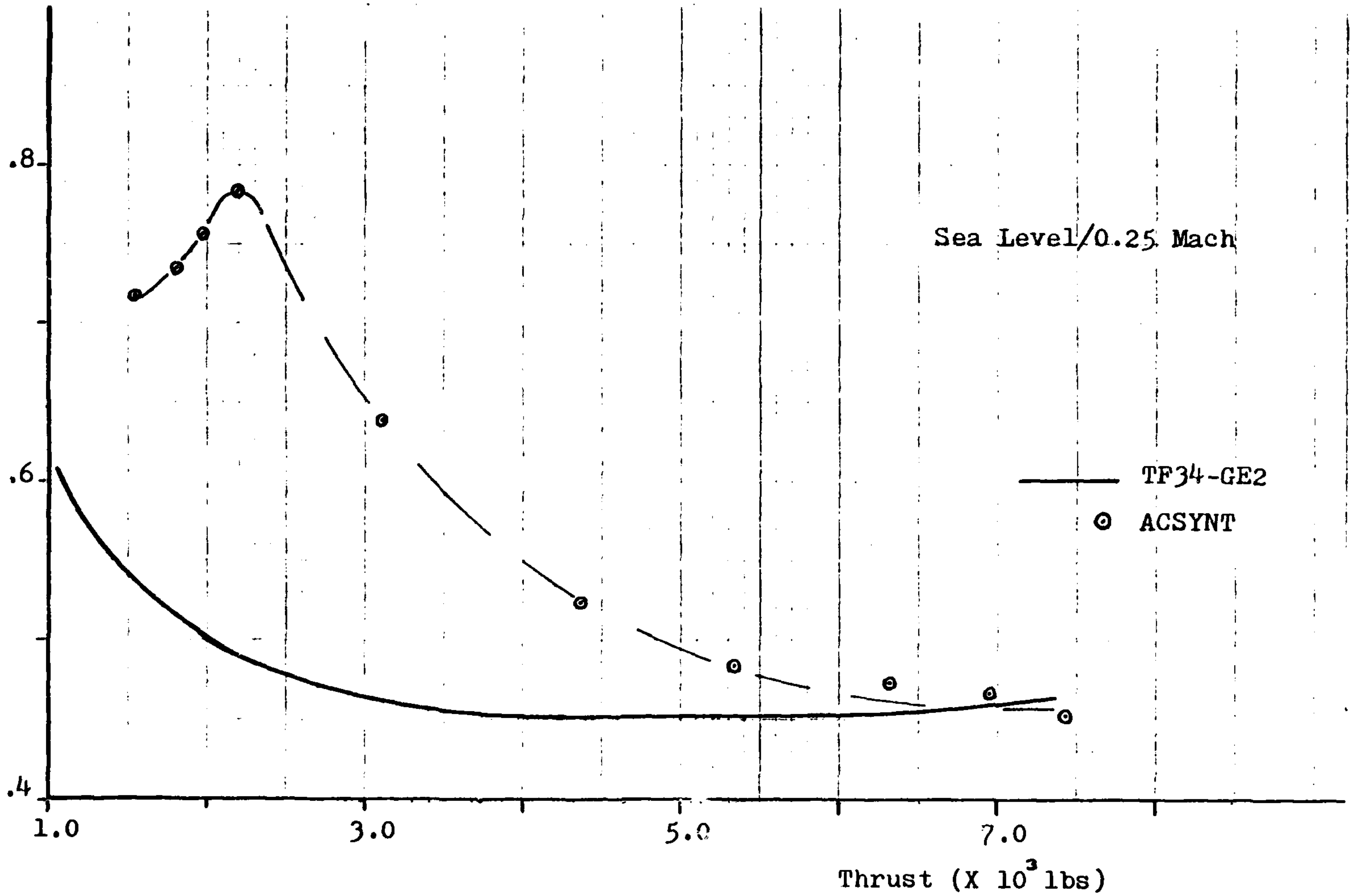
3. Accuracy of Correlation

A comparison of the ACSYNT performance prediction with the S-3A NATOPS manual [Ref. 5-3] fuel requirements for the specified mission is shown in Table 5-1. Total mission fuel as predicted by ACSYNT was within four percent of the NATOPS fuel required. The converged gross weight of the S-3A was only six and one half percent less than the actual aircraft for the selected mission. Figure 5-5 shows that ACSYNT consistently underestimates the total takeoff field length (distance to clear a 50 ft. obstacle) for the S-3A.



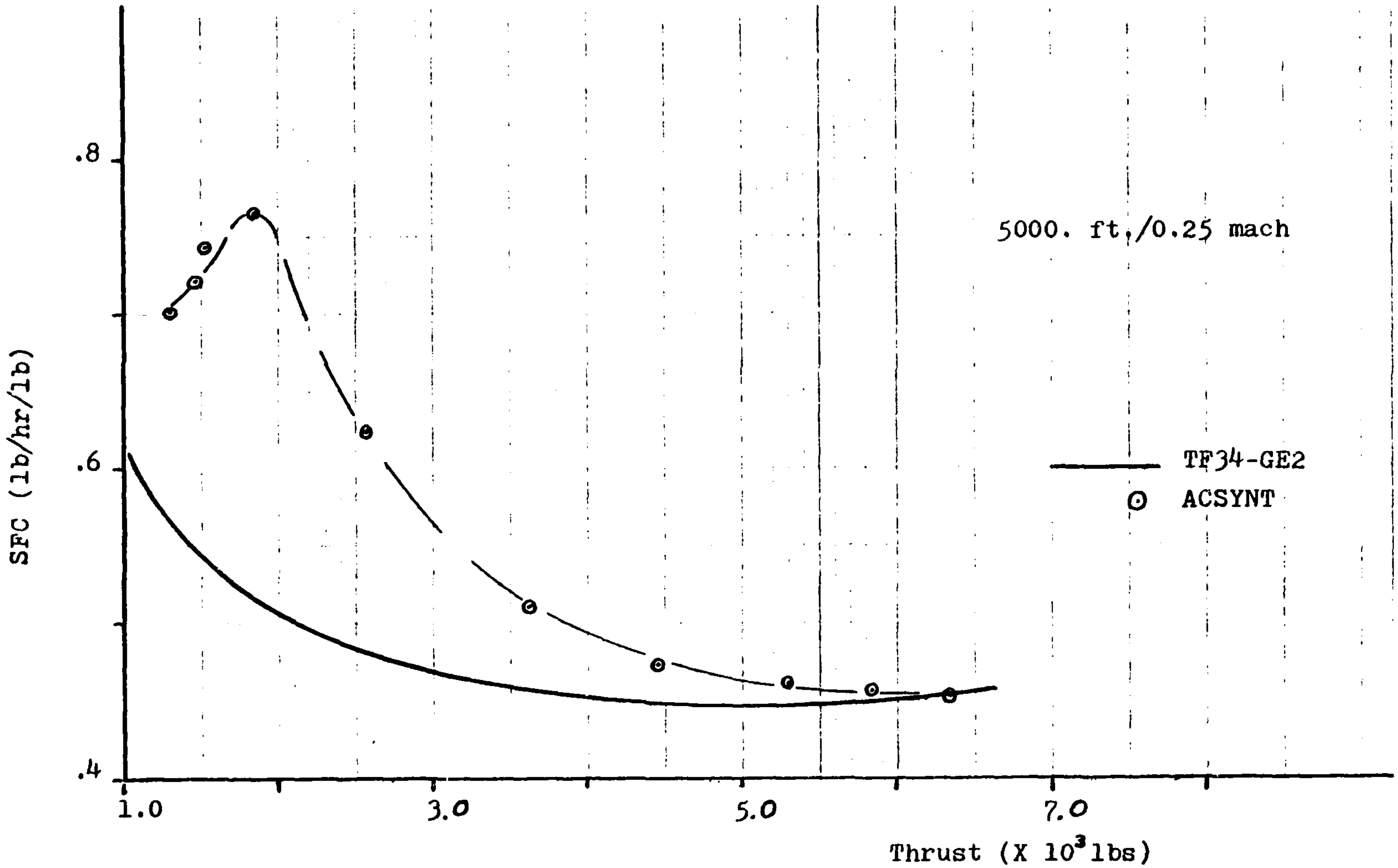
Sea Level/Static
TF34-GE2
ACSYNT
.6
.4
.2
SFC (lb/hr/lb)
1.0
3.0
5.0
7.0
9.0
Thrust (X 10³ lbs)

FIGURE 5-4(1)



.8
.6
.4
SFC (lb/hr/lb)
1.0
3.0
5.0
7.0
Thrust (X 10^3 lbs)
Sea Level/0.25 Mach
TF34-GE2
ACSYNT

FIGURE 5-4 (2)



5000. ft./0.25 mach
TF34-GE2
ACSYNT
SFC (lb/hr/lb)
.8
.6
.4
1.0
3.0
5.0
7.0
Thrust (X 10^3 lbs)

FIGURE 5-4 (3)

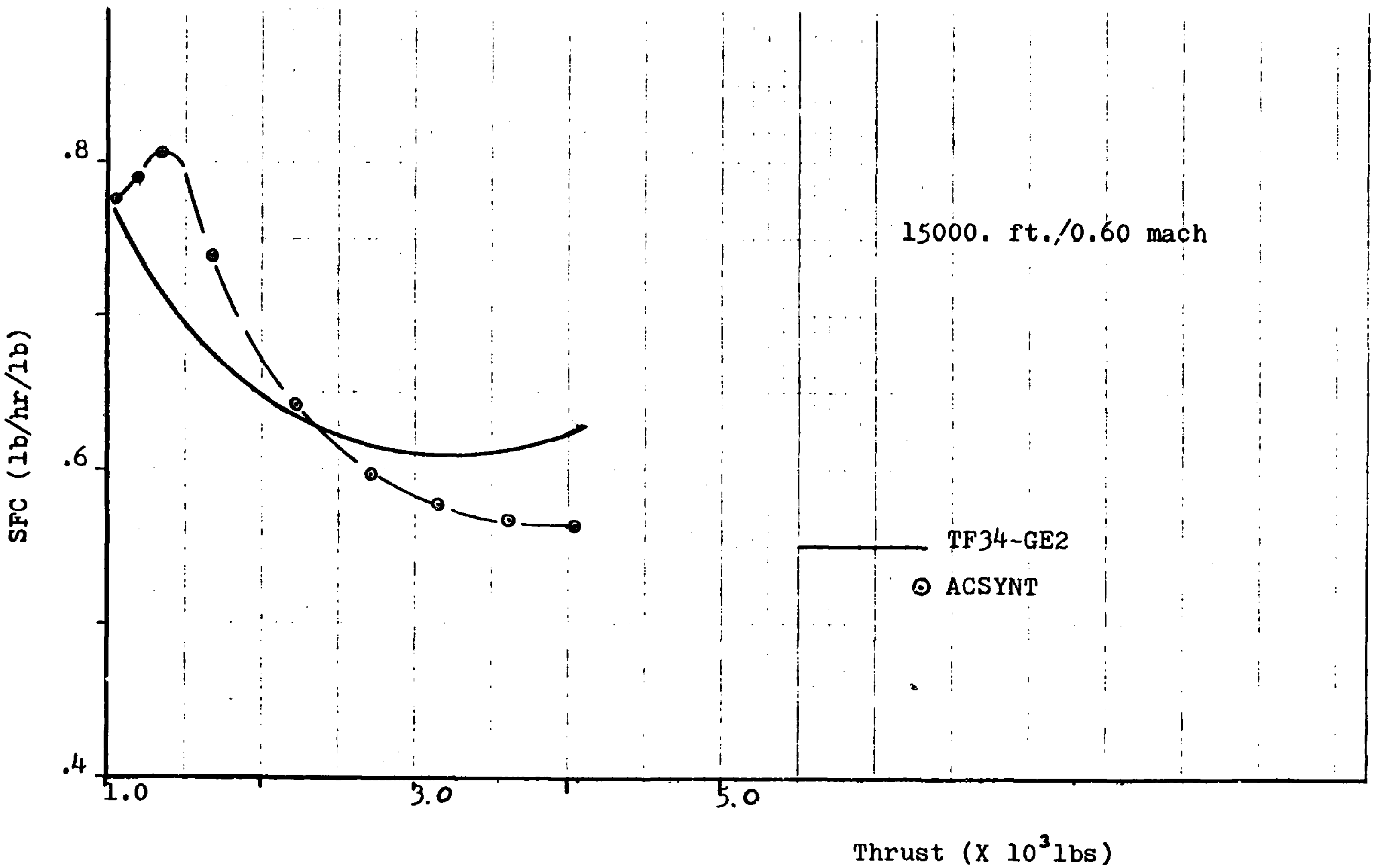

.8
.6
.4
SFC (lb/hr/lb)
1.0
3.0
5.0
Thrust (X 10³lbs)
15000. ft./0.60 mach
TF34-GE2
ACSYNT

FIGURE 5-4 (4)

S-3A FLIGHT PERFORMANCE COMPARISON

Trajectory Leg	Fuel Required (LBS.)		% Δ
	NATOPS (1)	ACSYNT (3)	
Start - Taxi - T/O - Accel. - Climb 1 - Accel.	850.0	549.9	- 35.3
Cruise out	4,254.0	4,180.6	- 1.7
Descent 1	30.0	-	(2)
Loiter 1	1,550.0	2,112.6	+ 36.3
Climb 2	200.0	174.4	- 12.8
Cruise back	4,233.0	4,487.1	+ 6.0
Descent 2	15.0	-	(2)
Loiter 2	115.0	161.4	+ 40.3
Total mission fuel	11,247.0	11,666.0	+ 3.7
Reserve fuel	1,897.0	614.0	- 67.6
*Trapped fuel	100.0	100.0	
Total fuel onboard	13,244.0	12,380.0	- 6.5

NOTES:

(1) Performance figured no wind, standard day.
Gross weight = 42,605. lbs

(2) ACSYNT descent leg inoperative

(3) ACSYNT converged weight = 41,741. lbs

TABLE 5 -1

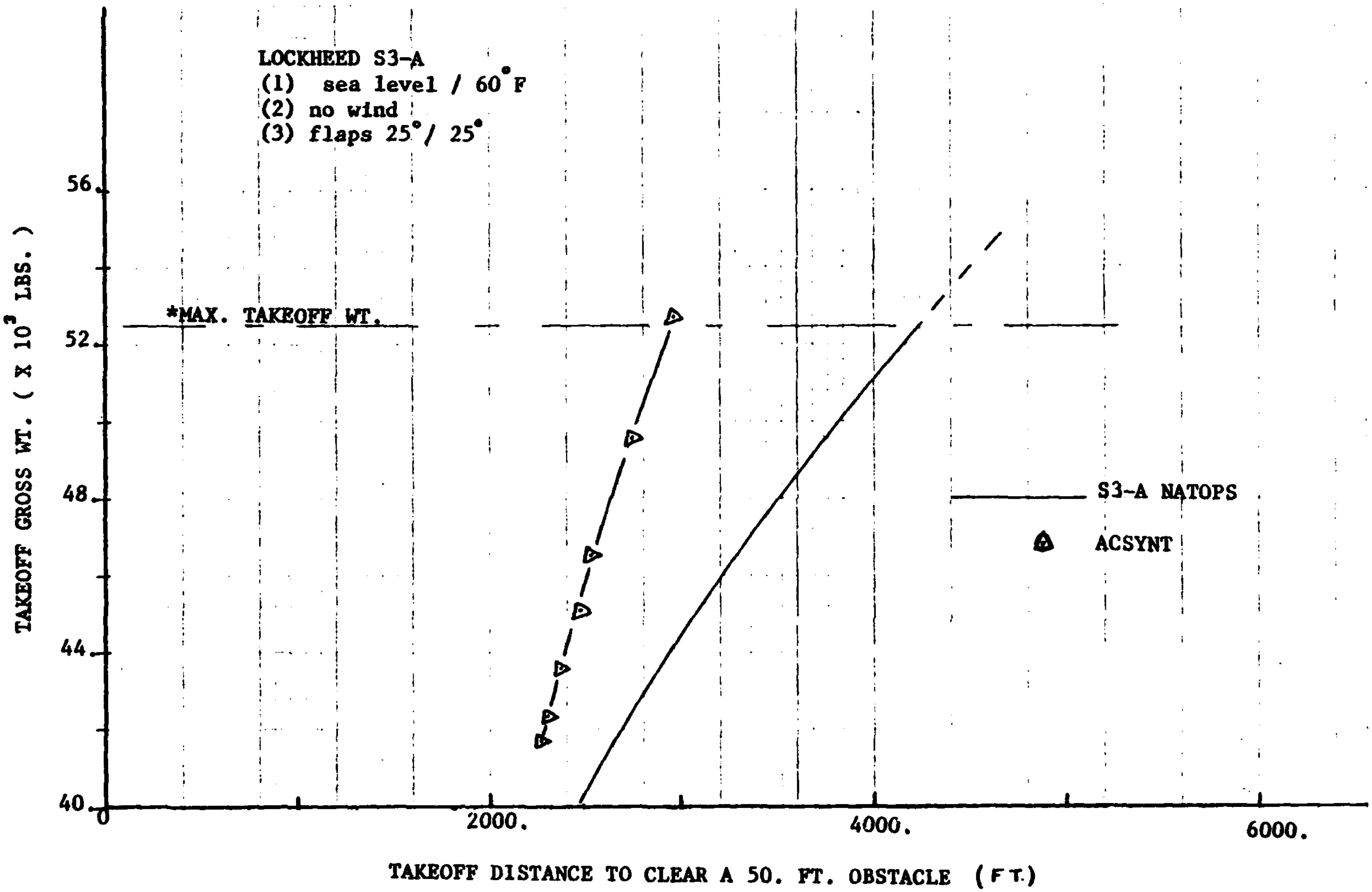

LOCKHEED S3-A
(1) sea level / 60°F
(2) no wind
(3) flaps 25°/ 25°
*MAX. TAKEOFF WT.
S3-A NATOPS
ACSYNT
56
52
48
44
40
0
2000.
4000.
6000.
TAKEOFF GROSS WT. (X 10³ LBS.)
TAKEOFF DISTANCE TO CLEAR A 50. FT. OBSTACLE (FT.)

FIGURE 5-5

C. STOL MODIFICATION OF THE S-3A

1. General

The "baseline" aircraft was modified to the STOL configuration proposed in Ref. 5-6. Figure 5-6 shows the proposed layout of the aircraft, hereafter referred to as the "STOL VIKING".

The U.S. Navy's interest in a multi-mission subsonic aircraft to satisfy several operational requirements was expressed in Ref. 2-6. The Lockheed S-3A, operational since 1975 as a carrier ASW asset, was selected [Ref. 5-6] as a STOL candidate primarily due to the cost savings and expected short-field performance to be accrued from such a modification. The proposed installation of the Lockheed AIBF system entails a modification of the internal wing structure and the addition of two more TF-34-GE-2 turbofan engines along with modified nacelles (Figure 5-6).

2. ACSYNT Input Data

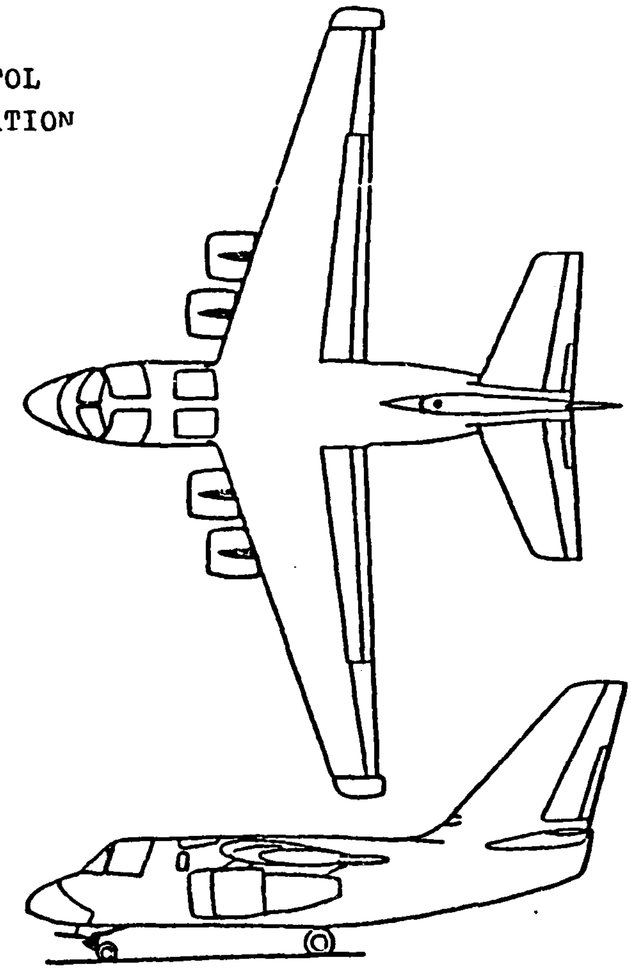
Specific details of the proposed subsystems/subassembly modifications to the S-3A were obtained from The Lockheed-California Company and alterations to the "baseline" aircraft input data are summarized below.

a. Geometry

The relocation of the horizontal tail and the additional two engines necessitated modifications to namelists HTAIL and WPOD. The fuselage, wing (fuel capacity) and vertical tail geometry remained unchanged.

S-3A STOL MODIFICATION

WING	AREA, FT^2 = 598 SPAN, FT = 68 ASPECT RATIO = 7.73 TAPER RATIO = 0.25 SWEEP (C/4), DEG = 15 MAC, IN = 118.24 ROOT CHORD, IN = 169 TIP CHORD, IN = 42
POWER PLANT	4 TF34-GE-2 TURBOFANS BYPASS RATIO (SLS) = 6.2 FAN PRESSURE RATIO (SLS) = 1.47 AIBF DUCT LOSS = 10% INST MAX THRUST (S.L.90°F) = 6841 LB/ENG



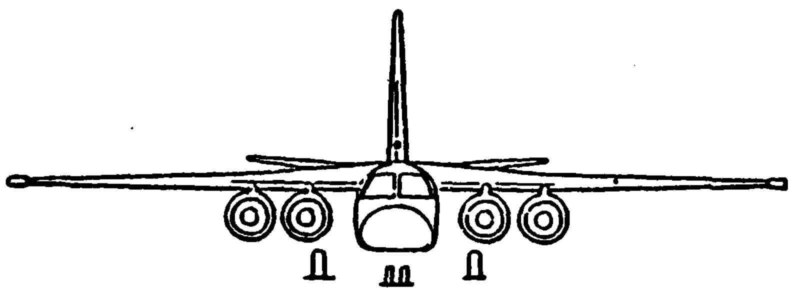


FIGURE 5-6 (Ref. 5-6)

b. Trajectory

The mission profile was altered only by a shortening of the cruise distance since onboard useable fuel was still 13,144 lbs. The "baseline" leg of 635 nm. was successively reduced until an acceptable mission-end fuel reserve was attained. It should be noted that the ACSYNT prediction of cruise performance is based on preliminary aerodynamic analysis only. The proposed STOL VIKING configuration has not been wind tunnel or flight tested. The purpose of utilizing ACSYNT was to look at predicted takeoff distance and to determine the compatibility of the TRAJECTORY module with design configurations utilizing propulsive-lift systems.

c. Aerodynamics

Leading and trailing edge flap settings (namelist IPDATA) were changed to conform to the AIBF configuration for both takeoff and landing. Variables CLTO, CLLAND, LDTO and LDLAND were defaulted.

d. Propulsion

As with the "baseline" aircraft, the TF-30-P100 was modified to TF-34 specifications and the maximum thrust available (TWOAB) was taken from Ref. 5-6. The powerplants were sized at sea level, static conditions (IPSIZE = - 3). Variables SFSFC1 and SFSFC2 were adjusted for the AIBF configuration.

e. Weights

Installation of the AIBF system entails a weight increase over the basic S-3A. New high lift devices, AIBF ducting, two additional engines and the horizontal tail modification results in an additional 4,403. lbs. of weight. For the specified mission profile WAMMUN = 1,060. lbs. and WGTO was set at 47,000. lbs.

Appendix B is the STOL VIKING computer printout which lists the input data.

3. Analysis of Takeoff Distance

a. ACSYNT Prediction

For the specified mission the converged weight of the STOL VIKING was 46,738. lbs. ACSYNT calculated a mission fuel weight of 12,231 lbs and a 644 lb reserve. The total takeoff field length was 2,152. ft. The reader is referred to the last page of Appendix B for a summary of aircraft and mission data.

The Trajectory module computes the takeoff balanced field length (variable FLTO) in subroutine TAKEOF [Ref. 5-7]. Reference 5-8 defines the <u>balanced field length</u> in terms of a "critical decision speed" at which the distance required to accelerate from rest and then stop, following the loss of an engine, becomes equal to the total takeoff distance to safely reach obstacle clearance height. It was noted in the above reference that the validity of the balanced field length/critical decision speed method had <u>not been</u> proven for STOL aircraft.

A hand calculation of the balanced field length (FLTO), as it appears in ACSYNT [Ref. 5-7], was performed and can be found in Appendix C.

b. First - Order Analysis

In Section III, page 77, an expression for the ground-run distance (S_G) was developed. This expression, when evaluated using the results of ACSYNT's STOL VIKING computer run, gives a first-order indication of the aircraft's STOL capability. The computation can be found in Appendix D.

c. Lockheed's STOL Prediction

Reference 5-6 contains the predicted takeoff performance of the AIBF-modified S-3A. For a wing loading of 78.2 lbs/ft^2, an AIBF flap setting of 30 degrees and the control flap deflections (δ_c) shown below, the predicted ground-run distances are:

δ_c (Degrees)	Distance (Ft.)
0	535.
10	400.
20	325.
30	280.

NOTES:

(1) S.L., 90 F., no wind.

(2) δ_c applied at V_{LO}.

(3) AIBF setting = 30 degrees

(4) $V_{LO} = 1.20 \, V_{Stall}$

d. Comparison of Results

The balanced field length equation developed in Ref. 5-8, and utilized in ACSYNT, is based on several assumptions in which empirical data, from a study of conventional takeoff and landing (CTOL) transport aircraft, is used. From Table 5-2, the calculation of balanced field length by ACSYNT results in a total takeoff run (from at rest to clearance of a 50 feet obstacle) in excess of 2100. feet. The hand calculation of Appendix C, using a value of $C_{L_{(Liftoff)}}$ derived from Lockheed data, yielded a total takeoff run which was approximately 18 percent less than the ACSYNT distance. This difference is due to the value of $C_{L_{(Liftoff)}}$ computed by the program since the variable CLTO was defaulted (see AERODYNAMICS data). Lockheed's STOL ground-run predictions [Ref. 5-6] seem reasonable when compared to the simple first-order analysis of Appendix D.

COMPARISON OF STOL VIKING TAKEOFF DISTANCES

Data from Appendices B and C:

$(W/S) = 78.2$ lb/sq.ft.

$(T/W) = 0.59$

$V_{LO} = 73.$ kts.

$C_{L_{(LO)}} = 4.31$

Calculation Method	Distance (Ft.)
ACSYNT (Appendix B)	2152. (1)
Balanced Field Length Equation (Ref. 5-7)	1772. (1)
First-Order Analysis (Appendix D)	402. (2)
Lockheed Prediction (Ref. 5-6)	535. (2), (3)

NOTES:

(1) Total distance to clear 50. ft. obstacle.

(2) Ground-run distance only.

(3) Sea Level, 90°F.

TABLE 5-2

D. CONCLUSIONS

The correlation study of the S-3A VIKING substantiated the claim that the gross weight of an aircraft could be predicted within ten percent by ACSYNT. In the computation of total takeoff run distance though, ACSYNT values were consistently less than S-3A NATOPS distances for the same takeoff conditions.

The STOL modification of the "baseline" aircraft demonstrated the flexibility of the ACSYNT program. Although the conclusion was drawn in Reference 5-2 that V/STOL performance could be satisfactorily predicted by the program, in its present form ACSYNT is incapable of computing accurate takeoff distances for aircraft incorporating powered high lift systems. Modifications to the TRAJECTORY module, presently being developed by NASA Ames, are required to take into account the jet-induced aerodynamic effects associated with such systems.

ACSYNT is a highly versatile aircraft synthesis program, which if continuously updated to account for technological advances, will prove to be even a more powerful aid in future design analysis.

APPENDIX A

S-3A Correlation Computer Printout

COPES

C O N T R O L   P R O G R A M

F O R

E N G I N E E R I N G   S Y N T H E S I S

T I T L E

** LOCKHEED S-3A "VIKING" - AIRCRAFT CORRELATION **

CARD IMAGES OF CONTROL DATA

```
CARD                    IMAGE

  1)  $          CHRCS CONTROL CARDS
  2)  ** LOCKHEED S-3A "VIKING" - AIRCRAFT CORRELATION **
  3)  $          BLOCK - A
  4)             1         0         0         0         0         0         0
  5)  $          BLOCKS C-I (DESIGN VARIBL. OPTIMIZATION) OMITTED
  6)  $          BLOCKS J.K (SENSITIVITY EVAL.) OMITTED
  7)  $          BLOCKS L-O OMITTED, N2VAR = 0
  8)  END
```

```
TITLE:
** LOCKHEED S-3A "VIKING" - AIRCRAFT CORRELATION **

CONTROL PARAMETERS:
CALCULATION CONTROL,                        NCALC  =    1
NUMBER OF GLOBAL DESIGN VARIABLES,           NDV   =    0
NUMBER OF SENSITIVITY VARIABLES,             NSV   =    0
NUMBER OF FUNCTIONS IN TWO-SPACE,           N2VAR  =    0
INPUT INFORMATION PRINT CODE,              IPNPUT  =    0
SENSITIVITY PRINT CODE,                    IPSENS  =    0
TWO-SPACE PRINT CODE,                      IP2VAR  =    0
DEBUG PRINT CODE,                           IPDBG  =    0

CALCULATION CONTROL, NCALC
VALUE    MEANING
  1      SINGLE ANALYSIS
  2      OPTIMIZATION
  3      SENSITIVITY
  4      TWO-VARIABLE FUNCTION SPACE

DATA STORAGE REQUIREMENTS

            REAL                              INTEGER
  INPUT  EXECUTION  AVAILABLE       INPUT  EXECUTION  AVAILABLE
    9        9        5000            1        1        1000
```

```
AAAAAAA   CCCCCCC   SSSSSSS   Y     Y   N     N   TTTTTTT
A     A   C         S          Y   Y    NN    N      T
A     A   C         S           Y Y     N N   N      T
AAAAAAA   C         SSSSSSS      Y      N  N  N      T
A     A   C               S      Y      N   N N      T
A     A   C               S      Y      N    NN      T
A     A   CCCCCCC   SSSSSSS      Y      N     N      T
```

N A S A - A M E S   P R O G R A M

F O R

A I R C R A F T   S Y N T H E S I S

NAVAL POSTGRADUATE SCHOOL VERSION 9 - 77
MONTEREY CALIFORNIA

T I T L E

** LOCKHEED S-3A "VIKING" - AIRCRAFT CORRELATION **   G=<E?Q? A

AIRCRAFT TYPE - BOMBER(ASW)

```
TITLE:
** LOCKEED S-3A "VIKING" - AIRCRAFT CORRELATION **            <E?07
                                                            G=      A

AIRCRAFT TYPE - BOMBER(ASW)

CONTROL PARAMETERS:
CALCULATION CONTROL,                          NCALC  =    2
READ CONTROL,                                 NREAD  =    5
EXECUTION CONTROL,                            NEXEC  =    3
WRITE CONTROL,                               NWRITE  =    5
NUMBER IDENTIFYING CONVERGENCE
  VARIABLE FOR CONVERGED VEHICLE,      IOBJ  =  570
NUMBER IDENTIFYING COMPARISON
  VARIABLE FOR CONVERGED VEHICLE,      JOBJ  =  585
SUMMARY OUTPUT PRINT CODE,                   IPSUM  =    0
GLOBAL ERRORS PRINT CODE,                   KGLOBP  =    0
GLOBAL COMMON INITIALIZATION CODE,           INIT   =    0
DEBUG PRINT CODE,                           IPDBG  =    2
GLOBAL PLOT CONTROL,                         IGPLT  =   -1
DATA TRANSFER INFORMATION FILE, IRDDTR      =    7
DATA TRANSFER INFORMATION PRINT, IPDTR      =    0

VEHICLE CONVERGENCE INFORMATION:
CONVERGENCE TOLERANCE, TOL = 0.10000E-03
ESTIM WCALC VS WEXT SLOPE  = 0.75000E 00
BOUNDING WEIGHT, WGMAX     = 0.60000E 05

MODULE IDENTIFICATION NUMBERS:

 NUMBER        MODULE
    1          GEOMETRY
    2          TRAJECTORY
    3          AERODYNAMICS
    4          PROPULSION
    5          STRUCTURES
    6          WEIGHTS
    7          CARGO
    8          GLOBAL I/C
    9          ECONOMICS
   10          NAVY

MODULES ARE CALLED FOR INPUT IN THE FOLLOWING ORDER:
    1     2     3     4     6

MODULES ARE CALLED FOR EXECUTION IN THE FOLLOWING ORDER:
    1     2     6

MODULES ARE CALLED FOR OUTPUT IN THE FOLLOWING ORDER:
    1     2     3     4     6
CALLING MODULE NUMBER  1
```

```
GEOMETRY DATA CARDS                    ACSYNT VERSION  4-76
 1** S-3A "VIKING" - BASIC A/C GEOMETRY **
 2 &WING  AR=7.73, AREA=598.0, ELLAPC=0.10, SWEEP=15.0, TAPER=0.25,
 3 TCROOT=0.163, TCTIP=0.120, TELAPC=0.254, XWING=0.398, ZROOT=0.628,
 4 WFFAC=0.579,  KSWEEP=1,      &END
 5 &HTAIL  AR=4.05, AREA=140.0, SWEEP=20.0, TAPER=0.43, XHTAIL=0.983,
 6 ZROOT=0.0, KSWEEP=1, SIZIT=F,     &END
 7 &VTAIL    AR=1.435, AREA=129.0, SWEEP=38.5, TAPER=0.35, TCROOT=0.12,
 8 ZROOT=1.053, KSWEEP=1, SIZIT=F,    &END
 9 &WPOD    X=-0.364, Y=0.226, Z=-0.485,      &END
10 &FUS    BDMAX=7.75, BCOL=49.25, FRAB=3.033, FRN=1.022,   &END
11 &ELEC    LENGTH=0.0,   &END
12 &ELEC    LENGTH=9.17,    &END
13 &CREW  LENGTH=13.42, WIDTH=7.167, NCREW=4,    &END
14 &FUEL    FFAC=0.0, WFUEL=13144.0,    &END

END OF GEOMETRY DATA CARDS
  14 CARDS READ
```

NASA/AMES A C S Y N T VERSION 4-76 INITIAL OUTPUT

FUSELAGE DEFINITION

X	R	AREA
0.0	0.0	0.0
0.98	1.30	5.31
1.97	2.08	13.59
2.95	2.66	22.31
3.94	3.11	30.38
4.92	3.45	37.42
5.91	3.69	42.69
6.89	3.83	45.95
7.88	3.87	47.17
8.86	3.88	47.17
9.85	3.88	47.17
10.83	3.88	47.17
11.82	3.88	47.17
12.80	3.88	47.17
13.79	3.88	47.17
14.77	3.88	47.17
15.76	3.88	47.17
16.74	3.88	47.17
17.73	3.88	47.17
18.71	3.88	47.17
19.70	3.88	47.17
20.68	3.88	47.17
21.67	3.88	47.17
22.65	3.88	47.17
23.64	3.88	47.17
24.63	3.88	47.17
25.61	3.88	47.17
26.59	3.87	47.03
27.58	3.86	46.74
28.56	3.83	46.16
29.55	3.80	45.33
30.53	3.75	44.28
31.52	3.70	42.97
32.50	3.63	41.44
33.49	3.55	39.70
34.47	3.47	37.76
35.46	3.37	35.67
36.44	3.26	33.30
37.43	3.13	30.81
38.41	3.00	28.19
39.40	2.85	25.44
40.38	2.68	22.59
41.37	2.50	19.67
42.35	2.31	16.71
43.34	2.09	13.75
44.32	1.86	10.84
45.31	1.60	8.03
46.29	1.31	5.40
47.28	0.98	3.04
48.26	0.59	1.11
49.25	0.0	0.0

FUSELAGE

MAX. DIAMETER........ 7.750
FINENESS RATIO...... 6.355
SURFACE AREA........ 999.015
VOLUME................ 1715.971

NASA/AMES A C S Y N T VERSION 4-76 INITIAL OUTPUT

DIMENSIONS OF PLANAR SURFACES

	WING	H.TAIL	V.TAIL	CANARD	UNITS
PLAN AREA............	593.0	180.0	129.0	0.0	(SQ.FT.)
SURFACE AREA........	1042.2	228.4	257.6	0.0	(SQ.FT.)
VOLUME...............	615.2	88.4	104.7	0.0	(CU.FT.)
SPAN.................	67.989	27.000	13.606	0.0	(FT.)
L.E. SWEEP...........	19.064	26.615	43.926	0.0	(DEG.)
C/4 SWEEP............	15.000	22.000	38.500	0.0	(DEG.)
T.E. SWEEP...........	2.019	3.931	16.265	0.0	(DEG.)
ASPECT RATIO	7.730	4.050	1.435	0.0	
ROOT CHORD...........	14.073	9.324	14.046	0.0	(FT.)
ROOT THICKNESS......	27.526	11.189	20.227	0.0	(IN.)
ROOT T/C	0.163	0.100	0.120	0.0	
TIP CHORD............	3.518	4.009	4.916	0.0	(FT.)
TIP THICKNESS.......	5.066	4.811	5.499	0.0	(IN.)
TIP T/C	0.120	0.100	0.100	0.0	
TAPER RATIO	0.250	0.430	0.350	0.0	
MEAN AERO CHORD.....	9.351	7.073	10.214	0.0	(FT.)
LE ROOT AT..........	16.083	39.069	35.204	0.0	(FT.)
C/4 ROOT AT.........	19.601	41.420	38.715	0.0	(FT.)
TE ROOT AT..........	30.156	48.413	49.250	********	(FT.)
LE M.A.C. AT........	20.782	41.795	40.704	********	(FT.)
C/4 M.A.C. AT.......	23.245	43.550	43.258	********	(FT.)
TE M.A.C. AT.......	30.633	48.815	50.918	********	(FT.)
Y M.A.C. AT.........	13.593	5.853	5.711	********	
LE TIP AT...........	27.331	45.331	48.209	0.000	(FT.)
C/4 TIP AT..........	26.710	46.333	49.538	0.000	(FT.)
TE TIP AT...........	31.349	49.340	53.225	-0.439	(FT.)
ELEVATION...........	2.432	0.0	4.235	0.0	(FT.)
VOLUME COEFF.		0.620	0.063	0.0	

F U E L T A N K S

TANK	VOLUME	WEIGHT	DENSITY
WING	263.	13134.	50.00
FUS#1	0.	0.	50.00
FUS#2	0.	0.	50.00
TOTAL		13134.	

MISSION FUEL REQUIRED= 13144.

AVAILABLE FUEL VOLUME IN WING= 454.

CALLING MODULE NUMBER 2

TRAJECTORY INPUT

TIMTO1 = 5.0	AENDUR = 0.	NCRUSE = 2	IPLOT = 0
TIMTO2 = 1.0	JMAX = 700.	IPSIZE =-3	HMINP = 0.
FREJPF = 0.05	ADESC = 40.0	IPSTO1 = 5	HMAXP = 40000.
DESLF = 3.50	WKFUEL = 1.000	IPSTO2 = 2	DELHP = 4000.
ULTLF = 5.25	CRMACH = 0.590	IPREG = 0	SMMINP = 0.300
RANGE = 1500.	WKLAND = 0.570	IENDUR = 0	SMMAXP = 0.900
WFUEL = 13144.	FLFAC = 1.000	IPRINT = 0	DELMP = 0.100
WFEXT = 0.	DECFL = 0.250	KERROR = 0	WCOMBP = 0.70
WFTRAP = 100.	NLEGCL = 0	NLEGCR = 0	NLEGLO = 0
ENGMAX = 1.200	TOL = 0.001	MTLCOM = 0	NMISS = 1

MISSION 1

PHASE	MACH START	MACH END	ALT START	ALT END	HORIZ DIST	TIME	NO. TURNS	VIND	IP IPRT	IX IB	IW IA
ACCEL	0.23	0.33	0.	500.	0.0	0.0	0.0	0.0	3 0	-1 0	0 0
CLIMB	0.33	0.44	500.	15000.	0.0	0.0	0.0	0.0	3 0	-1 0	0 0
ACCEL	0.44	0.59	-1.	15000.	0.0	0.0	0.0	0.0	3 0	-1 0	0 0
CRUISE	0.59	0.59	15000.	-1.	635.0	0.0	0.0	0.0	4 0	-1 0	0 1
LOITER	0.34	0.34	100.	100.	0.0	60.0	0.0	0.0	4 0	0 0	0 0
CLIMB	0.33	0.40	-1.	10000.	0.0	0.0	0.0	0.0	3 0	-1 0	0 0
CRUISE	0.34	0.34	10000.	-1.	635.0	0.0	0.0	0.0	4 0	-1 0	0 0
LOITER	0.34	0.0	100.	100.	0.0	5.0	0.0	0.0	4 0	0 0	0 0

CALLING MODULE NUMBER 3

```
                     AERODYNAMIC INPUT DATA

          ** S-3A "VIKING" - BASIC A/C AERODYNAMICS **

     XCG =     0.6000    FCDWB =     1.0000    FALTFL =    1.0000
     AMC =    40.0000      FXC =     2.0000      FXA=      0.6667
   SMNDP =     0.5000     FSFP =     1.0000     FCDW =     1.0000
    FCDF =     1.0000    FRLMT =     0.0100    FINTF =     1.0000
  FLSCDF =     1.0000   FLBCCF =     1.0000     FCPL =     1.0000
     FCL =     1.0000     FLNG =     1.0000  FLNOSE =     1.0000
  FDNOSE =     1.0000   FLECDP =     1.0000     FCDO =     1.0000
     FCD =     1.0000      FLD =     1.0000    FTRIM =     1.0000
    FCLH =     1.0000    FCDLH =     1.0000     FMCR =     1.0000
      YO =     0.0          IT =     0.0      GHOOT =     1.0000
    SFWF =     1.0000   FCLMAX =     1.0000   CLLAND =    -1.0000
  LDLAND =    -1.0000     CLTO =    -1.0000     LDTO =    -1.0000
  MACHIN =     0.7500   SWPMIN =     0.0      SWPMAX =    60.0000
  DELFTO =    25.0000   DELLTO =    25.0000   DFLFLD =    35.0000
  DELLFD =    25.0000     FLEX =     0.0        FMEX =     0.0
     SHK =    25.0000    ABOSA =     0.1500      CSF =     0.0
    ESSF =     1.0000     RTEF =     1.0000       SM =     0.0
     CGM =     0.0        FEXP =     0.7800      ROC =     0.0200

    IALP = 2      INTM = 0     I(CD) = 1       NALF =10     NMOTL = 6
    IALF = 0     IPLOT = 0     ALFLJ = 2     ISMNDP = 0    ISUPCR = 1
   INORM = 1    ECHOIN = 1    ECHOUT = 0     KERROR = 0      IXCD = 0
    IAXL = 0    IPEFIC = 0     IPENG = 0     IPLAVE = 0     IPEXT = 0
  IP3INT = 0    IPINTF = 0    IPLIFT = 0      IPMIN = 0     ITRAP = 0
   IFLEX = 0    ELLIPW = F    ELLIFH = F     ELLIPC = F

     SMN =          0.330          0.340          0.350          0.440          0.590
                    0.340          1.200          1.400          1.600          1.800
    ALTV =          1000.          5000.         10000.         15000.         15000.
                     100.         30000.         30000.         30000.         30000.
    ALIN =          0.0            0.500          1.000          1.500          2.000
                    4.000          6.000          8.000         10.000         12.000
   ISTRS =     0    0    0    0    0    0    0    0    0    0
     ITS =     0    0    0    0    0    0    0    0    0    0
     ITB =     0    0    0    0    0    0    0    0    0    0
   ITRIM =     1    1    1    1    1    1    0    0    0    0
  SMTANK =          0.0            0.200          0.400          0.600          0.800
                    1.000          1.200          1.400          1.600          1.800
   CDTNK =        0.0            0.0            0.0            0.0            0.0
                  0.0            0.0            0.0            0.0            0.0
  SMSTRS =          0.0            0.200          0.400          0.600          0.800
                    1.000          1.200          1.400          1.600          1.800
   CDSTR =        0.0            0.0            0.0            0.0            0.0
                  0.0            0.0            0.0            0.0            0.0
  SMNPMB =          0.0            0.200          0.400          0.600          0.800
                    1.000          1.200          1.400          1.600          1.800
   CDBMB =        0.0            0.0            0.0            0.0            0.0
                  0.0            0.0            0.0            0.0            0.0
     CLO =        0.0            0.0            0.0            0.0            0.0
                  0.0            0.0            0.0            0.0            0.0
     CMO =        0.0            0.0            0.0            0.0            0.0
                  0.0            0.0            0.0            0.0            0.0
  SMISWP =          0.0            0.200          0.400          0.600          0.800
                    1.000          1.200          1.400          1.600          1.800
    FEFF =          1.000          1.000          1.000          1.000          1.000
                    1.000          1.000          1.000          1.000          1.000
    CALLING MODULE NUMBER  4
```

```
                    PROPULSION INPUT
                     VERSION 04-76

** S-3A "VIKING" - PROPULSION ( TF-30 MODIF. ) **

AENDIA =      4.167  AENLE  =       8.675  AENWT  =  1425.000
ALTI   =      0.0    AM     =       0.0    ATURB  =     0.385
AUXENG =      0.0    AWAENG =       0.0    BA     =     6.230
DELPR  =      0.0    DELT57 =     100.000  DPRWCC =     2.000
DIAL   =      4.167  FAN1   =       0.750  FP1    =     0.940
FO1    =      0.750  ETAC1  =       0.800  ETAF1  =     0.900
ETATL  =      0.920  HTR    =       0.400  HVF    = 18600.000
MACH1  =      0.450  MACH2  =       0.500  PCDFAC =     1.000
POSA   =      8.903  PREQ   =       1.510  PNCF   =   100.000
P11P1  =      9.600  P2P1   =      14.500  R10A   =    -1.000
R42    =      0.570  R54    =       1.000  R54N   =     1.000
R711   =      1.000  R711N  =       1.000  SCP2   =     0.767
SFADP  =      0.0    SFADSP =       0.0    SFAUXP =     0.0
SFBPP  =      0.0    SFBPP  =       0.0    SFBTP  =     0.0
SFDIVP =      0.0    SFINSP =       0.0    SFIP   =     0.0
SFSFC1 =      0.980  SFSFC2 =       0.975  SFSFC3 =     1.000
S41    =      0.500  SCPG   =       1.000  TOSA   =   518.700
TR     =    520.000  TFAN   =       0.     TROAH  =     9275.
TWTO   =      0.500  T2     =       2657.  T5M    =     3400.
T51    =      0.     T7M    =       3400.  T71    =        0.
VC1    =      0.900  WCWAL  =       0.013  XMDES  =     0.850
XMT    =      0.850  YREN   =         72.  FRAT   =     0.0
FRPN   =      0.0    RDIAM  =       1.153  RLENG  =     1.537

IPR    =          1  IPRINT =           0  IPLOT  =         0
KERROR =          0  KCDE   =           2  KTS    =         0
KT7    =          0  MINPP  =           0  NAB    =         6
NOZZ   =          0  NPROP  =           6  NSUMM  =        10
                                           IENG   =         2 (TF30)

ALTD   =      0.         0.      5000.       5000.     15000.      15000.
XMACH  =      0.0        0.250   0.250       0.700     0.440       0.600
XMPKI  =      0.0        0.300   0.600       0.900     1.200       4.000
XPRI   =      1.000      1.000   1.000       1.000     1.000       1.000
   CALLING MODULE NUMBER  6
```

```
                    INITIAL WEIGHTS INPUT DATA
                    ACSYNT MODULE NUMBER 6
                    MODULE VERSION  4-76
AIRCRAFT TYPE:  BOMBER
TITLE:  ** S-3A "VIKING" - BASIC A/C WEIGHTS **

CONTROL OPTIONS

    IPRINT  =  1    IGRAPH  =  1    ITAIL   =  0    KERROR  =  0
    IFELT   =  0    ICHLTO  =  0    KWING   =  1    KBODY   =  1
    K1      = 1.00 K2       = 1.00 K3       =  1.0 K4       =  1.0
    K5      = 1.00 K6       = 1.00 AFMACH  = 0.79 MAXIT    =  1
    KB      = 0.00 KP1      = 0.00 KP2      = ****
    F2      = 1.00 TECHG    = 1.00 WCTJ     =     43000.
    STRESS  =    30000.             DENS     =  0.056      ELIFTF =0.0
    TECHI   =  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00
  SLOPE(1)  =  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00
  SLOPE(9)  =  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00

INITIAL ESTIMATES
 IF CODE = 1. WEIGHT IS FIXED

 QUANTITY         VALUE   CODE          QUANTITY        VALUE   CODE

  WAF      =     11180.    0            WAIRC    =       959.    1
  WAPU     =         0.    0            WARMAM   =      1060.    1
  WAFM     =       357.    1            WBAG     =         0.    0
  WAAL     =         0.    0            WBODY    =      5068.    1
  WCAND    =         0.    0            WCARGO   =         0.    0
  WCREW    =       853.    1            WE       =      2953.    1
  WELT     =      4323.    1            WEP      =      1086.    1
  WETANK   =         0.    0            WFEQ     =      5160.    0
  WFS      =       340.    1            WCC      =       300.    1
  WGEAR    =      1144.    1            WHDR     =       389.    1
  WHT      =       769.    1            WINST    =       174.    1
  WLG      =      1673.    1            WMISS    =         0.    0
  WNA      =       805.    1            WPA      =         0.    0
  WPASS    =         0.    0            WPL      =      3870.    0
  WPS      =      3296.    1            WSC      =      1604.    1
  WTSUM    =     43000.    0            WVT      =       585.    1
  WWING    =      4690.    1            WBOMB    =         0.    0
  WELVP    =         0.    0            WPIV     =         0.    0
  WLIFTF   =         0.    0


    * * * BEGIN VEHICLE CONVERGENCE -

    ESTIMATED GROSS WEIGHT  = 0.43000E 05
    CALLING MODULE NUMBER  1
    CALLING MODULE NUMBER  2
    CALLING MODULE NUMBER  6

    CALCULATED GROSS WEIGHT = 0.41944E 05

    SLOPE OF WCALC VS WEST LINE = 0.75000E 00

    ESTIMATED GROSS WEIGHT  = 0.38776E 05
    CALLING MODULE NUMBER  1
    CALLING MODULE NUMBER  2
    CALLING MODULE NUMBER  6

    CALCULATED GROSS WEIGHT = 0.41294E 05

    SLOPE OF WCALC VS WEST LINE = 0.32300E 00

    ESTIMATED GROSS WEIGHT  = 0.41752E 05
    CALLING MODULE NUMBER  1
    CALLING MODULE NUMBER  2
    CALLING MODULE NUMBER  6

    CALCULATED GROSS WEIGHT = 0.41743E 05
```

```
  CALLING MODULE NUMBER  1
  CALLING MODULE NUMBER  2
SUMMARY OF NON FATAL ERRORS ENCOUNTERED IN PROP
P5 IS LESS THAN P6S     1  TIMES

IN LOITER, PHASE 5, LEG 1, DRAG = 1.003 OF TN.
CD CHANGED TO MATCH THRUST.


IN LOITER, PHASE 5, LEG 2, DRAG = 1.003 OF TN.
CD CHANGED TO MATCH THRUST.


IN LOITER, PHASE 8, LEG 1, DRAG = 1.001 OF TN.
CD CHANGED TO MATCH THRUST.


    FOCT COULD NOT BE FOUND FROM FUN1      CL, CD, KP SHOULD BE CHECKED

    AERODYNAMICS MODULE

MACH OR ALTITUDE SEARCH CONCLUDED.
    X = 0.55197E 05  Y = 0.26506E 02
    CALLING MODULE NUMBER  6
    CALLING MODULE NUMBER  1
```

NASA/AMES A C S Y N T VERSION 4-76 FINAL OUTPUT

FUSELAGE DEFINITION X	R	AREA	NACELLE DEFINITION X-XNOSE	R	AREA	NACELLE LOCATION X	Y	Z
0.0	0.0	0.0	0.0	2.27	16.25	14.51	7.75	0.42
0.98	1.30	5.31	3.33	2.27	16.25	14.51	-7.75	0.42
1.97	2.08	13.56	10.00	2.27	16.25			
2.95	2.66	22.31	13.33	2.27	16.25			
3.94	3.11	30.48						
4.92	3.45	37.42						
5.91	3.69	42.69						
6.89	3.83	45.99						
7.88	3.87	47.17						
8.86	3.88	47.17						
9.85	3.88	47.17						
10.83	3.88	47.17						
11.82	3.88	47.17						
12.80	3.88	47.17						
13.79	3.88	47.17						
14.77	3.88	47.17						
15.76	3.88	47.17						
16.74	3.88	47.17						
17.73	3.88	47.17						
18.71	3.88	47.17						
19.70	3.88	47.17						
20.68	3.88	47.17						
21.67	3.88	47.17						
22.65	3.88	47.17						
23.64	3.88	47.17						
24.63	3.88	47.17						
25.61	3.88	47.17						
26.59	3.87	47.08						
27.58	3.86	46.74						
28.56	3.83	46.26						
29.55	3.80	45.33						
30.53	3.75	44.26						
31.52	3.70	42.97						
32.50	3.63	41.44						
33.49	3.55	39.70						
34.47	3.47	37.76						
35.46	3.37	35.62						
36.44	3.26	33.30						
37.43	3.13	30.81						
38.41	3.00	28.19						
39.40	2.85	25.44						
40.38	2.68	22.59						
41.37	2.50	19.67						
42.35	2.31	16.71						
43.34	2.09	13.75						
44.32	1.86	10.84						
45.31	1.60	8.03						
46.29	1.31	5.40						
47.28	0.98	3.04						
48.26	0.59	1.11						
49.25	0.0	0.0						

	FUSELAGE		**PODS**
MAX. DIAMETER........	7.750	.....	4.548
FINENESS RATIO.......	6.355		
SURFACE AREA........	597.015	.....	190.529 (EACH)
VOLUME.............	1713.973		

NASA/AMES A C S Y N T VERSION 4-76 FINAL OUTPUT

DIMENSIONS OF PLANAR SURFACES

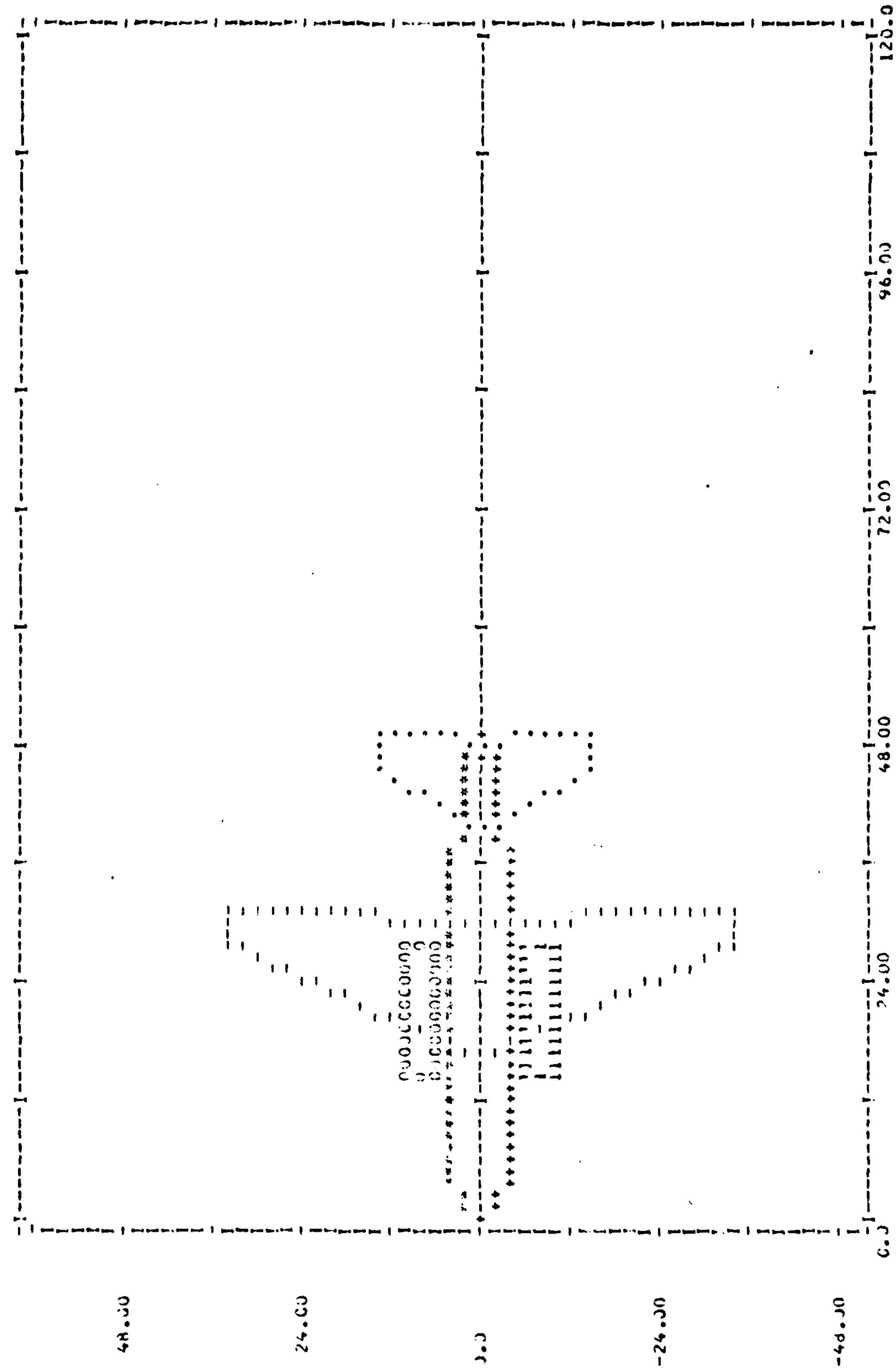
	WING	H.TAIL	V.TAIL	CANARD	UNITS
PLAN AREA..............	55[illegible].0	130.0	120.0	0.0	(SQ.FT.)
SURFACE AREA...........	1042.2	220.4	259.6	0.0	(SQ.FT.)
VOLUME.................	[illegible]15.2	[illegible].4	104.2	0.0	(CU.FT.)
SPAN...................	67.[illegible]	27.0[illegible]	13.606	0.0	(FT.)
L.E. SWEEP.............	19.[illegible]64	24.615	43.926	0.0	(DEG.)
C/4 SWEEP..............	15.000	23.000	38.500	0.0	(DEG.)
T.E. SWEEP.............	2.010	3.531	16.235	0.0	(DEG.)
ASPECT RATIO	7.730	4.050	1.435	0.0	
ROOT CHORD.............	14.073	[illegible].324	14.046	0.0	(FT.)
ROOT THICKNESS.........	27.526	11.159	20.227	0.0	(IN.)
ROOT T/C	0.163	0.100	0.120	0.0	
TIP CHORD..............	[illegible].518	4.039	4.916	0.0	(FT.)
TIP THICKNESS..........	5.366	4.811	5.899	0.0	(IN.)
TIP T/C	0.120	0.100	0.100	0.0	
TAPER RATIO	0.250	0.430	0.350	0.0	
MEAN AERO CHORD........	9.351	7.020	10.214	0.0	(FT.)
LE ROOT AT.............	1[illegible].983	39.089	35.204	0.0	(FT.)
C/4 ROOT AT............	19.501	41.420	33.715	0.0	(FT.)
TE ROOT AT.............	30.[illegible]56	48.413	49.250	********	(FT.)
LE M.A.C. AT...........	2[illegible].7[illegible]	41.755	40.704	********	(FT.)
C/4 M.A.C. AT..........	23.245	43.550	43.258	********	(FT.)
TE M.A.C. AT...........	30.633	48.815	50.918	********	(FT.)
Y M.A.C. AT............	1[illegible].5[illegible]	5.853	5.711	********	
LE TIP AT..............	27.[illegible]31	45.331	43.309	0.000	(FT.)
C/4 TIP AT.............	28.710	46.333	49.538	0.000	(FT.)
TE TIP AT..............	31.349	49.340	53.225	-0.439	(FT.)
ELEVATION..............	2.433	0.0	4.235	0.0	(FT.)
VOLUME COEFF.		0.620	0.063	0.0	

F U E L T A N K S

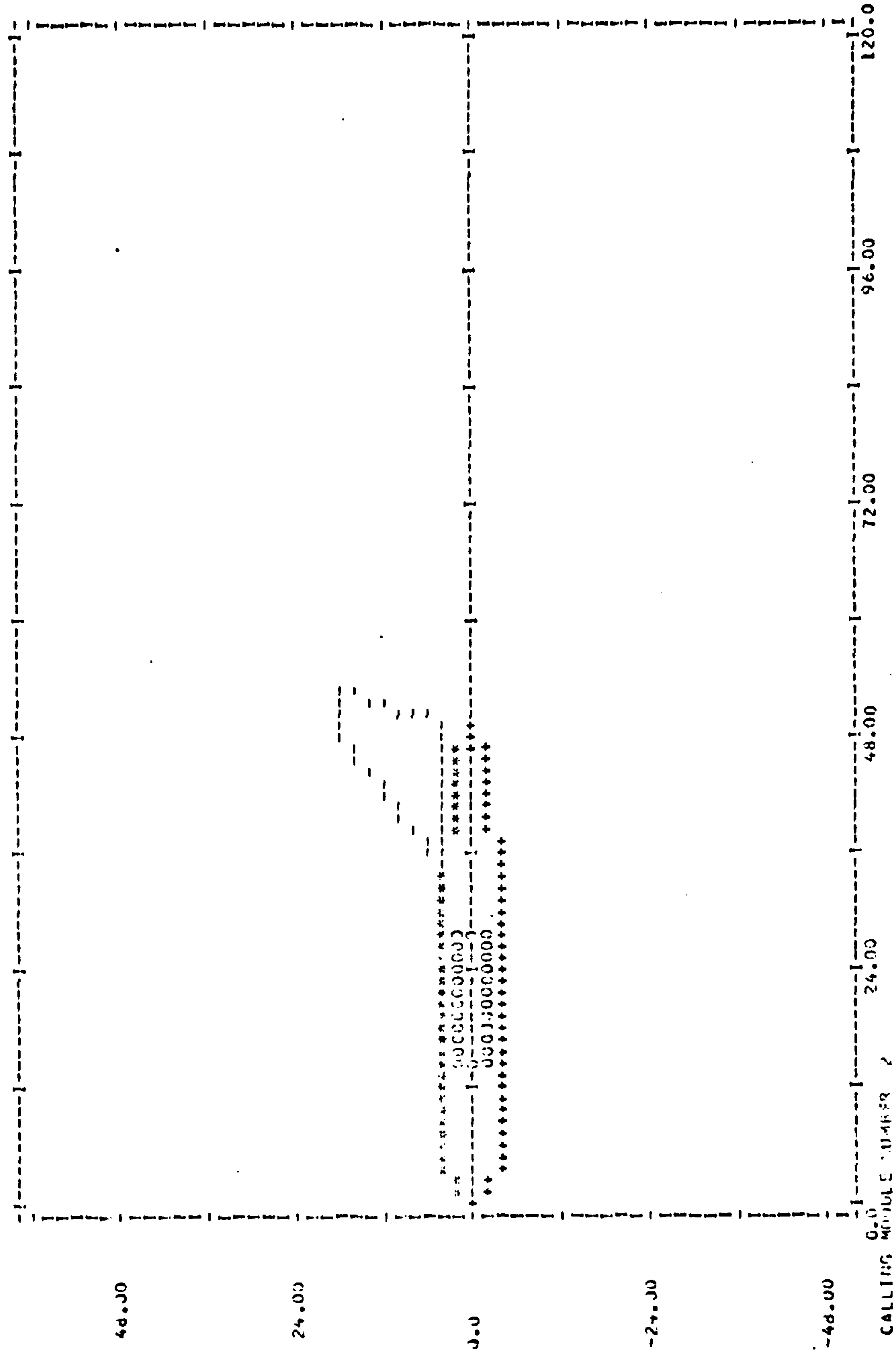
TANK	VOLUME	WEIGHT	DENSITY
WING	26[illegible].	13134.	50.00
FUS#1	0.	0.	50.00
FUS#2	0.	0.	50.00
TOTAL		13134.	

MISSION FUEL REQUIRED= 12380.

AVAILABLE FUEL VOLUME IN WING= 454.



120.0
96.00
72.00
48.00
24.00
0.0
48.00
24.00
-24.00
-48.00


48.00
24.00
0.0
-24.00
-48.00
CALLING MODULE NUMBER 2
0.0
24.00
48.00
72.00
96.00
120.0

TRAJECTORY OUTPUT

MISSION 1 (PAYLOAD = 2267. LB)

PHASE	M SFC(I) SFC(U)	H THRUST(I) THRUST(U)	CL CD CLINST	ALPHA L/D	WFUEL W	TIME WA PR	VEL Q X
ACCEL	0.33	0.	0.4308	3.67	23.2	0.21	368.
	0.48	14078.	0.0304	14.16	41559.1	309.43	161.
	0.48	14098.	0.0			1.00	1.
CLIMB	0.51	15000.	0.2923	2.33	313.7	3.20	542.
	0.53	6905.	0.0250	11.67	41245.3	219.19	220.
	0.53	6905.	0.0			1.00	14.
ACCEL	0.57	15000.	0.2366	1.80	54.0	0.74	624.
	0.56	6038.	0.0233	10.14	41191.3	211.69	291.
	0.56	8098.	0.0			1.00	4.
CRUISE	0.59	15000.	0.2186	1.66	4180.6	93.52	624.
	0.67	3547.	0.0229	9.54	35950.7	192.68	291.
	0.67	3547.	0.0			1.00	576.
LOITER	0.34	100.	0.3422	2.89	2112.6	60.00	385.
	0.75	2003.	0.0267	12.81	33838.1	251.21	176.
	0.75	2306.	0.0			1.00	228.
CLIMB	0.48	10000.	0.2257	1.82	174.4	1.65	520.
	0.54	10545.	0.0231	9.76	33663.7	254.81	237.
	0.54	10545.	0.0			1.00	7.
CRUISE	0.44	10000.	0.4172	3.54	4487.1	160.34	372.
	0.73	2169.	0.0303	13.75	29176.6	183.77	121.
	0.73	2175.	0.0			1.00	588.
LOITER	0.34	100.	0.2777	2.33	161.4	5.00	385.
	0.75	2537.	0.0246	11.23	29015.2	248.13	176.
	0.75	2585.	0.0			1.00	19.

FUEL SUMMARY

```
TAKEOFF FUEL
        AFTO1 =       47.        TOTAL FUEL =  12380.
        AFTO2 =      112.          EXTERNAL =      0.
 MISSION FUEL =    11606.          INTERNAL =  12380.
 RESERVE FUEL =      614.
 TRAPPED FUEL =      100.

                            BLOCK TIME   =    6.411   HOURS
                            BLOCK RANGE  = 1437.0     N. M.
TAKEOFF FIELD LENGTH(TOTAL RUN)          =   2276.    FEET
LANDING FIELD LENGTH(TOTAL RUN)          =   2914.    FEET
LANDING FIELD LENGTH(GROUND RUN)         =   1476.    FEET
  WEIGHT FOR LANDING CALCULATION         =  35092.    POUNDS
                         TAKEOFF WEIGHT  =  41741.    POUNDS
                         LANDING WEIGHT  =  29015.    POUNDS
                     ENDURANCE MACH NO.  =    0.3
                     ENDURANCE ALTITUDE  = 32197.     FEET
                     ENDURANCE TIME      =    3.839   HOURS
                          LOITER RADIUS  =    0.0     N. M.
                          LOITER RADIUS  =    0.0     N. M.
    CALLING MODULE NUMBER  3
```

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.330 ALTITUDE = 1000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

LITE =	0.0066	WAVE =	0.0
STORE =	0.0	FRIC =	0.0142
TANK =	0.0	PLANT =	0.0
BOMBS =	0.0	ENGINE =	0.0

FRICTION = 0.0023 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0208 (NO CAMBER)
= 0.0208 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0208	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0603	0.0209	0.0002	0.0	0.0000	0.0014	2.8786	0.0000	0.04	-0.0045
1.00	0.1200	0.0215	0.0007	0.0	0.0000	0.0028	5.5638	0.0001	0.08	-0.0170
1.50	0.1795	0.0224	0.0017	0.0	0.0001	0.0041	7.9562	0.0001	0.12	-0.0256
2.00	0.2380	0.0237	0.0029	0.0	0.0002	0.0055	10.0433	0.0002	0.16	-0.0343
4.00	0.4651	0.0323	0.0115	0.0	0.0008	0.0111	14.5063	0.0008	0.33	-0.0698
6.00	0.6909	0.0461	0.0254	0.0	0.0017	0.0166	14.9812	0.0018	0.51	-0.1070
8.00	0.9069	0.0650	0.0442	0.0	0.0031	0.0221	13.9561	0.0032	0.71	-0.1459
10.00	1.1156	0.0886	0.0679	0.0	0.0049	0.0277	12.5971	0.0049	0.92	-0.1868
12.00	1.3206	0.1169	0.0962	0.0	0.0070	0.0332	11.2943	0.0070	1.15	-0.2299

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.340 ALTITUDE = 5000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0066	WAVE =	0.0
STORE =	0.0	FRIC =	0.0144
TANK =	0.0	BLUNT =	0.0
BOMBS =	0.0	ENGINE =	0.0

FRICTION = 0.0023 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0209 (NO CAMBER)
= 0.0209 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0209	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0604	0.0211	0.0002	0.0	0.0000	0.0014	2.8604	0.0000	0.04	-0.0085
1.00	0.1203	0.0217	0.0007	0.0	0.0000	0.0028	5.5490	0.0001	0.08	-0.0171
1.50	0.1797	0.0226	0.0017	0.0	0.0001	0.0042	7.9480	0.0001	0.12	-0.0257
2.00	0.2386	0.0239	0.0030	0.0	0.0002	0.0055	9.9845	0.0002	0.16	-0.0345
4.00	0.4692	0.0325	0.0116	0.0	0.0008	0.0111	14.4334	0.0008	0.33	-0.0702
6.00	0.6925	0.0464	0.0255	0.0	0.0017	0.0166	14.9228	0.0018	0.51	-0.1075
8.00	0.9090	0.0653	0.0444	0.0	0.0031	0.0221	13.9110	0.0032	0.71	-0.1466
10.00	1.1192	0.0871	0.0662	0.0	0.0049	0.0276	12.5623	0.0050	0.92	-0.1878
12.00	1.3234	0.1175	0.0965	0.0	0.0070	0.0330	11.2671	0.0070	1.15	-0.2311

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.350 ALTITUDE = 10000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0066	WAVE =	0.0
STORE =	0.0	FRIC =	0.0145
TANK =	0.0	BLUNT =	0.0
BOMBS =	0.0	ENGINE =	0.0

FRICTION = 0.0023 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0212 (NO CAMBER)
= 0.0212 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	COSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0212	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0606	0.0214	0.0002	0.0	0.0000	0.0014	2.8324	0.0000	0.04	-0.0036
1.00	0.1207	0.0220	0.0008	0.0	0.0000	0.0028	5.4959	0.0001	0.08	-0.0172
1.50	0.1802	0.0229	0.0017	0.0	0.0001	0.0042	7.4745	0.0001	0.12	-0.0259
2.00	0.2392	0.0242	0.0030	0.0	0.0002	0.0055	9.3957	0.0002	0.16	-0.0347
4.00	0.4704	0.0328	0.0116	0.0	0.0009	0.0110	14.3362	0.0008	0.33	-0.0706
6.00	0.6942	0.0468	0.0256	0.0	0.0017	0.0166	14.8406	0.0019	0.51	-0.1081
8.00	0.9112	0.0658	0.0446	0.0	0.0031	0.0221	13.8497	0.0033	0.71	-0.1475
10.00	1.1217	0.0896	0.0684	0.0	0.0049	0.0275	12.5167	0.0050	0.93	-0.1888
12.00	1.3264	0.1131	0.0969	0.0	0.0070	0.0329	11.2326	0.0071	1.16	-0.2324

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.440 ALTITUDE = 15000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0067	WAVE =	0.0
STORE =	0.0	FRIC =	0.0113
TANK =	0.0	BLUNT =	0.0
BOMBS =	0.0	ENGINE =	0.0

FRICTION = 0.0023 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0211 (NO CAMBER)
= 0.0211 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0211	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0633	0.0213	0.0002	0.0	0.0000	0.0014	2.9266	0.0000	0.04	-0.0091
1.00	0.1240	0.0219	0.0008	0.0	0.0000	0.0028	5.6703	0.0001	0.08	-0.0182
1.50	0.1851	0.0228	0.0018	0.0	0.0001	0.0042	8.1060	0.0001	0.12	-0.0275
2.00	0.2457	0.0242	0.0031	0.0	0.0002	0.0055	10.1583	0.0002	0.16	-0.0368
4.00	0.4831	0.0332	0.0122	0.0	0.0008	0.0110	14.5313	0.0009	0.34	-0.0747
6.00	0.7127	0.0479	0.0268	0.0	0.0017	0.0166	14.8919	0.0021	0.53	-0.1141
8.00	0.9350	0.0677	0.0466	0.0	0.0031	0.0221	13.8069	0.0036	0.73	-0.1553
10.00	1.1504	0.0925	0.0715	0.0	0.0049	0.0276	12.4321	0.0056	0.95	-0.1985
12.00	1.3595	0.1221	0.1011	0.0	0.0070	0.0330	11.1316	0.0079	1.19	-0.2439

DETAILED AERODYNAMICS, DESIGN MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.550 ALTITUDE = 15000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0069	WAVE =	0.0
STORE =	0.0	FRIC =	0.0135
TANK =	0.0	BLUNT =	0.0
BOMBS =	0.0	ENGINE =	0.0

FRICTION = 0.0022 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0205 (NO CAMBER)
= 0.0205 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0205	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0665	0.0207	0.0002	0.0	0.0000	0.0014	3.2132	0.0000	0.04	-0.0103
1.00	0.1323	0.0214	0.0009	0.0	0.0000	0.0028	6.1923	0.0001	0.09	-0.0207
1.50	0.1976	0.0225	0.0020	0.0	0.0001	0.0041	8.7942	0.0002	0.13	-0.0312
2.00	0.2622	0.0240	0.0035	0.0	0.0002	0.0055	10.9330	0.0003	0.17	-0.0417
4.00	0.5151	0.0341	0.0137	0.0	0.0008	0.0111	15.0917	0.0012	0.36	-0.0843
6.00	0.7592	0.0504	0.0300	0.0	0.0017	0.0166	15.0514	0.0027	0.56	-0.1233
8.00	0.9949	0.0725	0.0520	0.0	0.0031	0.0222	13.7238	0.0046	0.77	-0.1739
10.00	1.2227	0.0999	0.0795	0.0	0.0049	0.0278	12.2336	0.0071	1.00	-0.2213
12.00	1.4341	0.2631	0.2626	0.0	0.0071	0.0334	5.0564	0.0067	2.43	-0.2703

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.340 ALTITUDE = 100.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

```
  LITE =   0.0066          WAVE =   0.0
 STRUT =   0.0             FRIC =   0.0141
  TANK =   0.0            BLUNT =   0.0
 BOMBS =   0.0           ENGINE =   0.0

                       FRICTION =   0.0023  (INCL. IN FRIC)
                       BOATTAIL =   0.0
                           COWL =   0.0

CD(MIN) =  0.0206  (NO CAMBER)
        =  0.0206  (WITH CAMBER)
```

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0206	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0604	0.0208	0.0002	0.0	0.0000	0.0014	2.9009	0.0000	0.04	-0.0085
1.00	0.1203	0.0214	0.0007	0.0	0.0000	0.0028	5.6254	0.0001	0.08	-0.0171
1.50	0.1797	0.0223	0.0017	0.0	0.0001	0.0041	8.0527	0.0001	0.12	-0.0257
2.00	0.2386	0.0236	0.0030	0.0	0.0002	0.0055	10.1095	0.0002	0.16	-0.0345
4.00	0.4692	0.0322	0.0116	0.0	0.0008	0.0111	14.5709	0.0008	0.33	-0.0702
6.00	0.6925	0.0461	0.0255	0.0	0.0017	0.0166	15.0183	0.0018	0.51	-0.1075
8.00	0.9091	0.0651	0.0444	0.0	0.0031	0.0222	13.9738	0.0032	0.71	-0.1466
10.00	1.1195	0.0888	0.0682	0.0	0.0049	0.0277	12.6033	0.0049	0.92	-0.1876
12.00	1.3236	0.1172	0.0966	0.0	0.0071	0.0332	11.2943	0.0070	1.15	-0.2303

CALLING MODULE NUMBER 4

ENGINE SUMMARY

```
ENGINE DIAMETER =      4.17 FEET
ENGINE LENGTH   =      8.67 FEET
ENGINE WEIGHT   =   1425.00 POUNDS
BYPASS RATIO    =      6.25
NO OF ENGINES   =       2.
DRAG REF AREA   =    509.00 SQ FEET
PXCC   = PERCENT OF ENGINE CORRECTED AIRFLOW
THRUST = ENGINE THRUST (POUNDS PER ENGINE)
SFC    = ENGINE SPECIFIC FUEL CONSUMPTION
THRUSTU= THRUST PER ENGINE IN LBS, W/O INSTAL DRAG CORR
SFCU   = SFC,1/HR, W/O INSTALLATION DRAG CORR
CDINS  = TOT INSTALLATION DRAG COEF PER A/C (SWING REF)
```

MACH	ALT	PWCC	THRUST	THRUSTU	SFC	SFCU	COINS
0.0	0.	100.	9275.	9275.	0.349	0.349	0.0
		98.	8712.	8712.	0.356	0.356	0.0
		96.	7676.	7676.	0.351	0.351	0.0
		94.	6966.	6966.	0.352	0.352	0.0
		92.	5597.	5597.	0.367	0.367	0.0
		90.	4359.	4359.	0.435	0.435	0.0
		88.	2394.	2394.	0.691	0.691	0.0
		86.	2160.	2160.	0.663	0.663	0.0
		84.	1931.	1931.	0.637	0.637	0.0
		82.	1708.	1708.	0.612	0.612	0.0
0.250	0.	100.	7414.	7414.	0.450	0.450	0.0
		98.	6790.	6790.	0.459	0.459	0.0
		96.	6138.	6138.	0.471	0.471	0.0
		94.	5338.	5338.	0.484	0.484	0.0
		92.	4370.	4370.	0.522	0.522	0.0
		90.	3120.	3120.	0.641	0.641	0.0
		88.	2229.	2229.	0.783	0.783	0.0
		86.	1991.	1991.	0.758	0.758	0.0
		84.	1761.	1761.	0.736	0.736	0.0
		82.	1536.	1536.	0.718	0.718	0.0
0.250	5000.	100.	6355.	6355.	0.455	0.455	0.0
		98.	5793.	5793.	0.458	0.458	0.0
		96.	5135.	5135.	0.460	0.460	0.0
		94.	4429.	4429.	0.473	0.473	0.0
		92.	3625.	3625.	0.509	0.509	0.0
		90.	2537.	2537.	0.626	0.626	0.0
		88.	1846.	1846.	0.765	0.765	0.0
		86.	1549.	1549.	0.741	0.741	0.0
		84.	1458.	1458.	0.720	0.720	0.0
		82.	1272.	1272.	0.702	0.702	0.0
0.700	5000.	100.	5518.	5518.	0.595	0.595	0.0
		98.	5244.	5244.	0.621	0.621	0.0
		96.	4829.	4829.	0.653	0.653	0.0
		94.	4136.	4136.	0.676	0.676	0.0
		92.	3414.	3414.	0.722	0.722	0.0
		90.	2637.	2637.	0.821	0.821	0.0
		88.	2136.	2136.	0.833	0.833	0.0
		86.	1590.	1590.	[illegible]	[illegible]	0.0
		84.	1653.	1653.	0.849	0.849	0.0
		82.	1424.	1424.	0.838	0.838	0.0
0.440	15000.	100.	4046.	4046.	0.524	0.524	0.0
		98.	3619.	3619.	0.524	0.524	0.0
		96.	3192.	3192.	0.531	0.531	0.0
		94.	2724.	2724.	0.550	0.550	0.0
		92.	2225.	2225.	0.593	0.593	0.0
		90.	1642.	1642.	0.705	0.705	0.0
		88.	1266.	1266.	0.798	0.798	0.0
		86.	1124.	1124.	0.778	0.778	0.0
		84.	987.	987.	0.760	0.760	0.0
		82.	854.	854.	0.748	0.748	0.0
0.600	15000.	100.	4056.	4056.	0.564	0.564	0.0
		98.	3614.	3614.	0.566	0.566	0.0
		96.	3168.	3168.	0.576	0.576	0.0
		94.	2710.	2710.	0.597	0.597	0.0
		92.	2226.	2226.	0.641	0.641	0.0
		90.	1692.	1692.	0.739	0.739	0.0
		88.	1349.	1349.	0.808	0.808	0.0
		86.	1195.	1195.	0.790	0.790	0.0
		[illegible]	[illegible]	[illegible]	0.776	0.775	0.0

WEIGHT STATEMENT - BOMBERS
** S-3A "VIKING" - BASIC A/C WEIGHTS **
ACSYNT MODULE NUMBER 6
MODULE VERSION 4-76

COMPONENT	POUNDS	KILOGRAMS	PERCENT
AIRFRAME STRUCTURE	13787.	6254.	33.03
WING	4890.	2218.	11.72
FUSELAGE	5068.	2299.	12.14
HORIZONTAL TAIL	769.	349.	1.84
VERTICAL TAIL	585.	265.	1.40
NACELLES	805.	365.	1.93
ALIGHTING GEAR	1670.	757.	4.00
PROPULSION	3298.	1496.	7.90
ENGINES (2)	2951.	1339.	7.07
FUEL SYSTEM	346.	157.	0.83
FIXED EQUIPMENT	10009.	4540.	23.98
HYD. + PNEU.	389.	176.	0.93
ELECTRICAL	1086.	493.	2.60
AVIONICS	4353.	1974.	10.43
INSTRUMENTATION	174.	79.	0.42
DE-ICE/AIR CONDITION	959.	435.	2.30
AUXILIARY GEAR	1144.	519.	2.74
FURNISH. + EQPT.	300.	136.	0.72
FLIGHT CONTROLS	1604.	728.	3.84
FUEL	12380.	5616.	29.66
PAYLOAD	2267.	1028.	5.43
FLIGHT CREW (4)	850.	386.	2.04
ARMAMENT	357.	162.	0.86
AMMUNITION	1060.	481.	2.54
MISSILES	0.	0.	0.0
BOMBS	0.	0.	0.0
EXTERNAL TANKS	0.	0.	0.0
ADV. WEAPONS 1	0.	0.	0.0
ADV. WEAPONS 2	0.	0.	0.0
CALCULATED WEIGHT	41741.	18934.	100.00
ESTIMATED WEIGHT	41741.	18934.	
PERCENT ERROR	-0.00		

CALLING MODULE NUMBER 11

SUMMARY --- ACSYNT OUTPUT --- NASA, AMES RESEARCH CENTER

ENGLISH UNITS -
DISTANCES IN FEET
WEIGHTS IN LBS.
FORCES IN LBS.
PRESSURES IN LBS/FT**2

** LOCKHEED S-3A "VIKING" - AIRCRAFT CORRELATION ** <F202

GENERAL		FUSELAGE			WING	HTAIL	VTAIL
WG	41741.	LENGTH	49.3	AREA	598.0	180.0	129.0
W/S	69.8	DIAMETER	7.8	WETTED AREA	1042.2	228.4	259.6
T/W	0.44	VOLUME	1714.0	SPAN	68.0	27.0	13.6
N(Z) ULT	5.3	WETTED AREA	995.6	L.E. SWEEP	19.1	24.6	46.5
CREW	4.	FINENESS RATIO	6.4	C/4 SWEEP	15.0	20.0	38.5
PASSENGERS	0.			ASPECT RATIO	7.7	4.0	1.4
				TAPER RATIO	0.3	0.4	0.3
ENGINE		WEIGHTS		T/C ROOT	0.16	0.10	0.12
				T/C TIP	0.12	0.10	0.10
NUMBER	2.		% WG	ROOT CHORD	14.1	9.3	14.0
LENGTH	8.7	STRUCT. 13757.	33.0	TIP CHORD	3.5	4.0	4.9
DIAM.	4.0	PROPUL. 3278.	7.9	M.A. CHORD	9.9	7.0	10.2
WEIGHT	1425.0	FIX. EQ. 10005.	24.0	LOC. OF L.E.	16.1	39.1	35.2
TSLS	9275.	FUEL 12380.	29.7				
SFCSLS	0.35	PAYLOAD 2267.	5.4				

MISSION SUMMARY

PHASE	MACH	ALT	FUEL	TIME	DIST	L/D	THRUST	SFC	Q
TAKEOFF		0.	159.	6.0	2278.0				
ACCEL	0.33	0.	25.	0.2	0.7	14.2	14098.0	0.5	161.3
CLIMB	0.51	15000.	314.	3.2	14.4	11.7	8904.7	0.5	220.0
ACCEL	0.59	15000.	54.	0.7	4.2	10.1	8097.6	0.6	291.1
CRUISE	0.59	15000.	4161.	93.5	575.7	9.5	3987.2	0.7	291.1
LOITER	0.34	100.	2113.	60.0	228.0	12.8	2807.5	0.8	175.7
CLIMB	0.46	10000.	174.	1.6	6.8	9.8	10544.7	0.5	237.3
CRUISE	0.34	30000.	4457.	160.3	598.2	13.7	2199.1	0.7	121.3
LOITER	0.34	100.	161.	5.0	19.0	11.3	2587.0	0.7	175.7
LANDING					2914.5				

BLOCK TIME = 5.411 HR
BLOCK RANGE = 1437.0 NM

IHC9001 EXECUTION TERMINATING DUE TO ERROR COUNT FOR ERROR NUMBER 217

IHC217I FIOCS - END OF DATA SET ON UNIT 5

TRACEBACK ROUTINE	CALLED FROM ISN	REG. 14	REG. 15	REG. 0	REG. 1
IBCOM		000A8E18	00114F78	00000000	00000000
CERE01	0012	420A748F	000A7A00	00000028	000A7244
MAIN		0000EF22	010A7008	FD000008	00138FF8

ENTRY POINT= 010A7008

APPENDIX B

STOL VIKING Computer Printout

CODES

CONTROL PROGRAM

FOR

ENGINE/WING SYNTHESIS

TITLE

*** LOCKHEED S3A "STOL VIKING" ANALYSIS ***

CARD IMAGES OF CONTROL DATA

```
CARD                    IMAGE

  1)   $          CERES CONTROL CARDS
  2)   ***  LOCKHEED S3A "STOL VIKING" ANALYSIS   ***
  3)   $          BLOCK = B
  4)              1          0          0          0          0          0          0
  5)   $          BLOCKS C-I (DESIGN VARIABL. OPTIMIZATION) OMITTED
  6)   $          BLOCKS J,K (SENSITIVITY EVAL.) OMITTED
  7)   $          BLOCKS L-P OMITTED, N2VAR = 0
  8)   END
```

```
TITLE:
*** LOCKHEED S3A USTOL VIKING ANALYSIS ***

CONTROL PARAMETERS:
CALCULATION CONTROL,                       NCALC  =    1
NUMBER OF GLOBAL DESIGN VARIABLES,           NDV  =    0
NUMBER OF SENSITIVITY VARIABLES,             NSV  =    0
NUMBER OF FUNCTIONS IN TWO-SPACE,          N2VAR  =    0
INPUT INFORMATION PRINT CODE,             IPNPUT  =    0
SENSITIVITY PRINT CODE,                   IPSENS  =    0
TWO-SPACE PRINT CODE,                     IP2VAR  =    0
DEBUG PRINT CODE,                          IPDBG  =    0

CALCULATION CONTROL, NCALC
VALUE    MEANING
  1      SINGLE ANALYSIS
  2      OPTIMIZATION
  3      SENSITIVITY
  4      TWO-VARIABLE FUNCTION SPACE

DATA STORAGE REQUIREMENTS

               REAL                              INTEGER
  INPUT   EXECUTION   AVAILABLE      INPUT   EXECUTION   AVAILABLE
    9         9         5000           1         1         1000
```

```
AAAAAAA   CCCCCCC   SSSSSSS   Y     Y   N     N   TTTTTTT
A     A   C         S          Y   Y    NN    N      T
A     A   C         S           Y Y     N N   N      T
AAAAAAA   C         SSSSSSS      Y      N  N  N      T
A     A   C               S      Y      N   N N      T
A     A   C               S      Y      N    NN      T
A     A   CCCCCCC   SSSSSSS      Y      N     N      T
```

N A S A - A M E S   P R O G R A M

F O R

A I R C R A F T   S Y N T H E S I S

NAVAL POSTGRADUATE SCHOOL VERSION 9 - 77
MONTEREY CALIFORNIA

T I T L E

*** LOCKHEED S3A "SEPL VIKING" ANALYSIS ***

AIRCRAFT TYPE - BOMBER(2SW)

```
TITLE:
***  LOCKHEED S3A "STOL VIKING" ANALYSIS  ***


AIRCRAFT TYPE - (CONVENTIONAL)


CONTROL PARAMETERS:
CALCULATION CONTROL,                        KCALC  =    2
READ CONTROL,                               MREAD  =    5
EXECUTION CONTROL,                          MAXEC  =    3
WRITE CONTROL,                              MWRITE =    5
NUMBER IDENTIFYING CONVERGENCE
  VARIABLE FOR CONVERGED VEHICLE,           IOBJ   =  570
NUMBER IDENTIFYING COMPARISON
  VARIABLE FOR CONVERGED VEHICLE,           JOBJ   =  595
SUMMARY OUTPUT PRINT CODE,                  ISUM   =    0
GLOBAL ERROR PRINT CODE,                    KLUMP  =    0
GLOBAL COMMON INITIALIZATION CODE,          INIT   =    0
DEBUG PRINT CODE,                           IDBUG  =    2
GLOBAL PLOT CONTROL,                        ICPLT  =   -1
DATA TRANSFER INFORMATION FILE, ISDATA          =    7
DATA TRANSFER INFORMATION PRINT, IPDTA          =    0


VEHICLE CONVERGENCE INFORMATION:
CONVERGENCE TOLERANCE, TOL  = 0.10000E-03
ESTIM WCALC VS WEXT SLOPE   = 0.75000E 00
BOUNDING WEIGHT, WGMAX      = 0.60000E 05


MODULE IDENTIFICATION NUMBERS:

 NUMBER       MODULE
    1         GEOMETRY
    2         TRAJECTORY
    3         AERODYNAMICS
    4         PROPULSION
    5         STRUCTURES
    6         WEIGHTS
    7         CARGO
    8         GLOBAL I/O
    9         ECONOMICS
   10         NAVY

MODULES ARE CALLED FOR INPUT IN THE FOLLOWING ORDER:
    1    2    3    4    6

MODULES ARE CALLED FOR EXECUTION IN THE FOLLOWING ORDER:
    1    2    6

MODULES ARE CALLED FOR OUTPUT IN THE FOLLOWING ORDER:
    1    2    6    7    9
CALLING MODULE NUMBER  1
```

```
GEOMETRY DATA CARDS                    [illegible] VERSION  4-76
 1 [illegible] VIKING [illegible]
 2 &WING  [illegible]=7.75, [illegible] CHLA[illegible]=0.10, SWEEP=15.0, TAPER=0.25,
 3 TC[illegible]=0.1[illegible], TCTIP=[illegible], THL[illegible]=0.294, XWING=0.398, Z[illegible]T=0.625,
 4 [illegible]=0.57[illegible], [illegible]=1,    &END
 5 &HTAIL  [illegible]  [illegible]0, SWEEP=20.0, TAPER=0.43, XHTAIL=1.08[illegible],
 6 Z[illegible]T=1.0, [illegible]=1, [illegible]IT=F,    &END
 7 &VTAIL   [illegible]=1.45[illegible], [illegible]=12[illegible], SWEEP=33.5, TAPER=0.35, TCROOT=0.12,
 8 Z[illegible]T=1.05[illegible], [illegible]=1, [illegible]IT=F,    &END
 9 &WPOD    X=-[illegible], Y=0.22[illegible], Z=-0.[illegible]5,    &END
10 &WPOD    X=-0.[illegible], Y=0.3[illegible], Z=-0.9[illegible]5,    &END
11 &FUS     [illegible]MAX=7.75, [illegible]=4[illegible].25, [illegible]=[illegible].333, [illegible]N=1.022,  &END
12 &FLIC    LENGTH=0.[illegible],  &END
13 &BLIC    LENGTH=[illegible].7,  &END
14 &CR[illegible]  LENGTH=13.42, WIDTH=7.167, N[illegible]=4,   &END
15 &FUEL    [illegible]AC=0.0, WFUEL=13144.0,   &END

END OF GEOMETRY DATA CARDS
  15 CARDS [illegible]
```

NASA/AMES ACSYNT VERSION 4-76 INITIAL OUTPUT

FUSELAGE DEFINITION

X	R	AREA
0.0	0.0	0.0
0.98	1.30	5.31
1.97	2.08	13.59
2.95	2.66	[illegible]
3.94	3.11	[illegible]
4.92	3.45	37.42
5.91	3.69	42.65
6.89	3.83	45.99
7.88	3.87	47.17
8.86	3.88	47.17
9.85	3.88	47.17
10.83	3.88	47.17
11.82	3.88	47.17
12.80	3.88	47.17
13.79	3.88	47.17
14.77	3.88	47.17
15.76	3.88	47.17
16.74	3.88	47.17
17.73	3.88	47.17
18.71	3.88	47.17
19.70	3.88	47.17
20.68	3.88	47.17
21.67	3.88	47.17
22.65	3.88	47.17
23.64	3.88	47.17
24.63	3.88	47.17
25.61	3.88	47.17
26.59	3.87	47.08
27.58	3.86	46.74
28.56	3.83	46.16
29.55	3.80	[illegible]
30.53	3.75	44.20
31.52	3.70	42.97
32.50	3.63	41.44
33.49	3.55	39.70
34.47	3.47	37.76
35.46	3.37	35.62
36.44	3.26	33.30
37.43	3.13	30.81
38.41	3.00	28.19
39.40	2.85	25.44
40.38	2.68	22.59
41.37	2.50	19.67
42.35	2.31	16.71
43.34	2.09	13.75
44.32	1.86	10.84
45.31	1.60	8.03
46.29	1.31	5.40
47.28	0.98	3.04
48.26	0.59	1.11
49.25	0.0	0.0

FUSELAGE

MAX. DIAMETER........ 7.750
FINENESS RATIO....... 6.355
SURFACE AREA......... 999.015
VOLUME............... 1713.976

NASA/AMES [illegible] VERSION 6-76 INITIAL OUTPUT

DIMENSIONS OF PLANAR SURFACES

	WING	H.TAIL	V.TAIL	CANARD	UNITS
PLAN AREA..............	556.0	[illegible]	[illegible]	0.0	(SQ.FT.)
SURFACE AREA...........	[illegible]	[illegible]	[illegible]	0.0	(SQ.FT.)
VOLUME.................	[illegible]	86.4	[illegible]	0.0	(CU.FT.)
SPAN...................	67.[illegible]	27.000	[illegible]	0.0	(FT.)
L.E. SWEEP.............	15.064	24.615	43.926	0.0	(DEG.)
C/4 SWEEP..............	12.000	20.000	[illegible]	0.0	(DEG.)
T.E. SWEEP.............	[illegible]	[illegible]	15.235	0.0	(DEG.)
ASPECT RATIO	7.750	[illegible]	[illegible]	0.0	
ROOT CHORD.............	14.[illegible]	5.[illegible]	[illegible]	0.0	(FT.)
ROOT THICKNESS.........	27.[illegible]	11.1[illegible]	[illegible]	0.0	(IN.)
ROOT T/C	0.165	0.100	0.120	0.0	
TIP CHORD..............	[illegible]	4.0[illegible]	[illegible]	0.0	(FT.)
TIP THICKNESS..........	[illegible]	4.811	5.[illegible]	0.0	(IN.)
TIP T/C	0.120	0.100	0.100	0.0	
TAPER RATIO	0.750	0.430	[illegible]	0.0	
MEAN AERO CHORD........	[illegible]	7.020	[illegible]	0.0	(FT.)
LE ROOT AT.............	[illegible]	44.[illegible]	[illegible]	0.0	(FT.)
C/4 ROOT AT............	[illegible]	[illegible]	33.715	0.0	(FT.)
TE ROOT AT.............	[illegible]	[illegible]	49.250	********	(FT.)
LE M.A.C. AT...........	[illegible]	47.[illegible]	[illegible]	********	(FT.)
C/4 M.A.C. AT..........	[illegible]	[illegible]	43.29[illegible]	********	(FT.)
TE M.A.C. AT...........	[illegible]	54.0[illegible]	[illegible]	********	(FT.)
Y M.A.C. AT............	[illegible]	[illegible]	5.711	********	
LE TIP AT..............	27.[illegible]	50.[illegible]	[illegible]	0.000	(FT.)
C/4 TIP AT.............	23.710	51.[illegible]	[illegible]	0.000	(FT.)
TE TIP AT..............	31.349	54.[illegible]	53.225	-0.439	(FT.)
ELEVATION..............	2.433	17.841	[illegible]	0.0	(FT.)
VOLUME COEFF.		0.780	0.06[illegible]	0.0	

F U E L   T A N K S

TANK	VOLUME	WEIGHT	DENSITY
WING	263.	13134.	50.00
FUS#1	0.	0.	50.00
FUS#2	0.	0.	50.00
TOTAL		13134.	

MISSION FUEL REQUIRED= 13144.

AVAILABLE FUEL VOLUME IN WING = 454.
CALLING MODULE NUMBER 2

TRAJECTORY INPUT

TIMTO1 = 5.0	[illegible] = 0.	[illegible] = 2	IPLOT = 0
TIMTO2 = 1.0	VMAX = 700.	IPSTAT = -3	HMINP = 0.
FRFURL = 0.05	XDESC = 40.0	IPSTO1 = 5	HMAXP = 40000.
DESLF = 3.50	WKFUEL = 1.000	IPSTO2 = 2	DELHP = 4000.
ULTLF = 5.25	CEMACH = 0.550	[illegible] = 0	SMMINP = 0.300
RANGE = 1500.	[illegible] = 0.570	[illegible] = 0	SMMAXP = 0.900
WFUFL = 13144.	FLFAC = 1.000	IPRINT = 0	DELMP = 0.100
WFFXT = 0.	[illegible] = 0.250	[illegible] = 0	ACOMPP = 0.70
WFTRAP = 100.	NLFCOL = 0	NLFOCF = 0	NLFGLD = 0
FWGMAX = 1.200	TOL = 0.001	NTLCOM = 0	NMISS = 1

MISSION 1

PHASE	MACH START	MACH END	ALT START	ALT END	HORIZ DIST	TIME	NO. TURNS	VIND	IP IPRT	IX IR	IW IA
ACCEL	0.23	0.33	0.	500.	0.0	0.0	0.0	0.0	3 0	-1 0	0 0
CLIMB	0.33	0.44	500.	15000.	0.0	0.0	0.0	0.0	3 0	-1 0	0 0
ACCEL	0.44	0.59	-1.	15000.	0.0	0.0	0.0	0.0	3 0	-1 0	0 0
CRUISE	0.59	0.59	15000.	-1.	503.0	0.0	0.0	0.0	4 0	-1 0	0 1
LOITER	0.34	0.34	100.	100.	0.0	60.0	0.0	0.0	4 0	0 0	0 0
CLIMB	0.35	0.40	-1.	10000.	0.0	0.0	0.0	0.0	3 0	-1 0	0 0
CRUISE	0.34	0.34	10000.	-1.	503.0	0.0	0.0	0.0	4 0	-1 0	0 0
LOITER	0.34	0.0	100.	100.	0.0	5.0	0.0	0.0	4 0	0 0	0 0

CALLING MODULE NUMBER 3

AERODYNAMIC INPUT DATA

*** LOCKHEED S[illegible] VERSION - AERODYNAMICS ***

```
    XC[illegible] =   0.[illegible]    FC[illegible] =   1.0[illegible]   FA[illegible] =   1.0[illegible]
     A[illegible] =  40.0[illegible]      [illegible] =   [illegible]      [illegible] =   0.[illegible]
  SMNDF =   0.5000       [illegible] =   1.[illegible]      [illegible] =   1.0[illegible]
   FC[illegible] =   1.0000     FA[illegible] =   0.0[illegible]    [illegible] =   1.0[illegible]
 FLSCOR =   1.0[illegible]   FLF[illegible] =   1.[illegible]    [illegible] =   1.[illegible]
    FCL =   1.0000       [illegible] =   1.[illegible]00   FL[illegible] =   1.0000
 FDN[illegible] =   1.0[illegible]    FL[illegible] =   1.[illegible]00    [illegible] =   1.0[illegible]00
    FC[illegible] =   1.0000       FL[illegible] =   1.[illegible]    [illegible] =   1.0000
   FCLH =   1.0000     FCDLH =   1.[illegible]000     [illegible] =   1.0000
     YG =   0.0           IT =   0.0       [illegible] =   1.0000
   SFA[illegible] =   1.0000    RCLMAX =   1.0[illegible]    CL[illegible] =  -1.0000
 LOLAND =  -1.0000      CL[illegible] =  -1.0000      [illegible] =  -1.0000
  MACHN =   0.7500    SWPNT[illegible] =   0.0       SWP[illegible] =  60.0000
 DELFT[illegible] =  50.0000   DELT[illegible] =  [illegible].[illegible]  DELFL[illegible] =  60.0000
 DELF[illegible] =  [illegible].0000      FL[illegible]X =   0.0        [illegible] =   0.0
    SH[illegible] =  25.0000     [illegible] =   0.1[illegible]      CSF =   0.0
   [illegible]SSF =   1.0000       [illegible] =   1.0000        S[illegible] =   0.0
    CGM =   0.0         FFA[illegible] =   0.7[illegible]      [illegible] =   0.0[illegible]00

   IALP = 2     INT[illegible] = 0     ICOD = 1     [illegible] =10    [illegible]TL = 6
   IALF = 0     IPLOT = 0     [illegible] = 2   ISP[illegible] = 0   ISU[illegible] = 1
  INC[illegible] = 1    FCF[illegible] = 1    [illegible] = 0   K[illegible] = 0     IXC[illegible] = 0
   IAX[illegible] = 0    IPF[illegible] = 0     [illegible] = 0   IP[illegible] = 0    IPEXT = 0
 IP[illegible] = 0    IPINTF = 0    IPLIFT = 0    I[illegible] = 0    ITRAP = 0
  IFL[illegible] = 0    ELLIPW = F   [illegible] = F  ELLIPC = F

    SM[illegible] =     0.[illegible]      0.[illegible]      0.[illegible]      0.440      0.590
               0.[illegible]40      1.[illegible]      1.400      1.600      1.[illegible]00
   ALTV =      1000.      [illegible].     1[illegible].     15000.     15000.
               [illegible].     [illegible].     [illegible].     30000.     30000.
   ALIN =      0.0        0.[illegible]      1.000      1.500      2.000
               [illegible].[illegible]      [illegible]      6.000     10.000     12.000
  ISTAS =  0 0 0 0 0 0 0 0 0 0
    ITS =  0 0 0 0 0 0 0 0 0 0
    IT[illegible] =  0 0 0 0 0 0 0 0 0 0
  ITH[illegible] =  1 1 1 1 1 1 0 0 0 0
 SMTANK =      0.0        0.2[illegible]      [illegible].400      0.600      0.800
               1.000      [illegible]      1.[illegible]      1.700      1.800
  COTNK =      0.0        0.0        0.0        0.0        0.0
               0.0        0.0        0.0        0.0        0.0
 SMSTRS =      0.0        0.200      [illegible].400      0.600      0.800
               1.000      1.200      1.400      1.600      1.[illegible]00
  COSTR =      0.0        0.0        0.0        0.0        0.0
               0.0        0.0        0.0        0.0        0.0
 SM[illegible] =      0.0        0.[illegible]      0.400      0.600      0.[illegible]00
               1.000      [illegible]      1.400      1.600      1.800
  COBMB =      0.0        0.0        0.0        0.0        0.0
               0.0        0.0        0.0        0.0        0.0
    CLG =      0.0        0.0        0.0        0.0        0.0
               0.0        0.0        0.0        0.0        0.0
    CMG =      0.0        0.0        0.0        0.0        0.0
               0.0        0.0        0.0        0.0        0.0
 SM[illegible]SAP =      0.0        0.200      [illegible].[illegible]00      0.600      0.800
               1.[illegible]      1.[illegible]      1.400      1.600      1.[illegible]00
   F[illegible]F =      1.000      1.000      1.[illegible]      1.000      1.000
               1.000      1.000      1.[illegible]      1.000      1.000
   CALLING MODULE [illegible]  4
```

PROPULSION INPUT
VERSION 04-76

*** S3A "STOL VIKING" - [illegible] PROPULSION (TF- [illegible]) ***

ALPHTA	=	4.167	AEPLE	=	0.875	AEPWT	=	1425.000
ALTI	=	0.0	AP	=	0.0	ATOP3	=	0.385
AJACNG	=	0.0	[illegible]	=	0.0	[illegible]	=	6.230
DELFR	=	0.0	DELTS7	=	100.000	DEPACC	=	2.000
DIAL	=	4.167	[illegible]	=	0.750	SR	=	0.340
EDL	=	0.750	ETAC1	=	0.800	[illegible]	=	0.300
ETATI	=	0.920	HTF	=	0.600	HVF	=	18600.000
MACH1	=	0.490	MACH2	=	0.500	PCDFAC	=	1.000
P0SA	=	0.503	PREB	=	1.510	PWCC	=	100.000
P1LP1	=	0.600	PREF1	=	14.500	R10A	=	-1.000
R3P	=	0.370	[illegible]	=	1.000	[illegible]	=	1.000
R7L1	=	1.000	[illegible]	=	1.000	SCOP	=	0.967
SFADP	=	0.0	[illegible]	=	0.0	SFAUXP	=	0.0
SFHEP	=	0.0	SFADP	=	0.0	SFRTP	=	0.0
SFOTVP	=	0.0	SFTNCP	=	0.0	SFIP	=	0.0
SFSFC1	=	1.012	SFSFC2	=	0.955	SFSFC3	=	1.000
SPL	=	0.500	SFP	=	1.000	TOSA	=	518.700
TA	=	520.000	TWAB	=	0.	TWCAB	=	6841.
TATO	=	0.500	T5	=	2557.	T5M	=	3400.
T51	=	0.	T7A	=	3400.	T7'	=	0.
VC1	=	0.960	VCAN1	=	0.018	XMDES	=	0.850
XMF	=	0.650	[illegible]	=	72.	FPRT	=	0.0
FARN	=	0.0	RCIAN	=	1.150	[illegible]	=	1.537

IPR	=	1	IPRINT	=	0	IPLOT	=	0
KERROR	=	0	KCF	=	2	KT5	=	0
KT7	=	0	[illegible]	=	0	NAR	=	6
NOZZ	=	0	NFROP	=	6	NSUMM	=	10
						IENG	=	2 (TF30)

ALTD	=	0.	0.	5000.	5000.	15000.	15000.
XMACH	=	0.0	0.250	0.250	0.700	0.440	0.600
XMPET	=	0.0	0.300	0.600	0.900	1.200	4.000
XPRI	=	1.000	1.000	1.000	1.000	1.000	1.000

CALLING MODULE NUMBER 6

```
AIRCRAFT TYPE: [illegible]
TITLE: *** S3A [illegible] VIKING [illegible] WEIGHTS ***

CONTROL OPTIONS

    IPRINT =  1    [illegible] =  1    ITAIL   =  [illegible]   [illegible] =  0
    I[illegible]  =  0    [illegible] =  [illegible]    [illegible]  =  1    [illegible] =  1
    K1      = 1.00 K[illegible]      = [illegible]  [illegible]          = 1.[illegible] K[illegible]        = 1.0
    K5      = 1.00 K[illegible]      = 1.00  [illegible]  = [illegible]  [illegible]XIT  =  1
    K[illegible]      = 0.[illegible] KP1     = 0.[illegible]  K[illegible]      = [illegible]
    [illegible]      = 1.00 T[illegible]   = 1.00  [illegible]      =    47000.
    STRESS  =    [illegible].             [illegible]      =  0.356     [illegible] =0.0
    [illegible]   =  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00
  SLOPE(1) =  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00
  SLOPE(9) =  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00  1.00

INITIAL ESTIMATES
 IF CODE = 1, WEIGHT IS FIXED

QUANTITY       VALUE    CODE            QUANTITY       VALUE    CODE

  WAF       =     12226.     0            [illegible]     =      959.     1
  WAPU      =         0.     0            [illegible]     =     1060.     1
  W[illegible]       =       357.     1            [illegible]     =        0.     0
  W[illegible]       =         0.     0            [illegible]     =     5064.     1
  W[illegible]       =         0.     0            W[illegible]-GO =        0.     0
  W[illegible]       =       [illegible].     1            W[illegible]     =     5594.     1
  W[illegible]LT     =       [illegible].     1            W[illegible]     =     [illegible].     1
  W[illegible]TAIL   =         [illegible].     0            W[illegible]     =     [illegible]40.     0
  W[illegible]       =       [illegible].     1            W[illegible]     =      [illegible].     1
  WGEAR     =      [illegible].     1            W[illegible]     =      [illegible].     1
  WIT       =       [illegible].     1            WIEST     =      174.     1
  WLG       =      [illegible].     1            W[illegible]SS    =        0.     0
  W[illegible]       =      [illegible].     1            W[illegible]     =        0.     0
  WPASS     =         [illegible].     0            W[illegible]     =     [illegible].     0
  WPS       =      [illegible].     1            WSC       =     [illegible].     1
  WTSUM     =     [illegible].     0            W[illegible]     =      515.     1
  W[illegible]     =      [illegible].     1            [illegible]     =        0.     0
  WFUVE     =         0.     0            W[illegible]     =        0.     0
  WLIFT     =         0.     0

* * * BEGIN VEHICLE CONVERGING

ESTIMATED GROSS WEIGHT  = 0.47000E 05
CALLING MODULE [illegible]   1
CALLING MODULE [illegible]   2
CALLING MODULE [illegible]   6

CALCULATED GROSS WEIGHT = 0.[illegible]E 05

SLOPE OF WCALC VS WEST LINE = 0.75[illegible]E 00

ESTIMATED GROSS WEIGHT  = 0.46111E 05
CALLING MODULE [illegible]   1
CALLING MODULE [illegible]   2
CALLING MODULE [illegible]   6

CALCULATED GROSS WEIGHT = 0.4[illegible]E 05

SLOPE OF WCALC VS WEST LINE = 0.30[illegible]E 00

ESTIMATED GROSS WEIGHT  = 0.46742E 05
CALLING MODULE [illegible]   1
CALLING MODULE [illegible]   2
CALLING MODULE [illegible]   6

CALCULATED GROSS WEIGHT = 0.46757E 05

SLOPE OF WCALC VS WEST LINE = 0.[illegible]E 00
```

```
    A   ... VEHICLE ... ENDS
    CALLING MODULE NUMBER  1
    CALLING MODULE NUMBER  2
SUMMARY OF NON FATAL ERRORS ENCOUNTERED IN PROP
P5 IS LESS THAN P65     1  TIMES

IN LOITER, PHASE 5, LEG 1, FRAG = 1.001 OF TN.
CD CHANGED TO MATCH THRUST.

IN LOITER, PHASE 5, LEG 2, DRAG = 1.001 OF TN.
CD CHANGED TO MATCH THRUST.

IN LOITER, PHASE 8, LEG 1, DRAG = 1.001 OF TN.
CD CHANGED TO MATCH THRUST.

    FJET COULD NOT BE FOUND FROM FUNI     CL, CD, KP SHOULD BE CHECKED

    AERODYNAMICS MODULE

MACH OR ALTITUDE SEARCH CONCLUDED.
    X = 0.33157E 05  Y = 0.21813E 02
    CALLING MODULE NUMBER  6
    CALLING MODULE NUMBER  1
```

NASA/AMES A.C.S.Y.N.T [illegible] VERSION [illegible] FINAL OUTPUT

FUSELAGE GEOMETRY

X	R	AREA
0.0	0.0	0.0
0.98	1.20	5.[illegible]1
1.97	2.03	13.[illegible]6
2.95	2.66	22.[illegible]1
3.94	3.[illegible]1	3[illegible].46
4.92	3.45	37.42
5.91	3.69	42.65
6.89	3.83	45.95
7.88	3.87	47.17
8.86	3.88	47.17
9.85	3.88	47.17
10.83	3.88	47.17
11.82	3.88	47.17
12.80	3.88	47.17
13.79	3.88	47.17
14.77	3.88	47.17
15.76	3.88	47.17
16.74	3.88	47.17
17.73	3.88	47.17
18.71	3.88	47.17
19.70	3.88	47.17
20.68	3.88	47.17
21.67	3.88	47.17
22.65	3.88	47.17
23.64	3.88	47.17
24.62	3.88	47.17
25.61	3.88	47.17
26.59	3.87	47.08
27.58	3.85	46.74
28.56	3.83	46.16
29.55	3.80	4[illegible]
30.53	3.75	[illegible]
31.52	3.70	42.97
32.50	3.65	41.94
33.49	3.55	3[illegible].70
34.47	3.47	37.76
35.46	3.37	[illegible]
36.44	3.26	3[illegible].50
37.43	3.15	3[illegible].[illegible]1
38.41	3.03	[illegible]
39.40	2.85	2[illegible].94
40.38	2.66	22.55
41.37	2.50	1[illegible].[illegible]7
42.35	2.31	16.71
43.34	2.09	13.75
44.32	1.86	10.[illegible]9
45.31	1.60	8.03
46.29	1.31	5.40
47.28	0.98	3.04
48.26	0.59	1.11
49.25	0.0	0.0

NACELLE GEOMETRY / NACELLE LOCATION

X-RATIO	R	AREA	X	Y	Z
0.0	1.75	11.93	14.51	7.75	3.70
3.33	1.95	11.93	14.51	-7.75	3.70
10.00	1.75	11.93	16.42	13.19	3.70
13.33	1.95	11.93	16.42	-13.19	3.70

	FUSELAGE		**PODS**
MAX. DIAMETER.......	7.756	.....	3.906
FINENESS RATIO.....	[illegible]		
SURFACE AREA........	555.115	.....	164.630 (EACH)
VOLUME..............	[illegible]		

NASA/AMES [illegible] FINAL OUTPUT

DIMENSIONS OF [illegible] SURFACES

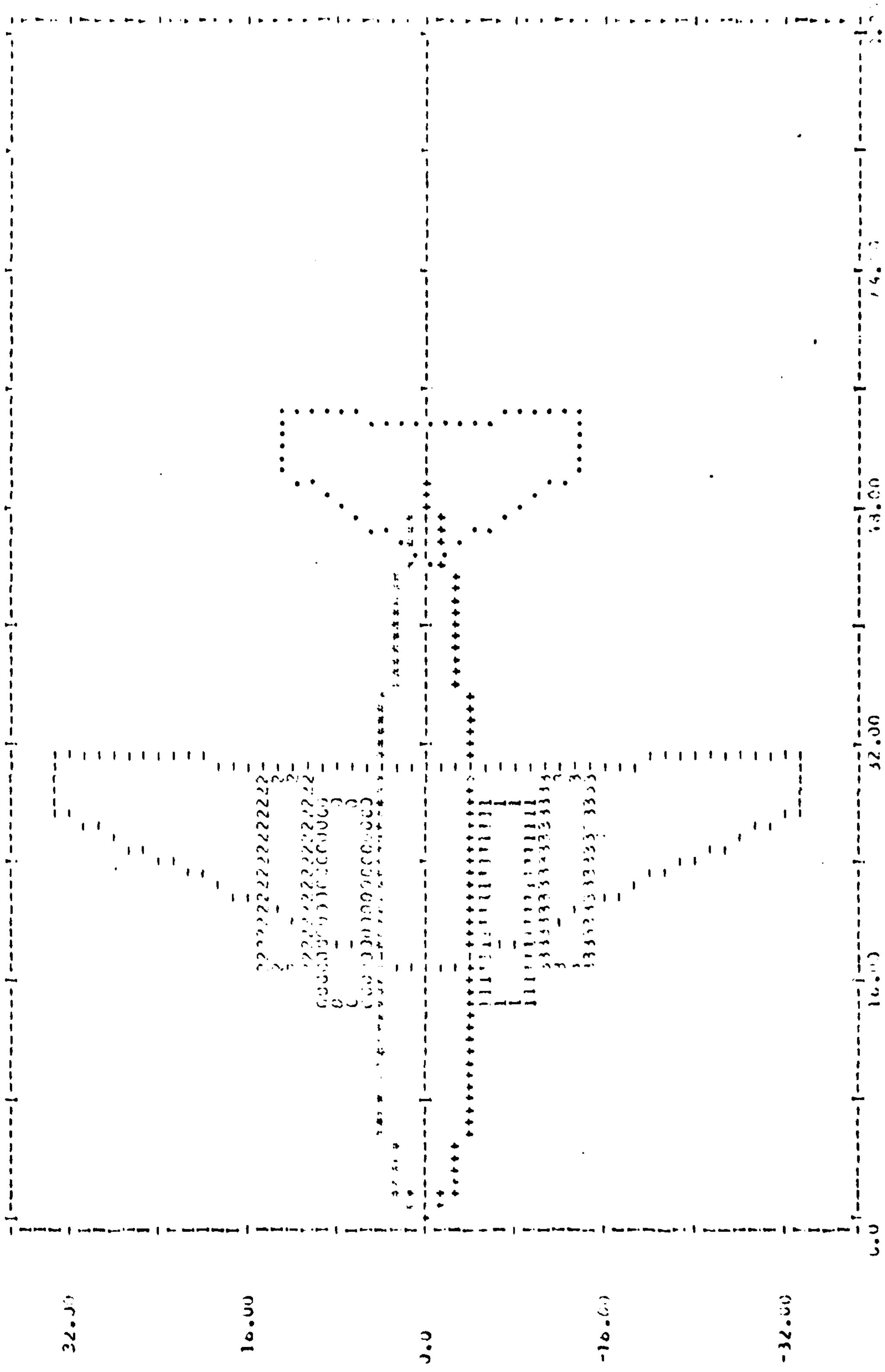
	WING	H.TAIL	V.TAIL	CANARD	UNITS
PLAN AREA...........	[illegible]	[illegible]	[illegible]	0.0	(SQ.FT.)
SURFACE AREA........	[illegible]	[illegible]	[illegible]	0.0	(SQ.FT.)
VOLUME..............	[illegible]	[illegible]	[illegible]	0.0	(CU.FT.)
SPAN................	[illegible]	27.000	[illegible]	0.0	(FT.)
L.E. SWEEP..........	[illegible]	[illegible]	[illegible]	0.0	(DEG.)
C/4 SWEEP...........	[illegible]	[illegible]	[illegible]	0.0	(DEG.)
T.E. SWEEP..........	[illegible]	3.921	16.2[illegible]	0.0	(DEG.)
ASPECT RATIO........	7.73[illegible]	4.0[illegible]	[illegible]	0.0	
ROOT CHORD..........	14.[illegible]	7.324	14.[illegible]	0.0	(FT.)
ROOT THICKNESS......	27.[illegible]	11.16[illegible]	2[illegible]	0.0	(IN.)
ROOT T/C............	[illegible]	[illegible]	[illegible]	0.0	
TIP CHORD...........	[illegible]	4.[illegible]	[illegible]	0.0	(FT.)
TIP THICKNESS.......	[illegible]	[illegible]	[illegible]	0.0	(IN.)
TIP T/C.............	[illegible]	[illegible]	[illegible]	0.0	
TAPER RATIO.........	[illegible]	[illegible]	[illegible]	0.0	
M.A.C. CHORD........	[illegible]	7.[illegible]	1[illegible]	0.0	(FT.)
LE ROOT AT..........	[illegible]	[illegible]	[illegible]	0.0	(FT.)
C/4 ROOT AT.........	[illegible]	[illegible]	33.715	0.0	(FT.)
TE ROOT AT..........	[illegible]	[illegible]	49.[illegible]	**********	(FT.)
LE M.A.C. AT........	20.7[illegible]	47.[illegible]	[illegible]	**********	(FT.)
C/4 M.A.C. AT.......	[illegible]	[illegible]	[illegible]	**********	(FT.)
TE M.A.C. AT........	[illegible]	[illegible]	[illegible]	**********	(FT.)
Y M.A.C. AT.........	[illegible]	[illegible]	[illegible]	**********	
LE TIP AT...........	27.331	[illegible]	48.309	0.000	(FT.)
C/4 TIP AT..........	28.71[illegible]	51.[illegible]	4[illegible]	0.000	(FT.)
TE TIP AT...........	31.34[illegible]	[illegible]	53.225	-0.439	(FT.)
ELEVATION...........	[illegible]	17.841	4.[illegible]	0.0	(FT.)
VOLUME COEFF.		0.730	0.063	0.0	

F U E L T A N K S

TANK	VOLUME	WEIGHT	DENSITY
WING	2[illegible]	13134.	50.00
FUSE1	0.	0.	50.00
FUSE2	0.	0.	50.00
TOTAL		13134.	

MISSION FUEL REQUIRED= 12[illegible]4.

AVAILABLE FUEL VOLUME IN WING= 454.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16.00
0.0
-16.00
-32.00
32.00

AD-A089 492 NAVAL POSTGRADUATE SCHOOL MONTEREY CA F/G 1/3
APPLICATION OF POWERED HIGH LIFT SYSTEMS TO STOL AIRCRAFT DESIG--ETC(U)
SEP 79 F D AMEEL

UNCLASSIFIED NL

3 of 3
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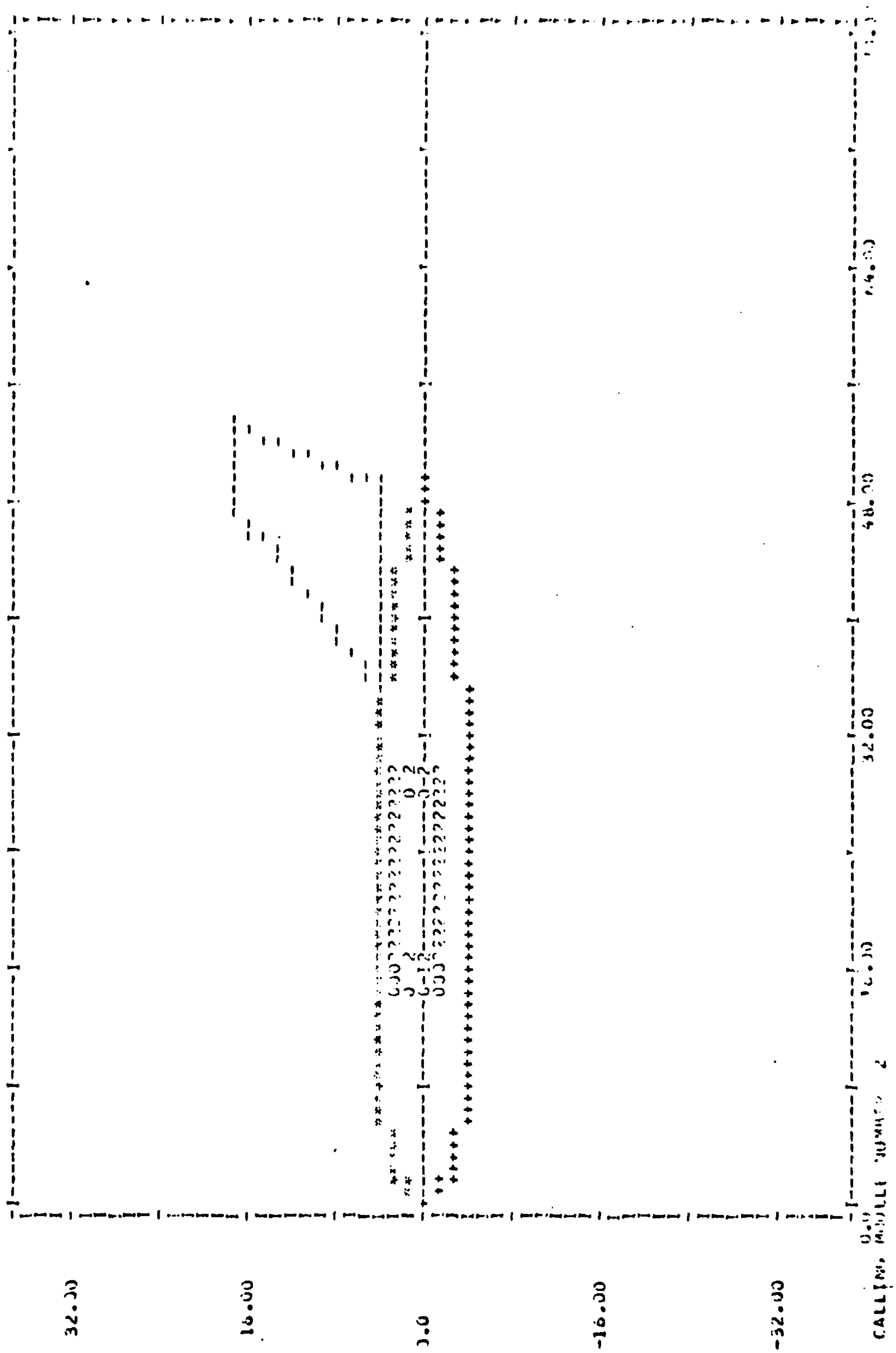
32.00
16.00
0.0
-16.00
-32.00
32.00
48.00

TRAJECTORY OUTPUT

MISSION 1 (PAYLOAD = 2247. LB)

PHASE	M SFC(I) SFC(L)	H THRUST(I) THRUST(L)	CL CD CDIRST	ALPHA L/D	WFUEL W	TIME DT ??	VEL Q X
ACCEL	0.35 0.50 0.50	0. 20797. 20797.	0.4317 0.0354 0.0	4.09 13.02	25.7 46468.8	0.15 228.23 1.00	568. 161. 1.
CLIMB	0.56 0.56 0.56	15000. 13114. 13114.	0.2776 0.0267 0.0	2.16 10.33	339.8 46128.9	2.26 165.54 1.00	591. 261. 10.
ACCEL	0.59 0.56 0.56	15000. 11945. 11945.	0.2047 0.02[illegible] 0.0	2.01 10.05	53.2 46075.8	0.48 156.14 1.00	624. 291. [illegible]5.
CRUISE	0.59 0.61 0.61	15000. 4479. 449[illegible].	0.2459 0.0253 0.0	1.56 9.59	4497.3 40518.4	73.80 139.67 1.00	624. 291. 454.
LOITER	0.34 0.77 0.77	100. 3233. [illegible]	0.3857 0.0308 0.0	3.24 12.53	2497.5 34020.9	60.00 171.90 1.00	335. 176. 228.
CLIMB	0.62 0.56 0.56	10000. 15435. 15435.	0.2214 0.0252 0.0	1.75 8.80	195.1 37825.8	1.21 190.85 1.00	558. 273. 5.
CRUISE	0.74 0.74 0.74	10000. 2591. 2591.	0.4775 0.0357 0.0	4.04 13.37	4187.4 33638.5	126.21 130.85 1.00	372. 121. 46[illegible].
LOITER	0.34 0.78 0.78	100. 2973. 2973.	0.3232 0.0265 0.0	2.63 11.32	192.6 33445.9	5.00 177.08 1.00	335. 176. 19.

```
                         FUEL SUMMARY

TAKEOFF FUEL
         WETO1 =      72.      TOTAL FUEL =  12974.
         WETO2 =     173.        EXTERNAL =      0.
  MISSION FUEL =  12[illegible].        INTERNAL =  12974.
  RESERVE FUEL =     644.
  TRAPPED FUEL =     100.

                           BLOCK TIME  =   4.455   HOURS
                          BLOCK RANGE  = 1153.0    N. M.
TAKEOFF FIELD LENGTH(TOTAL FUEL)       =   2192.   FEET
LANDING FIELD LENGTH(TOTAL FUEL)       =   3776.   FEET
LANDING FIELD LENGTH(RESERVE FUEL)     =   3033.   FEET
  WEIGHT FOR LANDING CALCULATION       =  39765.   POUNDS
                       TAKEOFF WEIGHT  =  46736.   POUNDS
                       LANDING WEIGHT  =  33940.   POUNDS
                  ENDURANCE MACH NO.   =   0.3
                  ENDURANCE ALTITUDE   =  3[illegible]197.   FEET
                  ENDURANCE TIME       =   0.618   HOURS
                       LOITER RADIUS   =   0.0     N. M.
                       LOITER RADIUS   =   0.0     N. M.
    CALLING MODULE NUMBER  3
```

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.330 ALTITUDE = 1000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0000	WAVE =	0.0
STORE =	0.0	FRIC =	0.0165
TANK =	0.0	BLUNT =	0.0
BOMBS =	0.0	ENGINE =	0.0

FRICTION = 0.0038 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0231 (NO CAMBER)
= 0.0231 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CMNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0231	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0607	0.0233	0.0002	0.0	0.0000	0.0014	2.6074	0.0000	0.04	-0.0123
1.00	0.1207	0.0235	0.0005	0.0	0.0000	0.0028	5.0232	0.0001	0.09	-0.0243
1.50	0.1804	0.0249	0.0017	0.0	0.0001	0.0041	7.2607	0.0001	0.12	-0.0363
2.00	0.2396	0.0261	0.0030	0.0	0.0002	0.0055	9.1637	0.0002	0.15	-0.0485
4.00	0.4713	0.0345	0.0113	0.0	0.0008	0.0111	13.5077	0.0009	0.37	-0.0967
6.00	0.6957	0.0491	0.0260	0.0	0.0017	0.0166	14.1626	0.0020	0.57	-0.1510
8.00	0.9134	0.0685	0.0454	0.0	0.0031	0.0221	13.3512	0.0035	0.77	-0.2057
10.00	1.1247	0.0923	0.0693	0.0	0.0049	0.0277	12.1155	0.0054	1.03	-0.2623
12.00	1.5302	0.1219	0.0988	0.0	0.0070	0.0332	10.9125	0.0076	1.23	-0.3227

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.340 ALTITUDE = 5000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0066	WAVE =	0.0		
STORE =	0.0	FRIC =	0.0167		
TANK =	0.0	PLUMT =	0.0		
BLADS =	0.0	ENGINE =	0.0		
		FRICTION =	0.0039	(INCL. IN FRIC)	
		BOATTAIL =	0.0		
		COWL =	0.0		

CD(MIN) = 0.0233 (NO CAMBER)
= 0.0233 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDMISE	CLMISE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0233	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0608	0.0235	0.0002	0.0	0.0000	0.0014	2.5900	0.0000	0.04	-0.0122
1.00	0.1211	0.0241	0.0008	0.0	0.0000	0.0028	5.0349	0.0001	0.09	-0.0241
1.50	0.1807	0.0250	0.0017	0.0	0.0001	0.0042	7.2344	0.0001	0.13	-0.0364
2.00	0.2401	0.0263	0.0030	0.0	0.0002	0.0055	9.1240	0.0002	0.18	-0.0487
4.00	0.4724	0.0352	0.0119	0.0	0.0008	0.0111	13.4390	0.0009	0.37	-0.0991
6.00	0.6973	0.0494	0.0262	0.0	0.0017	0.0166	14.1049	0.0020	0.57	-0.1517
8.00	0.9155	0.0689	0.0456	0.0	0.0031	0.0221	13.2865	0.0035	0.79	-0.2065
10.00	1.1272	0.0933	0.0700	0.0	0.0049	0.0276	12.0810	0.0054	1.03	-0.2640
12.00	1.3330	0.1225	0.0992	0.0	0.0070	0.0330	10.8860	0.0076	1.28	-0.3240

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.550 ALTITUDE = 10000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0067	WAVE =	0.0
STORE =	0.0	FRIC =	0.0169
TANK =	0.0	BLUNT =	0.0
ROMBS =	0.0	ENGINE =	0.0

FRICTION = 0.0039 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0236 (NO CAMBER)
= 0.0236 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0236	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0610	0.0235	0.0002	0.0	0.0000	0.0014	2.5634	0.0000	0.04	-0.0121
1.00	0.1214	0.0244	0.0008	0.0	0.0000	0.0023	4.9842	0.0001	0.09	-0.0242
1.50	0.1814	0.0253	0.0017	0.0	0.0001	0.0042	7.1670	0.0001	0.13	-0.0365
2.00	0.2403	0.0266	0.0030	0.0	0.0002	0.0055	9.1383	0.0002	0.18	-0.0489
4.00	0.4750	0.0355	0.0119	0.0	0.0008	0.0110	13.3381	0.0009	0.37	-0.0996
6.00	0.6950	0.0439	0.0263	0.0	0.0017	0.0166	14.0226	0.0020	0.57	-0.1523
8.00	0.9170	0.0654	0.0458	0.0	0.0031	0.0221	13.2250	0.0036	0.79	-0.2074
10.00	1.1238	0.0949	0.0703	0.0	0.0049	0.0275	12.0354	0.0055	1.03	-0.2651
12.00	1.3359	0.1231	0.0995	0.0	0.0070	0.0329	10.8517	0.0077	1.29	-0.3254

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.640 ALTITUDE = 15000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0008		WAVE =	0.0
STORE =	0.0		FRIC =	0.0166
TANK =	0.0		BUMPT =	0.0
BOMBS =	0.0		ENGINE =	0.0

FRICTION = 0.0039 (INCL. IN FRIC)
POSTTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0234 (NO CAMBER)
= 0.0234 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0234	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0627	0.0236	0.0002	0.0	0.0000	0.0014	2.6501	0.0000	0.04	-0.0125
1.00	0.1245	0.0242	0.0008	0.0	0.0000	0.0028	5.1457	0.0001	0.09	-0.0254
1.50	0.1863	0.0252	0.0018	0.0	0.0001	0.0042	7.3807	0.0001	0.13	-0.0357
2.00	0.2471	0.0266	0.0032	0.0	0.0002	0.0055	9.2881	0.0003	0.19	-0.0511
4.00	0.4864	0.0359	0.0125	0.0	0.0008	0.0110	13.5438	0.0010	0.37	-0.1039
6.00	0.7175	0.0509	0.0275	0.0	0.0017	0.0166	14.0996	0.0022	0.54	-0.1583
8.00	0.9415	0.0713	0.0478	0.0	0.0031	0.0221	13.2122	0.0039	0.81	-0.2159
10.00	1.1946	0.0997	0.0733	0.0	0.0049	0.0276	11.9764	0.0060	1.05	-0.2755
12.00	1.3691	0.1271	0.1037	0.0	0.0070	0.0330	10.7729	0.0085	1.31	-0.3378

DETAILED AERODYNAMICS
AESYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.550 ALTITUDE = 15000.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0070	WAVE =	0.0
STORE =	0.0	FRIC =	0.0157
TANK =	0.0	BLUNT =	0.0
BOMBS =	0.0	ENGINE =	0.0

FRICTION = 0.0037 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0227 (NO CAMBER)
= 0.0227 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0227	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0669	0.0230	0.0002	0.0	0.0000	0.0014	2.9115	0.0000	0.05	-0.0140
1.00	[illegible]	[illegible]	0.0009	0.0	0.0000	0.0028	5.6306	0.0001	0.09	-0.0280
1.50	[illegible]	[illegible]	0.0020	0.0	0.0001	0.0041	8.0275	0.0002	0.14	-0.0422
2.00	[illegible]	[illegible]	0.0036	0.0	0.0002	0.0055	10.0277	0.0003	0.19	-0.0566
4.00	0.5154	0.0367	0.0139	0.0	0.0008	0.0111	14.1364	0.0013	0.39	-0.1142
6.00	0.7642	0.0533	0.0366	0.0	0.0017	0.0166	14.3280	0.0028	0.60	-0.1739
8.00	1.0015	0.0759	0.0531	0.0	0.0031	0.0222	13.1991	0.0048	0.83	-0.2357
10.00	1.2513	0.1039	0.0812	0.0	0.0049	0.0278	11.3420	0.0074	1.07	-0.2993
12.00	1.4440	0.2901	0.2673	0.0	0.0071	0.0334	4.9778	0.0094	2.61	-0.3662

DETAILED AERODYNAMICS
ACSYNT MODULE NUMBER 3
MODULE VERSION 04-76

MACH = 0.340 ALTITUDE = 100.

DRAG COEFFICIENTS, INDEPENDENT OF ALPHA

INTF =	0.0066	WAVE =	0.0
STRAL =	0.0	FRIC =	0.0163
TANK =	0.0	BLUNT =	0.0
BUMPS =	0.0	ENGINE =	0.0

FRICTION = 0.0038 (INCL. IN FRIC)
BOATTAIL = 0.0
COWL = 0.0

CD(MIN) = 0.0229 (NO CAMBER)
= 0.0229 (WITH CAMBER)

ALPHA DEPENDENT COEFFICIENTS

ALPHA	CL	CD	CDI	CDSEP	CDNOSE	CLNOSE	L/D	CDTRIM	ALTRIM	CM
0.0	0.0	0.0229	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.50	0.0603	0.0231	0.0002	0.0	0.0000	0.0014	2.6283	0.0000	0.04	-0.0120
1.00	0.1211	0.0237	0.0008	0.0	0.0000	0.0029	5.1074	0.0001	0.09	-0.0241
1.50	0.1809	0.0247	0.0017	0.0	0.0001	0.0041	7.3343	0.0001	0.13	-0.0364
2.00	0.2401	0.0260	0.0030	0.0	0.0002	0.0055	9.2442	0.0002	0.18	-0.0487
4.00	0.4724	0.0348	0.0119	0.0	0.0008	0.0111	13.5712	0.0009	0.37	-0.0991
6.00	0.6974	0.0491	0.0262	0.0	0.0017	0.0166	14.2032	0.0020	0.57	-0.1516
8.00	0.9155	0.0686	0.0456	0.0	0.0031	0.0222	13.3524	0.0035	0.79	-0.2065
10.00	1.1277	0.0930	0.0700	0.0	0.0049	0.0277	12.1247	0.0054	1.03	-0.2636
12.00	1.3332	0.1221	0.0952	0.0	0.0071	0.0332	10.9153	0.0076	1.28	-0.3237

CALLING MODULE NUMBER 4

ENGINE SUMMARY

```
ENGINE DIAMETER =      4.17 FEET
ENGINE LENGTH   =      8.67 FEET
ENGINE WEIGHT   =   1425.00 POUNDS
BYPASS RATIO    =      6.25
NO. OF ENGINES  =      4.
DRAG REF AREA   =    5[illegible].00 SQ FEET
PACU    = PER CENT OF ENGINE CORRECTED AIRFLOW
THRUST  = ENGINE THRUST (POUNDS PER ENGINE)
SFC     = ENGINE SPECIFIC FUEL CONSUMPTION
THRUSTU = THRUST PER ENGINE IN LBS, W/O INSTAL DRAG CORR
SFCU    = SFC,1/HR, W/O INSTALLATION DRAG CORR
CDINS   = TOT INSTALLATION DRAG COEF PER A/C (SWING REF)
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MACH	ALT	PWRC	THRUST	THRUSTN	SFC	SFCN	COINS
0.0	0.	100.	6841.	6841.	0.361	0.361	0.0
		98.	6426.	6426.	0.368	0.368	0.0
		96.	5805.	[illegible]	0.363	0.363	0.0
		94.	5128.	5133.	0.363	0.363	0.0
		92.	4350.	4350.	0.379	0.379	0.0
		90.	3295.	3295.	0.449	0.449	0.0
		88.	1766.	1766.	0.713	0.713	0.0
		86.	1553.	1553.	0.685	0.685	0.0
		84.	1425.	1425.	0.657	0.657	0.0
		82.	1260.	1260.	0.632	0.632	0.0
0.250	0.	100.	5468.	5468.	0.464	0.464	0.0
		98.	5148.	5148.	0.494	0.494	0.0
		96.	4964.	4964.	0.487	0.487	0.0
		94.	3937.	3937.	0.500	0.500	0.0
		92.	3273.	[illegible]	0.529	0.529	0.0
		90.	2301.	2301.	0.662	0.662	0.0
		88.	1644.	1644.	0.808	0.808	0.0
		86.	1469.	1469.	0.783	0.783	0.0
		84.	1299.	1299.	0.760	0.760	0.0
		82.	1133.	1133.	0.742	0.742	0.0
0.250	5000.	100.	4687.	4687.	0.470	0.470	0.0
		98.	4273.	4273.	0.473	0.473	0.0
		96.	3738.	3733.	0.475	0.475	0.0
		94.	3267.	3267.	0.499	0.498	0.0
		92.	2674.	2674.	0.526	0.526	0.0
		90.	1908.	1908.	0.647	0.647	0.0
		88.	1362.	1362.	0.790	0.790	0.0
		86.	1217.	1217.	0.765	0.765	0.0
		84.	1075.	1075.	0.744	0.744	0.0
		82.	928.	928.	0.725	0.725	0.0
0.700	5000.	100.	4070.	4070.	0.615	0.615	0.0
		98.	3868.	3868.	0.642	0.642	0.0
		96.	3562.	3562.	0.675	0.675	0.0
		94.	3051.	3051.	0.699	0.699	0.0
		92.	2518.	2518.	0.747	0.747	0.0
		90.	1945.	1945.	0.848	0.848	0.0
		88.	1576.	1576.	0.913	0.913	0.0
		86.	1394.	1394.	0.893	0.893	0.0
		84.	1215.	1219.	0.877	0.877	0.0
		82.	1050.	1050.	0.867	0.867	0.0
0.440	15000.	100.	2984.	2984.	0.541	0.541	0.0
		98.	2665.	2669.	0.541	0.541	0.0
		96.	2347.	2347.	0.548	0.548	0.0
		94.	2009.	2009.	0.567	0.567	0.0
		92.	1641.	1641.	0.613	0.613	0.0
		90.	1211.	1211.	0.728	0.728	0.0
		88.	934.	934.	0.824	0.824	0.0
		86.	829.	829.	0.803	0.803	0.0
		84.	728.	723.	0.785	0.785	0.0
		82.	630.	630.	0.772	0.772	0.0
0.600	15000.	100.	2991.	2991.	0.583	0.583	0.0
		98.	2665.	2665.	0.585	0.585	0.0
		96.	2337.	2337.	0.595	0.595	0.0
		94.	1999.	1999.	0.617	0.617	0.0
		92.	1642.	1642.	0.662	0.662	0.0
		90.	1243.	1243.	0.764	0.764	0.0
		88.	995.	995.	0.835	0.835	0.0
		86.	881.	881.	0.817	0.817	0.0
		84.	771.	771.	0.801	0.801	0.0
		82.	665.	665.	0.791	0.791	0.0

WEIGHT STATEMENT - NUMBERS
T-1 C3A "STOL VIKING" - WEIGHTS ***
ACSYNT MODULE NUMBER 6
MODULE VERSION 4-76

COMPONENT	POUNDS	KILOGRAMS	PERCENT
AIRFRAME STRUCTURE	15238.	6912.	32.60
WING	5505.	2497.	11.73
FUSELAGE	5058.	2299.	10.64
HORIZONTAL TAIL	800.	363.	1.71
VERTICAL TAIL	585.	265.	1.25
NACELLES	1610.	730.	3.45
ALIGHTING GEAR	1670.	757.	3.57
PROPULSION	6250.	2835.	13.37
ENGINES (4)	5904.	2678.	12.63
FUEL SYSTEM	346.	157.	0.74
FIXED EQUIPMENT	10029.	4549.	21.47
HYD. + PNEU.	389.	176.	0.83
ELECTRICAL	1085.	492.	2.32
AVIONICS	4353.	1974.	9.31
INSTRUMENTATION	174.	79.	0.37
DE-ICE/AIR CONDITION	959.	435.	2.05
AUXILIARY GEAR	1144.	519.	2.45
FURNISH. + EQPT.	300.	136.	0.64
FLIGHT CONTROLS	1604.	728.	3.43
FUEL	12974.	5885.	27.76
PAYLOAD	2267.	1028.	4.85
FLIGHT CREW (4)	850.	386.	1.82
ARMAMENT	357.	162.	0.76
AMMUNITION	1060.	481.	2.27
MISSILES	0.	0.	0.0
BOMBS	0.	0.	0.0
EXTERNAL TANKS	0.	0.	0.0
ADV. WEAPONS 1	0.	0.	0.0
ADV. WEAPONS 2	0.	0.	0.0
CALCULATED WEIGHT	46738.	21200.	100.00
ESTIMATED WEIGHT	46736.	21200.	
PERCENT ERROR	0.00		

CALLING MODULE NUMBER 11

SUMMARY --- ACSYNT OUTPUT --- NASA, AMES RESEARCH CENTER

ENGLISH UNITS -
[illegible]
WEIGHTS IN LBS.
SPEEDS IN KTS.
PRESSURES IN LBS/FT**2

*** LOCKHEED S3A "STOL VIKING" ANALYSIS ***

GENERAL		FUSELAGE	
WG	46736.	LENGTH	49.4
W/S	78.2	DIAMETER	7.8
T/W	0.59	VOLUME	1714.0
N(Z) ULT	5.3	WETTED AREA	999.0
CREW.	4.	FINENESS RATIO	6.4
PASENGERS	0.		

	WING	HTAIL	VTAIL
AREA	568.0	180.0	139.3
WETTED AREA	1042.2	361.5	259.6
SPAN	68.0	27.0	15.6
L.E. SWEEP	19.1	24.8	48.5
C/4 SWEEP	15.0	20.0	38.5
ASPECT RATIO	7.7	4.0	1.4
TAPER RATIO	0.3	0.4	0.3
T/C ROOT	0.16	0.10	0.12
T/C TIP	0.12	0.10	0.10
ROOT CHORD	14.1	9.3	14.0
TIP CHORD	3.5	4.0	4.9
M.A. CHORD	9.9	7.0	10.2
LOC. OF L.E.	19.1	44.3	35.2

ENGINE		WEIGHTS	W	%WG
NUMBER	4.			
LENGTH	8.7	STRUCT.	15238.	32.6
DIAM.	2.4	PROPUL.	6290.	13.4
WEIGHT	1925.0	FIX. EQ.	10009.	21.4
TSLS	6841.	FUEL	12974.	27.8
SFCSLS	0.36	PAYLOAD	2267.	4.5

MISSION SUMMARY

PHASE	MACH	ALT	FUEL	TIME	DIST	L/D	THRUST	SFC	Q
TAKEOFF		0.	242.	5.0	2151.8				
ACCEL	0.33	0.	36.	0.2	0.5	13.6	20796.0	0.5	161.3
CLIMB	0.56	15000.	340.	2.3	10.4	10.4	13113.3	0.6	261.2
ACCEL	0.59	15000.	53.	0.5	2.7	10.1	11945.2	0.6	[illegible]
CRUISE	0.59	15000.	4497.	73.8	494.4	9.5	4493.9	0.8	291.1
LOITER	0.34	100.	2497.	60.0	228.0	12.5	3232.5	0.8	175.7
CLIMB	0.32	10000.	195.	1.2	5.0	8.9	15434.9	0.6	273.0
CRUISE	0.34	10000.	4187.	126.2	463.0	13.4	2591.0	0.7	121.3
LOITER	0.34	100.	193.	5.0	19.0	11.3	2975.2	0.8	175.7
LANDING					3775.7				

BLOCK TIME = 4.495 HR
BLOCK RANGE = 1163.0 NM

IHC900I EXECUTION TERMINATING DUE TO ERROR COUNT FOR ERROR NUMBER 217

IHC217I FIOCS - END OF DATA SET ON UNIT 5

TRACEBACK	ROUTINE	CALLED FROM ISN	REG. 14	REG. 15	REG. 0	REG. 1
	IBCOM		00045EA8	0008AF78	00000000	00000000
	COPEOI	0012	4204D4BE	0004DA00	00000028	00040044
	MAIN		0000EE22	0104D006	F0000008	00051FF3

ENTRY POINT= 0104D0F6

APPENDIX - C

CALCULATION OF BALANCED FIELD LENGTH

A. Expression for $C_{L_{LO}}$

1. At liftoff: $\text{LIFT} = \text{WEIGHT} = C_{L_{LO}} q_{LO} S = C_{L_{LO}} (\frac{1}{2} \rho V_{LO}^2) S$

$$C_{L_{LO}} = 2(W/S) / \rho V_{LO}^2$$

2. In Section III the liftoff velocity was defined.

a. $V_{LO} = 1.20\ V_S$; $V_S \equiv V_{STALL}$

b. $V_S^2 = 2(W/S) / \rho C_{L_{MAX}}$

c. $V_{LO}^2 = (1.20)^2 V_S^2 = 1.44 \left[2(W/S) / \rho\, C_{L_{MAX}} \right]$

3. Then substituting in the expression for $C_{L_{LO}}$ in 1. above:

$$C_{L_{LO}} = C_{L_{MAX}} / 1.44 = (0.694)\ C_{L_{MAX}}$$

B. Determination of V_{LO} and $C_{L_{LO}}$

1. Since the dynamic pressure at liftoff (q_{LO}) is a function of (V_{LO}^2) and by definition the momentum coefficient (C_μ) is an inverse function of q_{LO}, the following scheme was utilized to determine V_{LO} for the STOL VIKING computer run (APPENDIX B).

a. Select a liftoff velocity (V_{LO}).

b. Calculate V_S : $V_S = V_{LO} / 1.20$

c. Calculate q_{LO}: $q_{LO} = (0.001711)\ V_S^2$ at sea level, standard day conditions.

d. From LOCKHEED data look up engine fan thrust ($T_{F_{LO}}$) at the selected V_{LO}.

e. Calculate C_μ : $C_\mu = 4(\ T_{F_{LO}}) / q_{LO} S$

f. From plot of "$C_{L_{MAX}} - C_\mu$", Figure C-1, determine $C_{L_{MAX}}$.

g. Calculate $C_{L_{LO}}$: $C_{L_{LO}} = (0.694)\ C_{L_{MAX}}$

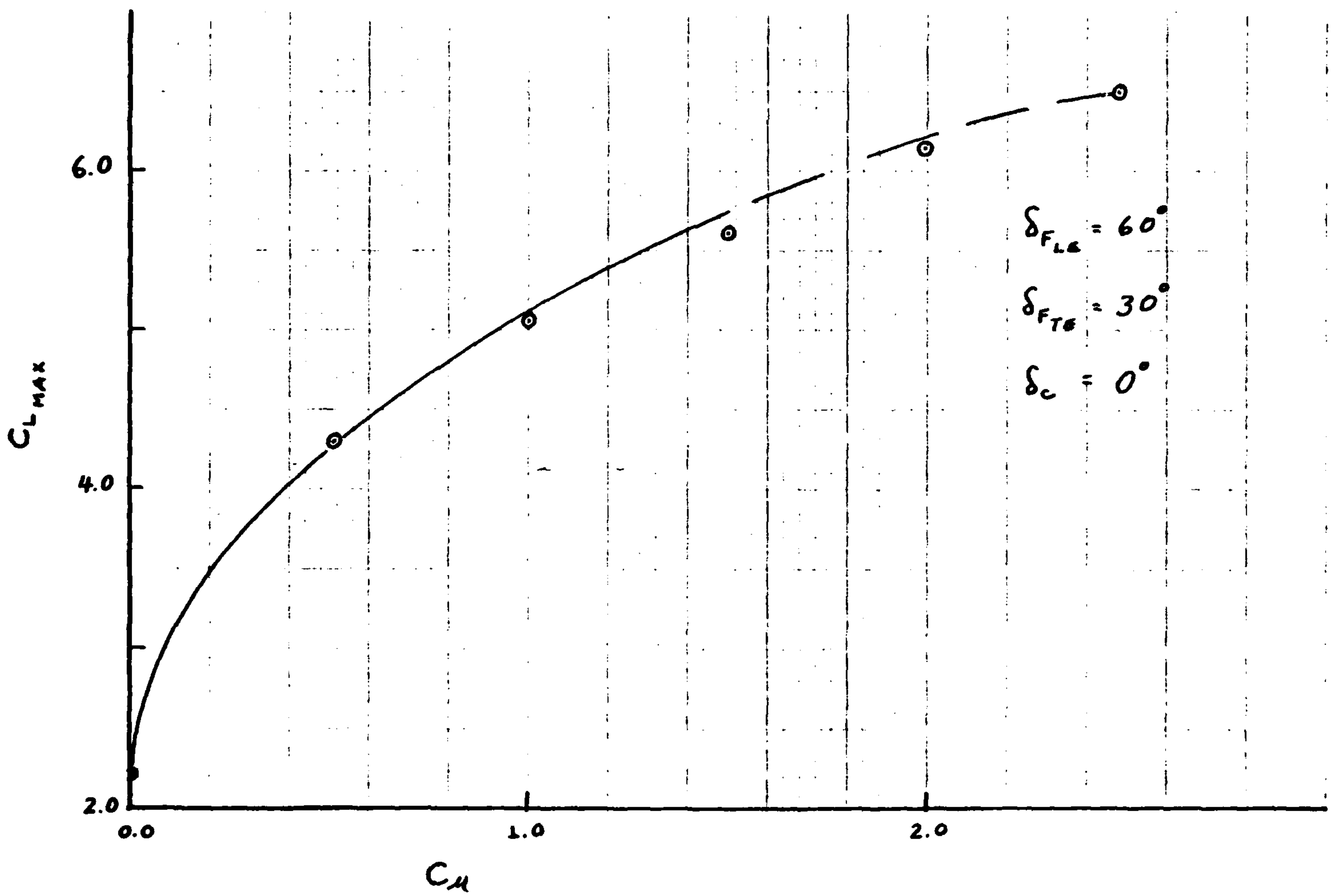

FIGURE C-1 (REF. 5-6)

h. Now calculate V_{Lo_c} using: $V_{Lo_c} = \left\{ 2(W/S) \; / \; \rho \, C_{L_{Lo}} \right\}^{1/2} = (256.52)/\sqrt{C_{L_{Lo}}}$

i. If $\dfrac{V_{Lo} - V_{Lo_c}}{V_{Lo}} \leqslant 0.005$, the liftoff velocity is valid.

If not, iterate by selecting another V_{Lo} and repeat steps b. through i. until convergence is established.

2. Table C-1 depicts the above process for the STOL VIKING case.

3. Summary:

$V_{Lo} = 73.\text{kts}$

$C_{L_{Lo}} = 4.31$

$C_{\mu} = 2.08$

C. Calculation of Balanced Field Length(FLTO).

1. From the summary of APPENDIX B:

$W/S = 78.2 \text{ lb/ft}^2$

$T/W = 0.59$

$W = 46{,}738. \text{ lbs}$

$S = 598. \text{ ft}^2$

Also: $\rho = 0.0023769 \text{ lb-s}^2/\text{ft}^4$, TOOBHT = 50. ft, and

$k_t = \left[0.75(\ 5.00 + 6.23\ )\ /\ (\ 4.00 + 6.23\ )\right] = 0.8233$

2. From Reference 5-8, the expression for Balanced Field Length is:

$\text{FLTO} = (A \cdot B \cdot C + D)$, where

a. $A = 2.10$

b. $B = \dfrac{0.01163\ (\ W/S\ )}{\rho\, C_{L_{Lo}}} + 0.374\ (\ \text{TOOBHT}\ )$

$= (\ 88.78\ ) + (\ 18.78\ ) = 107.48$

c. $C = \dfrac{1.00}{k_t(\ T/W\ ) - 0.04} + 2.70 = (\ 2.243\ ) + (\ 2.70\ )$

$= 4.943$

d. $D = g / p^{1/2} = 656.36$

Then: FLTO = (1115.64) + (656.36) = 1772. ft

NOTE: $T_F = k_\perp(\ 27{,}364.\ )$ lbs, $\delta_F = 30^\circ$, $\delta_c = 0^\circ$

V_{LO} (kts)	V_S (kts)	V_S (fps)	q_{LO} (psf)	$k_\perp$	$T_{F_{LO}}$ (lbs)	C_μ	$C_{L_{MAX}}$	$C_{L_{LO}}$	V_{LO_c} (kts)	ΔV_{LO} (kts)	%Δ
65.00	54.17	91.43	14.30	0.813	22,247.	2.60	6.50	4.51	71.48	6.48	+9.97
68.00	56.67	95.65	15.66	0.815	22,302.	2.38	6.40	4.44	72.03	4.03	+5.94
70.00	58.33	98.47	16.59	0.816	22,329.	2.25	6.32	4.39	72.49	2.49	+3.56
72.00	60.00	101.28	17.55	0.817	22,357.	2.13	6.25	4.34	72.89	0.89	+1.24
73.00	60.83	102.69	18.04	0.818	22,384.	2.08	6.21	4.31	73.13	0.13	+0.02

Table C-1

APPENDIX - D

CALCULATION OF GROUND RUN DISTANCE

A. Ground run distance (S_G): see Figure 3-1.

Work done = ΔK.E.

1. $(\text{thrust})(S_G) = \frac{1}{2}m(V_{LO}^2 - V_O^2) = \frac{1}{2}(W/g)\, V_{LO}^2$

2. At liftoff: $\text{LIFT} = \text{WEIGHT} = C_{L_{LO}}(\frac{1}{2} \rho V_{LO}^2)\, S$

Then: $S_G = (W/S)(W/T)(1/C_{L_{LO}})(1/g \rho)$

$= (78.2)/(0.59)(4.31)(32.2)(.0023769)$

$= 402.$ ft

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