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PRELIMINARY DESIGN OF A SUPERSONIC SHORT TAKEOFF AND VERTICAL LANDING (STOVL) FIGHTER AIRCRAFT

PREPARED FOR: NASA USRA ADVANCED DESIGN PROGRAM

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ABSTRACT

The preliminary design study of a supersonic Short Takeoff and Vertical Landing (STOVL) fighter is presented. The study started with a brief historical survey of powered lift vehicles followed by a technology assessment of the latest supersonic STOVL engine cycles under consideration by industry and government in the US and UK. A survey of operational fighter/attack aircraft and the modern battlefield scenario were completed to develop, respectively, the performance requirements and mission profiles for the study. Three configurations were initially investigated with the following engine cycles: a hybrid fan vectored thrust cycle, a lift+lift/cruise cycle, and a mixed flow vectored thrust cycle.

The lift+lift/cruise aircraft configuration was selected for detailed design work which consisted of: 1) a material selection and structural layout, including engine removal considerations, 2) an aircraft systems layout, 3) a weapons integration model showing the internal weapons bay mechanism, 4) inlet and nozzle integration, 5) an aircraft suckdown prediction, 6) an aircraft stability and control analysis, including a takeoff, hover, and transition control analysis, 7) a performance and mission capability study, and 8) a life cycle cost analysis.

A supersonic fighter aircraft with Short Takeoff and Vertical Landing (STOVL) capability with the lift+lift/cruise engine cycle seems a viable option for the next generation fighter.

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LIST OF SYMBOLS

SYMBOLS

Symbol	Definition	Units
A	Aspect ratio	--
$\overline{A}$	Root mean squared gust-induced g-level	g/ft/sec
Al	Aluminum	--
b	Wing span	ft.
B.L.	Buttock line	--
c	Chord	ft.
C_L	Airplane lift coefficient	--
C_D	Airplane drag coefficient	--
C_M	Airplane pitching moment	--
C_D_o	Parasite drag	--
C_D_u	Variation of drag coefficient with speed	--
C_D_α	Variation of drag coefficient with angle of attack	1/rad
C_L_u	Variation of lift coefficient with speed	--
C_L_α	Airplane lift curve slope	1/rad
C_L_α dot	Variation of lift coefficient with rate of change of angle of attack	--
C_L_q	Variation of lift coefficient with pitch rate	--
C_M_u	Variation of pitching moment coefficient with speed	--
C_M_α	Variation of pitching moment coefficient with angle of attack	1/rad

C_M_α dot	Variation of pitching moment coefficient with rate of change of angle of attack	--
C_M_q	Variation of pitching moment coefficient with pitch rate	--
C_y_ß	Variation of side force coefficient with sideslip angle	1/rad
C_y_r	Variation of side force coefficient with yaw rate	--
C_y_p	Variation of side force coefficient with roll rate	--
C_y_δR	Variation of side force coefficient with rudder angle	1/rad
C_n_ß	Variation of yawing moment coefficient with sideslip angle	1/rad
C_n_r	Variation of yawing moment coefficient with yaw rate	--
C_n_p	Variation of yawing moment coefficient with roll rate	--
C_n_δR	Variation of yawing moment coefficient with rudder angle	1/rad
C_n_δA	Variation of yawing moment coefficient with aileron angle	1/rad
C_l_ß	Variation of rolling moment coefficient with sideslip angle	1/rad
C_l_r	Variation of rolling moment coefficient with yaw rate	--
C_l_p	Variation of rolling moment coefficient with roll rate	--
C_l_δR	Variation of rolling moment coefficient with rudder angle	1/rad
C_l_δA	Variation of rolling moment coefficient with rudder angle	1/rad

Cu	Copper	--
D	Drag	lbs
e	Oswald's efficiency factor	--
F.S.	Fuselage station	--
H(subscript)	Horizontal tail	--
I_xx	Moment of inertia about x-axis	slug-ft^2
I_yy	Moment of inertia about y-axis	slug-ft^2
I_zz	Moment of inertia about z-axis	slug-ft^2
I_xz	Product of inertia	slug-ft2
i_H	Stabilator incidence	deg
K	Correction factor	--
L	Rolling moment	ft-lbs
l	Perturbed rolling moment	ft-lbs
L_ß	Dimensional variation of rolling moment about X_s with sideslip angle	1/sec^2
L_p	Dimensional variation of rolling moment about X_s with roll rate	1/sec
L_r	Dimensional variation of rolling moment about X_s with yaw rate	1/sec
L_δA	Dimensional variation of rolling moment about X_s with aileron angle	1/sec^2
L_δR	Dimensional variation of rolling moment about X_s with rudder angle	1/sec^2
M	Mach number	--
M	Pitching moment	--
m	Perturbed pitching moment	--
Mk	Mark	--

M_u	Dimensional variation of pitching moment with speed	1/ft-sec
M_{T_u}	Dimensional variation of pitching moment with speed	1/ft-sec
M_α	Dimensional variation of pitching moment with angle of attack	$1/sec^2$
M_{T_α}	Dimensional variation of pitching moment with angle of attack	$1/sec^2$
$M_{\dot{\alpha}}$	Dimensional variation of pitching moment with rate of change of angle of attack	1/sec
M_q	Dimensional variation of pitching moment with pitch rate	1/sec
$M_{\delta E}$	Dimensional variation of pitching moment with elevator angle	$1/sec^2$
Mo	Molybdenum	--
N	Yawing moment	ft-lbs
n	Perturbed yawing moment	ft-lbs
nm	Nautical miles	--
N_β	Dimensional variation of yawing moment about Z_s with sideslip angle	$1/sec^2$
N_{T_β}	Dimensional variation of yawing moment about Z_s with sideslip angle	$1/sec^2$
N_p	Dimensional variation of yawing moment about Z_s with roll rate	1/sec
N_r	Dimensional variation of yawing moment about Z_s with yaw rate	1/sec
$N_{\delta A}$	Dimensional variation of yawing moment about Z_s with aileron angle	$1/sec^2$
$N_{\delta R}$	Dimensional variation of yawing moment about Z_s with rudder deflection	$1/sec^2$
P	Roll rate	rad/sec

p	Perturbed roll rate	rad/sec
Q	Pitch rate	rad/sec
q	Perturbed pitch rate	rad/sec
q_bar	Dynamic pressure	lb/sq.ft.
R	Yaw rate	rad/sec
r	Perturbed yaw rate	rad/sec
S	Planform area	sq. ft.
Ti	Titanium	--
Treq	Thrust required	lbs
U	Forward velocity	ft/sec
u	Perturbed forward velocity	ft/sec
V	Side velocity	ft/sec
v	Perturbed side velocity	ft/sec
V	Vanadium	--
V(subscript)	Vertical tail	--
W	Downward velocity	ft/sec
w	Perturbed downward velocity	ft/sec
W.L.	Water line	--
X_u	Dimensional variation of X_s force with speed	1/sec
X_T_u	Dimensional variation of X_s force with speed	1/sec
X_α	Dimensional variation of X_s force with angle of attack	ft/sec^2
X_δE	Dimensional variation of X_s force with elevator angle	ft/sec^2

Y_ß	Dimensional variation of Y_s force with sideslip angle	ft/sec^2
Y_p	Dimensional variation of Y_s force with roll rate	ft/sec
Y_r	Dimensional variation of Y_s force with yaw rate	ft/sec
Y_δA	Dimensional variation of Y_s force with aileron angle	ft/sec^2
Y_δR	Dimensional variation of Y_s force with rudder angle	ft/sec^2
Z_u	Dimensional variation of Z_s force with speed	1/sec
Z_α	Dimensional variation of Z_s force with angle of attack	ft/sec^2
Z_α dot	Dimensional variation of Z_s force with rate of change of angle of attack	ft/sec
Z_q	Dimensional variation of Z_s force with pitch rate	ft/sec
Z_δE	Dimensional variation of Z_s force with elevator angle	ft/sec^2

GREEK SYMBOLS

α	Angle of attack	deg
ß	Sideslip angle	deg
δ	Deflection angle	deg
ω	Natural Frequency	rad/sec
ξ	Damping Ratio	--

ACRONYMS

AE	Aerospace Engineering	--
AEP	Average Estimated Price	US dollars
AFCS	Automatic Flight Control System	--
AGARD	Advisory Group for Advanced Research and Development for NATO	--
AGM	Air-to-Ground Missile	--
AIM	Air Intercept Missile	--
AIV	Annular Invertor Valve	--
AMRAAM	Advanced Medium Range Air-to-Air Missile	--
APU	Auxiliary Power Unit	--
ASRAAM	Advanced Short Range Air-to-Air Missile	--
AWACS	Airborne Warning And Control System	--
BAe	British Aerospace	--
BAI	Battlefield Air Interdiction	--
BCM/A	Best Cruise Mach and Altitude	--
BIT	Built In Test	--
BVR	Beyond Visual Range	--
CA	Counter Air	--
CAD	Computer Aided Design	--
ECM	Electronic Counter Measures	--
ECS	Environmental Control System	--
EFA	European Fighter Aircraft	--
FLIR	Forward Looking Infrared	--
FOD	Foreign Object Damage	--

HARM	High Speed Anti-Radiation Missile	--
HFVT	Hybrid Fan Vectored Thrust	--
HGR	Hot Gas Reingestion	--
HOTAS	Hands On Throttle And Stick	--
HUD	Head Up Display	--
IFF	Identification Friend or Foe	--
IIR	Imaging Infrared	--
ILS	Instrument Landing System	--
IR	Infrared	--
ITR	Inlet Temperature Rise	--
JFS	Jet Fuel Starter	--
KEAS	Knots Equivalent Airspeed	knots
KU	Kansas University	--
LCC	Life Cycle Cost	US dollars
LIFT	Lift + Lift/Cruise Engine Cycle	--
LTE	Launch-To-Eject Cycle	--
MFVT	Mixed Flow Vectored Thrust	--
MRM	Medium Range Missile	--
NACA	National Advisory Committee on Aeronautics	--
NASA	National Aeronautics and Space Administration	--
NATO	North Atlantic Treaty Organization	--
NBC	Nuclear, Biological, Chemical gear	--
OBOGS	On-Board Oxygen Generating System	--
RCS	Reaction Control System	--
RCV	Reaction Control Valve	--

RDTE	Research, Development, Test and Evaluation	--
RF	Radio Frequency (Radar)	--
RWR	Radar Warning Receiver	--
S & A	Safe and Arm	--
SAS	Stability Augmentation System	--
SLS	Sea-Level Standard	--
SRM	Short Range Missile	--
STOL	Short Take Off and Landing	--
STOVL	Short Take Off and Vertical Landing	--
TACAN	Tactical Air Navigation	--
TQM	Total Quality Management	--
UHF	Ultra High Frequency	--
USD	United States Dollars	--
USRA	Universities Space Research Association	--
VHF	Very High Frequency	--
VTOL	Vertical Take Off and Landing	--
V/STOL	Vertical/Short Take Off and Landing	--

1. INTRODUCTION

The survivability of long, hard surface runways at Air Force Main Operating Bases is fundamental to the current operations of the Air Force Tactical Air Command. Without the use of these runways, the effectiveness of the Tactical Air Command is severely degraded. One possible solution to this runway denial situation is to include a Short Takeoff and Vertical Landing (STOVL) capability in a supersonic fighter/attack vehicle. Design teams at the University of Kansas, through the sponsorship of the NASA/USRA program, have completed a conceptual design study of three supersonic STOVL aircraft and based on this study, selected one aircraft for detailed design work.

The cooperation between the NASA/USRA Advanced Design Program and the design efforts at the University of Kansas are discussed in Section 1.1. Section 1.2 presents the study plan and objectives of the design study.

1.1 BACKGROUND

The NASA/USRA Advanced Design Program is, from Reference 1.1, "a unique national program that brings together NASA engineers with students and faculty from United States engineering schools by integrating current and future NASA space/aeronautics curriculum." The University of Kansas is one of approximately forty five universities selected for this program.

The USRA Advanced Design Program course is taught in addition to the existing design courses at the University of Kansas. Table 1.1 shows how the USRA and KU design courses are offered to the students. Each design course is worth 4 hours of engineering design. All students are required to take AE 521 to learn the basic methods of design. The student is free to choose among the remaining five design courses to fulfill the eight hours of design required for a degree in Aerospace.

Although the USRA design courses are offered as a graduate level course, most students are undergraduates that wish to have more than the required amount of engineering design hours. Section 1.2 discusses in more detail the USRA design courses for the 1989-90 academic year.

1.2 STUDY PLAN

The supersonic STOVL started in the fall semester (Phase I) with a brief historical survey of powered lift vehicles followed by a technology assessment of the latest supersonic STOVL engine cycles under consideration by industry and government in the US and UK. A survey of operational fighter/attack aircraft and the modern battlefield scenario were completed to develop, respectively, the performance requirements and mission profiles for the study. Three aircraft were selected for initial investigations. The following engine cycles were used: a hybrid fan vectored thrust cycle, a lift+lift/cruise cycle, and a mixed flow vectored thrust cycle. Chapter 2 shows the results of the Phase I aircraft study. Chapter 3 presents the Phase I aircraft comparison and selection of the aircraft for the second semester (Phase II), in which the lift+lift/cruise aircraft was selected detailed design work.

Table 1.1 USRA and KU Design Courses

	Fall Semester	Spring Semester
KU	AE 521: Aircraft Design * Preliminary Analysis * Individual Work	AE 522: Aircraft Design * Detailed Analysis * Team/Individual Work * National Competition
		AE 523: Engine Design * Detailed Analysis * Team Work * National Competition
		AE 524: Space Design * Detailed Analysis * Team Work
USRA	AE 621: Aircraft Design * Preliminary Analysis * Team Work	AE 622: Aircraft Design * Detailed Analysis * Team Work

Chapter 3 also discusses the design changes of this aircraft. Chapter 4 gives the aircraft description. The weight and balance is presented in Chapter 5 and the propulsion system integration is shown in Chapter 6. A takeoff, hover, and transition analysis is given in Chapter 7. The performance and mission capability of the aircraft is presented in Chapter 8. Chapter 9 presents the stability and control of the aircraft. Chapter 10 presents the material selection and structural layout of the aircraft and discusses accessibility and maintainability considerations, including the engine removal. The aircraft systems layout is given in Chapter 11. The weapons integration is shown in Chapter 12. The life cycle cost is shown in Chapter 13. Chapter 14 gives conclusions and recommendations for the study.

REFERENCES FOR CHAPTER 1

1.1 NASA/USRA University Advanced Design Program, Program Handbook for Faculty Teaching Assistants & Students, 1989-90 Academic Year.

2. PHASE I AIRCRAFT DESCRIPTIONS

The purpose of this chapter is to present the results of the Phase I aircraft study. For each aircraft the following is given: a description of the configuration, a three view with geometric data, an inboard profile, an area rule of the configuration, and a weight summary. Complete documentation of the Phase I aircraft are in References 2.1-2.3. The mission profile and specifications for the Phase I study are given in Section 2.1. Section 2.2 presents the Lift + Lift/Cruise configuration (the Monarch), Section 2.2 presents the hybrid fan vectored thrust configuration (the Viper), and Section 2.3 presents the mixed flow vectored thrust configuration (the Nemesis).

2.1 PHASE I AIRCRAFT MISSION PROFILE AND SPECIFICATIONS

Reference 2.4 states the responsibilities of a fighter/attack aircraft in the European theater as a balance between counter air and close air support/battlefield air interdiction. The design team chose to study a fighter aircraft having a primary mission as counter air and a secondary mission as battlefield air interdiction. The intent is to have the counter air mission size the aircraft with the battlefield air interdiction mission as a fallout.

The selected profile and specifications of the counter air mission and the battlefield air interdiction mission were developed using References 2.5 and 2.6. Reference 2.5 contributed the following information: the battlefield scenario for a STOVL fighter, the threats to a STOVL fighter (land based anti-aircraft and aircraft threats), research of similar aircraft mission profiles, and the stores and ammunition selection. Reference 2.6 contributed a mission capability trade study. This study investigated the sensitivity of aircraft weight to mission range and Mach number. Figures 2.1 and 2.2, respectively, show the counter air (CA) mission and battlefield air interdiction (BAI) mission profiles. Three BAI missions were selected for the study. The mission profile is the same for each, but the ordnance carried varies. The specifications for the missions are given in Table 2.1.

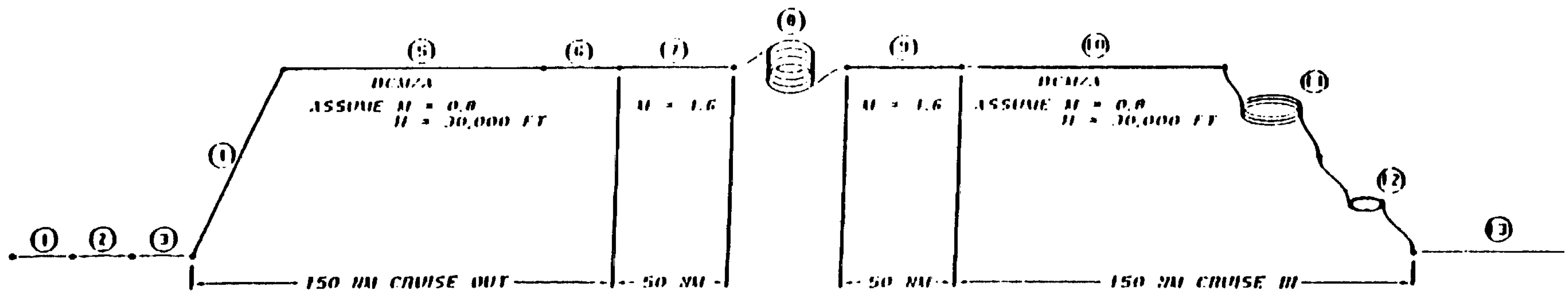
The point performance data was selected with the suggestion of Reference 2.7 that reasonable performance for a STOVL fighter can be selected by slightly bettering the performance of the Northrop F-20 (Reference 2.8). The selected performance is also shown in Table 2.1.

2.2 LIFT + LIFT/CRUISE AIRCRAFT DESCRIPTION (MONARCH)

The overall configuration of the aircraft consists of a conventional wing and fuselage with a canard and strake. The crew consists of one pilot. Payload requirements are given in the mission specification. The Monarch aircraft employs an unconventional internal mounting system for the counter air mission weapons. The engine cycle consists of one dedicated lift engine in the forebody of the aircraft and a lift/cruise engine in the aft end of the fuselage. The landing gear is of the tricycle type. A three view of the Monarch aircraft, including its geometric parameters, is shown in Figure 2.3. The inboard profile of this aircraft is shown in Figure 2.4.

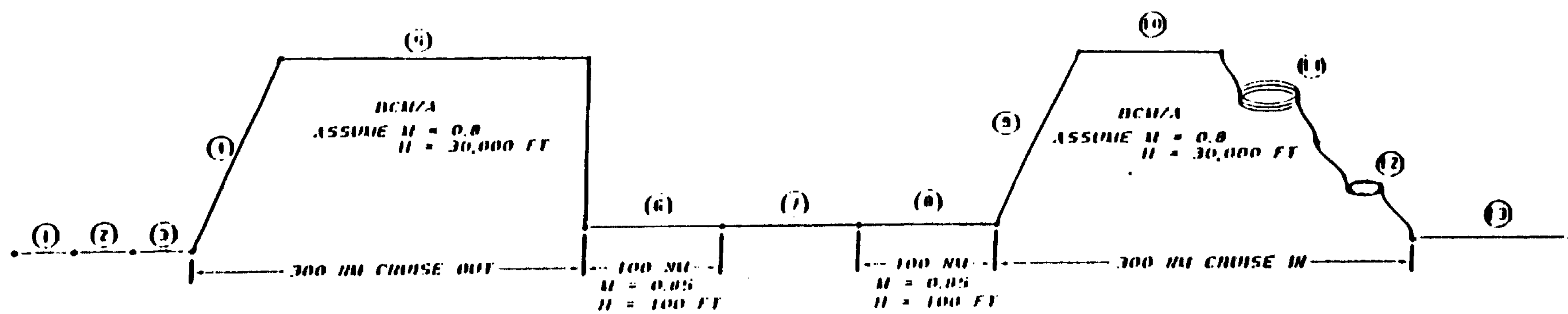
Major design considerations for this aircraft include:

* volume requirements for internal weapons,



1. ENGINE START/WARM-UP
2. TAXI
3. STO
4. CLIMB
5. BCM/A CRUISE OUT
6. ACCELERATE TO SUPERSONIC CRUISE MACH NUMBER, M = 1.6
7. SUPERSONIC CRUISE OUT - M = 1.6, H = 30,000 FT
8. COMBAT (EXPEND HALF STORES) H = 30,000 FT
9. SUPERSONIC CRUISE IN - M = 1.6, H = 30,000 FT
10. BCM/A CRUISE IN
11. 10 MINUTE LOITER - 1,000 FT
12. ONE MINUTE HOVER
13. LAND WITH 5% INTERNAL FUEL

Figure 2.1 Counter Air Superiority Mission Profile



1. ENGINE START/WARM-UP
2. TAXI
3. STO
4. CLIMB TO CRUISE OUT ALTITUDE
5. BCM/A CRUISE OUT
6. DASH OUT
7. WEAPONS DROP HALF LOAD
8. DASH IN
9. CLIMB TO CRUISE IN ALTITUDE
10. BCM/A CRUISE IN
11. 10 MINUTE LOITER - 1,000 FT
12. ONE MINUTE HOVER
13. LAND WITH 5% INTERNAL FUEL

Figure 2.2 Battlefield Air Interdiction Mission Profile

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Table 2.1 Phase I Aircraft Mission Specification

CREW: One Pilot, (225 lbs)

ARMAMENT: One internal M61A1 Vulcan cannon, and 400 rounds of 20mm ammo

PAYLOAD:

Counter Air
Two ASRAAM's (stored internally), and
Two AMRAAM's (stored internally)

Battlefield Air Interdiction
Six Mk 82 Bombs (externally stored), or
Four AGM-65 Mavericks (externally stored), or
Six AGM-88A HARMs (externally stored)

PERFORMANCE:

Performance Characteristic	Value
Time to Climb	40k in 2 minutes
1g Specific Excess Energy	
(2A) 30k 0.9M	500 ft/sec
(2B) 10K 0.9M	1,000 ft/sec
Sustained Turn Rate	
(3A) 0.8M/15k ft	15 deg/sec
(3B) 0.9M/30k ft	9 deg/sec
(3C) 1.2M/30k ft	8 deg/sec
(3D) 0.9M/15k ft	6.5 g
(3E) 1.6M/30k ft	4.5 g
Acceleration	
(4A) 30k ft 0.9M to 1.6M	70 sec
(4B) 0.5M to 1.4M	80 sec
(4C) 10k ft 0.3M to 0.9M	22 sec
Landing Distance	
Without Chute	2,200 ft

GROUNDRUN: Takeoff - 300 ft, Vertical Landing

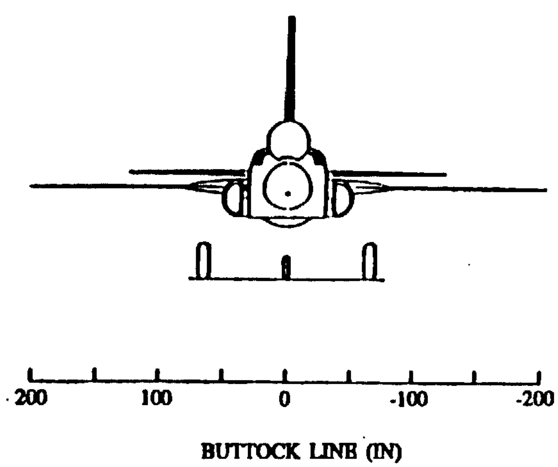
CERTIFICATION: Military

RANGE AND ALTITUDE: See mission profile

	Wing	Canard	Vertical Tail
Area	322 sq. ft	50 sq. ft.	42.9 sq. ft.
Span	33.5 ft.	7.9 ft.	8.4 ft.
Aspect Ratio	6	2.5	1.65
Sweep Angle	37.8 deg	38 deg	40.6 deg
Taper Ratio	0.19	0.30	0.35
Thickness Ratio			
Root	8%	6%	6%
Tip	6%	6%	6%
Airfoil			
Root	NACA 64-208	NACA 64-206	NACA 64-206
Tip	NACA 64-206	NACA 64-206	NACA 64-206
Dihedral Angle	0 deg	0 deg	0 deg
Incidence Angle	0 deg	0 deg	0 deg

Flaperon Chord Ratio 0.25   Flaperon Span Ratio 0.70
Rudder Chord Ratio 0.35

	Fuselage	Overall
Length	58.33 ft.	58.33 ft.
Max Width	5.00 ft.	33.50 ft.
Max Height	5.17 ft.	13.50 ft.



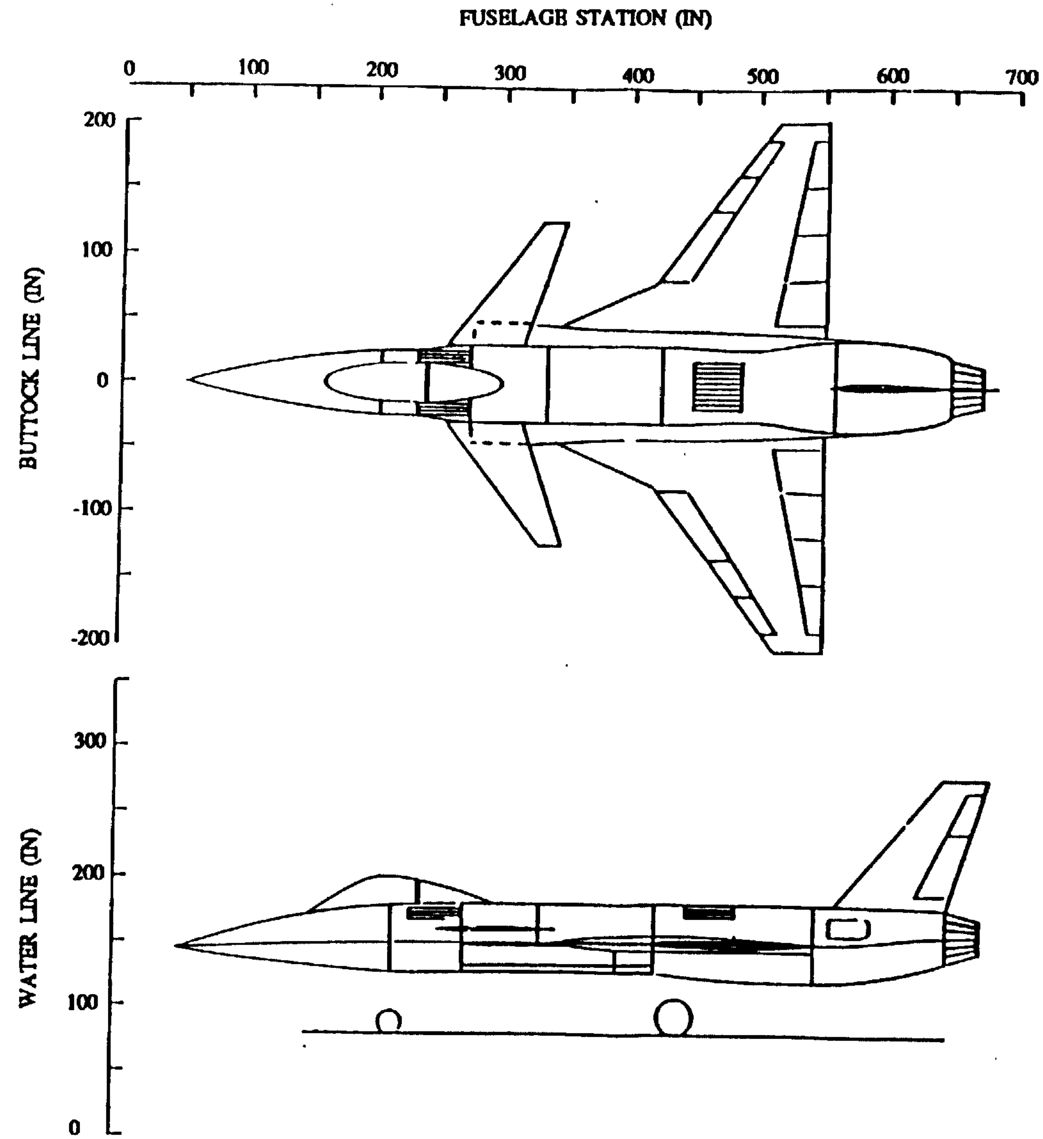


Figure 2.3 Lift + Lift/Cruise Aircraft Three View

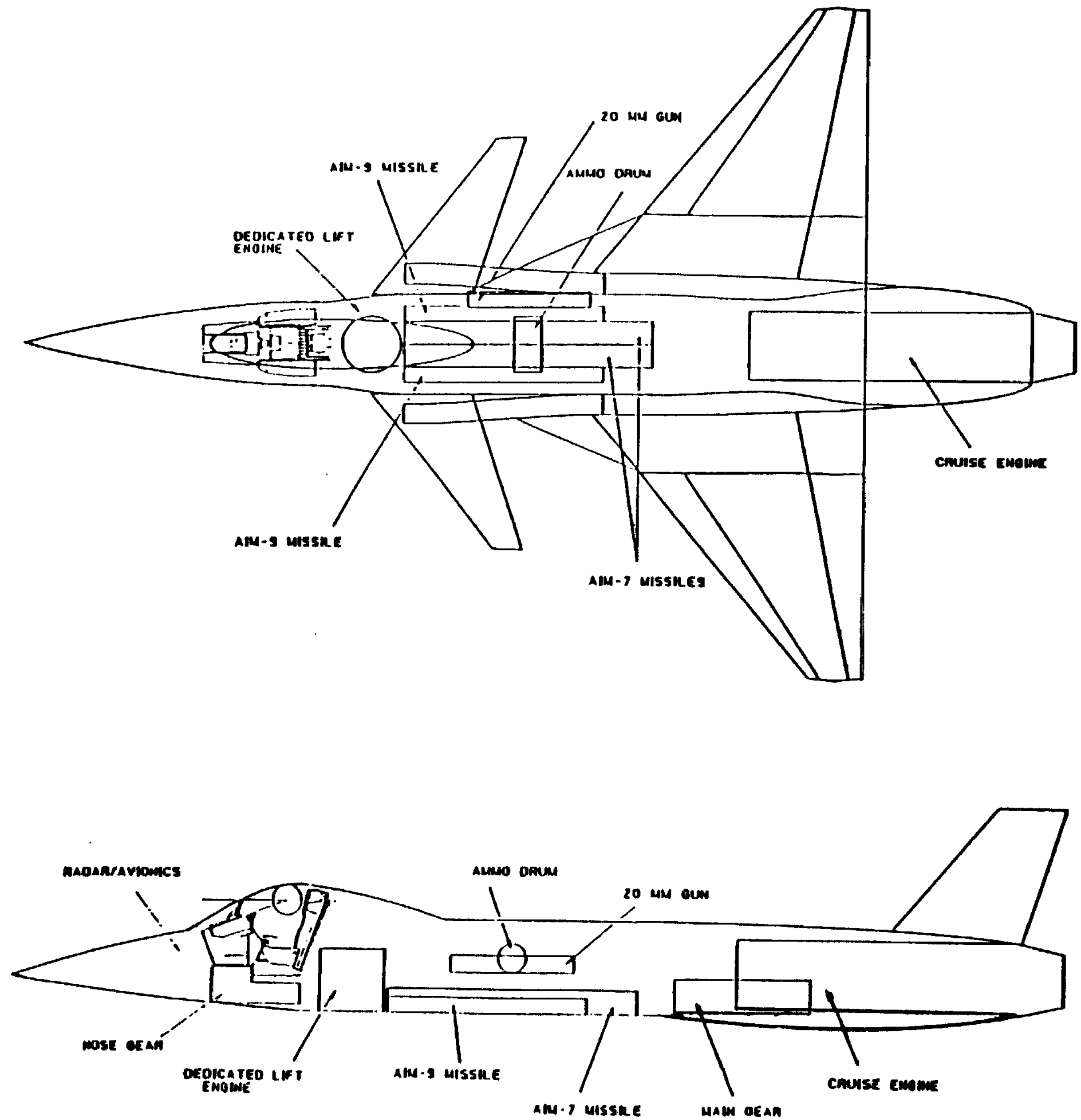


Figure 2.4 Lift + Lift/Cruise Aircraft Internal Layout

* engine sizing and balance considerations for hover flight,
* and supersonic flight requirements.

The Monarch aircraft has a cantilever wing configuration to provide primary lift for the aircraft in cruise flight. Full span leading and trailing edge surfaces provide high lift and roll control. The wing is mounted mid-fuselage with a 37.8 degree leading edge sweep. The addition of a strake to the aircraft provides delayed wing stall at high angles of attack, additional fuel volume and structural support for weapon hard points. The airfoils for the wing are 8% thick at the root and 6% at the tip. The empennage of the aircraft consists of a conventional vertical tail and forward mounted canard. The vertical tail is a single fin and houses a rudder to provide directional control. This rudder consists of two individual pieces with separate actuators. This was done to provide redundancy against battle damage.

Primary design considerations for the fuselage layout include the requirements for internal weapons and shaping to reduce wave drag. Unconventional sizing was required to create internal volume for the counter air mission weapons and the dedicated lift engine. This lead to the lower fuselage being flat for most of the aircraft length. "Coke-bottling" was incorporated at the wing fuselage interface in an effort to improve the area ruling.

The results of area ruling (Mach = 1) for the Monarch aircraft are presented in Figure 2.5. Area ruling is a method used for shaping a fuselage to minimize wave drag in transonic and supersonic flight. From Figure 2.5 it can be seen that the Monarch aircraft slightly exceeds the ideal (Mach = 1) area rule model. Removing the fuselage coke bottling in the vicinity of the wing may provide a more favorable area distribution. However, such a design change would add wetted area. This may result in a net drag increase which would negate the area rule improvement.

The engine cycle specified for the Monarch aircraft consisted of a Lift + Lift/Cruise system: a dedicated lift engine for hover and transition and a lift/cruise engine used for hover, transition, and cruise. Design considerations for sizing the engines included hover, supersonic flight, and transition from hover to horizontal flight.

The landing gear chosen for this aircraft is a retractable tricycle type. It consists of a nose gear and two main struts aft. The main gear retracts aft into a fuselage fairing along the lift/cruise engine. The nose gear retracts forward to a position underneath the cockpit.

The cockpit of the Monarch is sized for one pilot. An ejection seat, heads-up display and center control stick make the cockpit conventional in design for a small or medium sized fighter aircraft. The view from the cockpit was an important design consideration in the Monarch. Lack of visibility is detrimental to aerial combat effectiveness where the first sighting is very important. The pilot of the Monarch fighter will have a view of 14.5 degrees over the nose of the aircraft and 5 degrees over the tail. View over the side of the cockpit is 52 degrees.

Table 2.2 gives the weight summary for the Monarch aircraft.

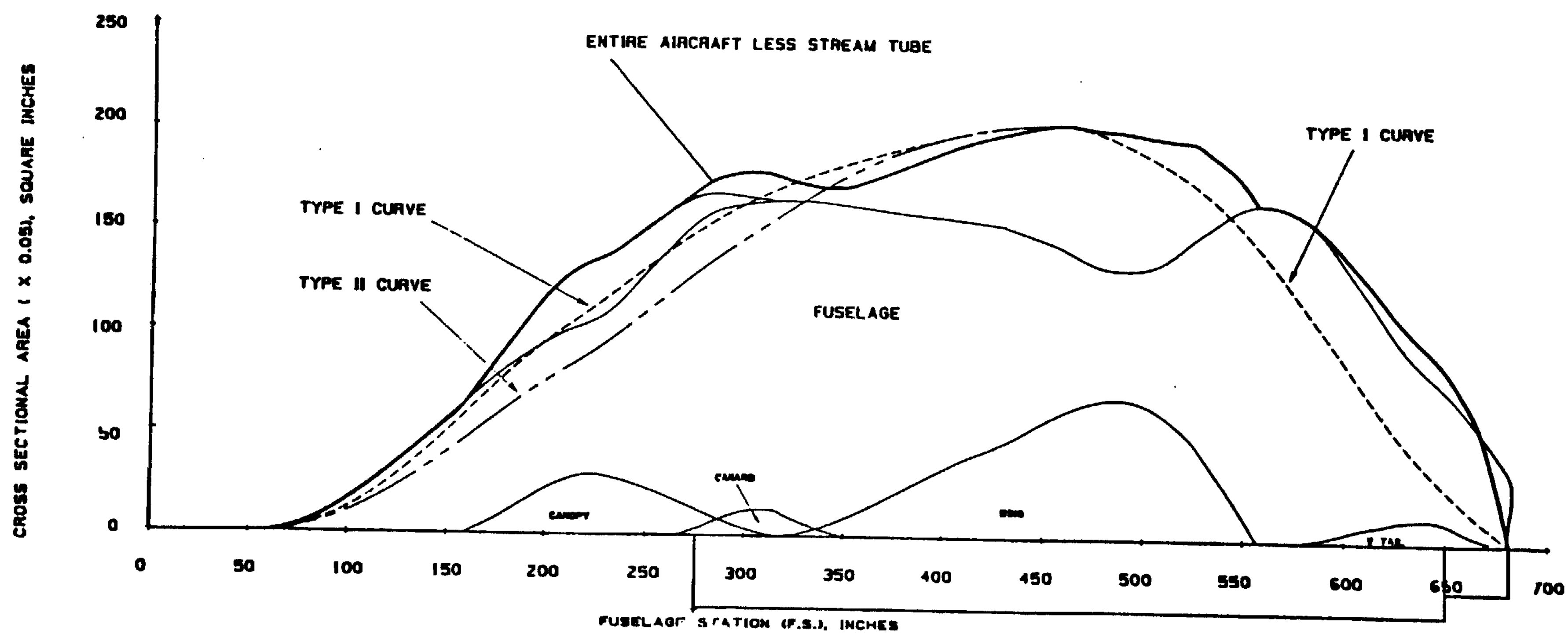


Figure 2.5 Lift + Lift/Cruise Aircraft Area Rule Plot

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Table 2.2 Lift + Lift/Cruise Aircraft Weight Summary
(See Table 2.1 for CA, BAI#1, BAI#2, and BAI#3 Ordnance)

	MISSION			
	CA	BAI#1	BAI#2	BAI#3
STRUCTURE	(7349)			
Fuselage	4043			
Wing	1579			
Tails - Vertical	333			
- Canard	298			
Landing Gear - Main	931			
- Nose	165			
PROPULSION	(6235)			
Cruise Engine	4009			
Lift Engine	647			
Air Induction	876			
Fuel Bladder	474			
Fuel Dumping	26			
Engine Controls	43			
Starting System	139			
Water Injection	21			
FIXED EQUIPMENT	(4484)			
Flight Control	999			
Avionics	1164			
Electrical System	548			
Air Conditioning	254			
Oxygen System	17			
APU	257			
Furnishings	276			
Gun and Provisions	630			
Auxilary Gear, Paint	341			
STOVL EQUIPMENT	(690)			
Ventral Nozzle	300			
RCS Equipment	390			
TOTAL EMPTY WEIGHT	18758	18758	18758	18758
Crew	225	225	225	225
Total Fuel	10308	10308	10308	10308
Armament	(1290)	(3820)	(3820)	(4300)
ASRAAMS	400			
AMRAAMS	670			
HARM		3600		
Mk-82's			3600	
Maverick's				4080
Ammo - 200 rnds	220	220	220	220
TAKEOFF WEIGHT	30581	33111	33111	33591

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2.3 HYBRID FAN VECTORED THRUST AIRCRAFT DESCRIPTION (VIPER)

A three view of the Viper aircraft, including geometric parameters, is given in Figure 2.6 and the internal layout is shown in Figure 2.7. There are five major configuration related aspects that drive the Viper design. These are:

*) the forward swept wing,
*) the empennage and tail configuration,
*) the armament location,
*) the fuel volume,
*) and the powerplant and engine/airframe integration.

The overall aspects of the Viper design are discussed below.

The Viper is equipped with triple redundant, fly-by-wire flight control system. The vectoring of the exhaust nozzles, forward and aft, are also computer controlled for stability and to maximize performance in transition and hover.

Forward swept wings in supersonic fighter configurations offer some advantages when compared to conventional planforms. An important consideration is the improved pilot visibility over the sides of the aircraft. This aspect is particularly important during vertical operations as well as during combat. A forward swept wing may also produce a smoother Sears-Haack area distribution, giving better wave drag characteristics in the supersonic regime. This is important for a STOVL design which should not compromise its capabilities while operating in a conventional mode during supersonic cruise.

A forward swept wing configuration allows for a great deal of flexibility in terms of structural synergism. One advantage is that the wing main spar frame is used to attach the engine to the rest of the airframe. Another advantage is that the front spar and the kick spar are attached to improve the structural integrity of the wing.

Another characteristic of forward swept wings that is attractive in fighter applications is that the wing root will stall before the tip, allowing for continued aileron control at high angles of attack. Furthermore, forward sweep allows for the wing center of gravity to be very close to the airplane center of gravity, decreasing the need for longitudinal trim as fuel and payload are expended.

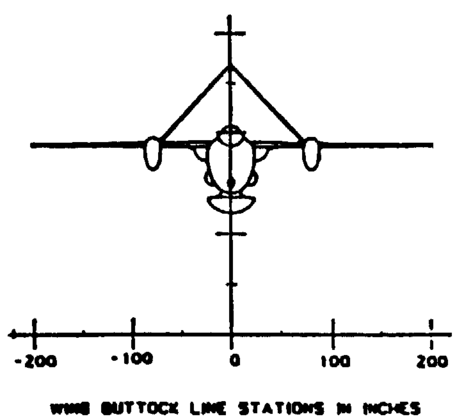
A drawback that needs to be considered in the forward swept wing concept is that it is prone to body freedom flutter. However, this can be solved through aeroelastic tailoring, such as is done in the X-29.

The use of twin booms, like the forward swept wing, also allows for a great deal of synergism. They provide wing bending moment relief and volume for fuel and weapon storage. According to Reference 2.9, twin booms can also tailor the configuration for low wave drag, while a certain degree of combat survivability and redundancy is added.

Some problems, however, are associated with twin boom designs. They are:

* In long boom configurations, critical loads on the tail lead to large boom cross sections.

	Wing	Horizontal (projected)	Vertical (projected)
Area (ft²)	321.9	62.0	62.0
Span (ft)	33.57	13.16	6.58
MGC eff. (ft)	11.92	4.917	4.917
Aspect ratio	3.5	2.79	0.70
LE sweep (deg)	-30	60	60
Taper ratio	0.207	0.53	0.53
Thickness ratio			
root	0.06	0.06	0.06
tip	0.06	0.06	0.06
Airfoil			
root	NACA 65-006	NACA 65-006	NACA 65-006
tip	NACA 65-006	NACA 65-006	NACA 65-006



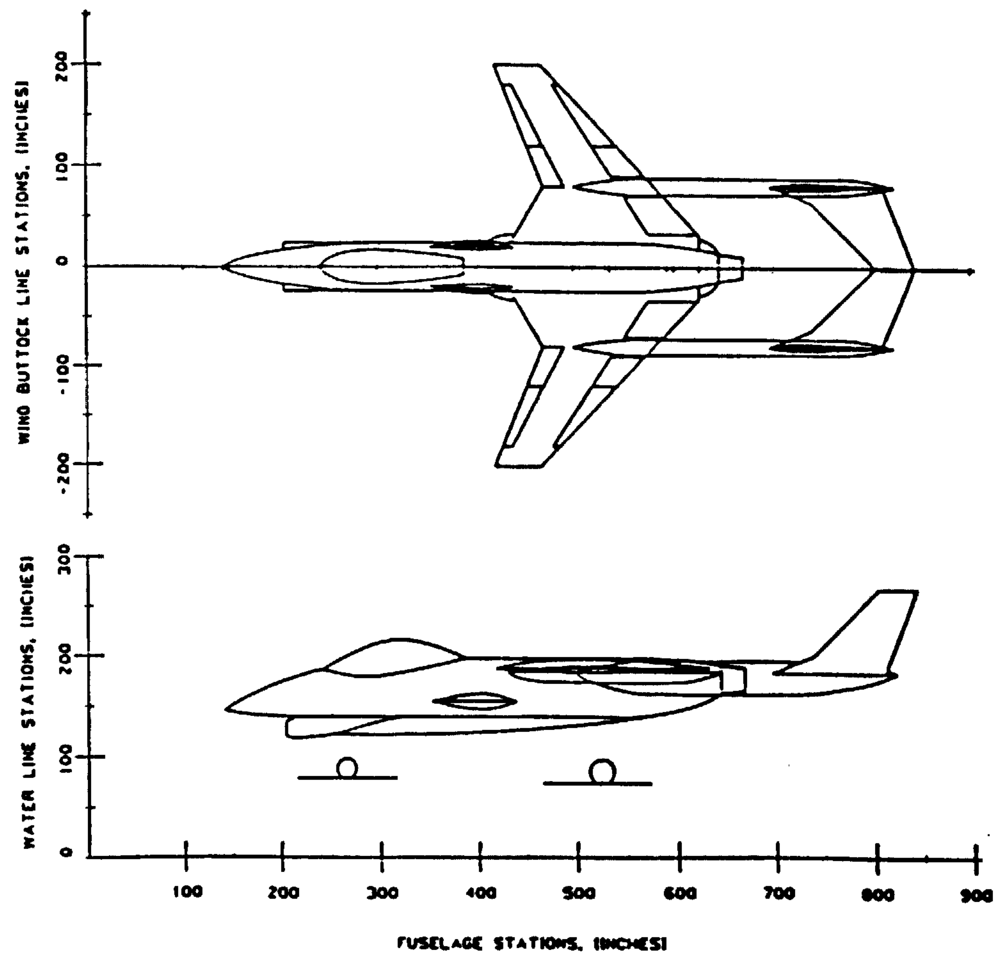


Figure 2.6 Hybrid Fan Vectored Thrust Aircraft Three View

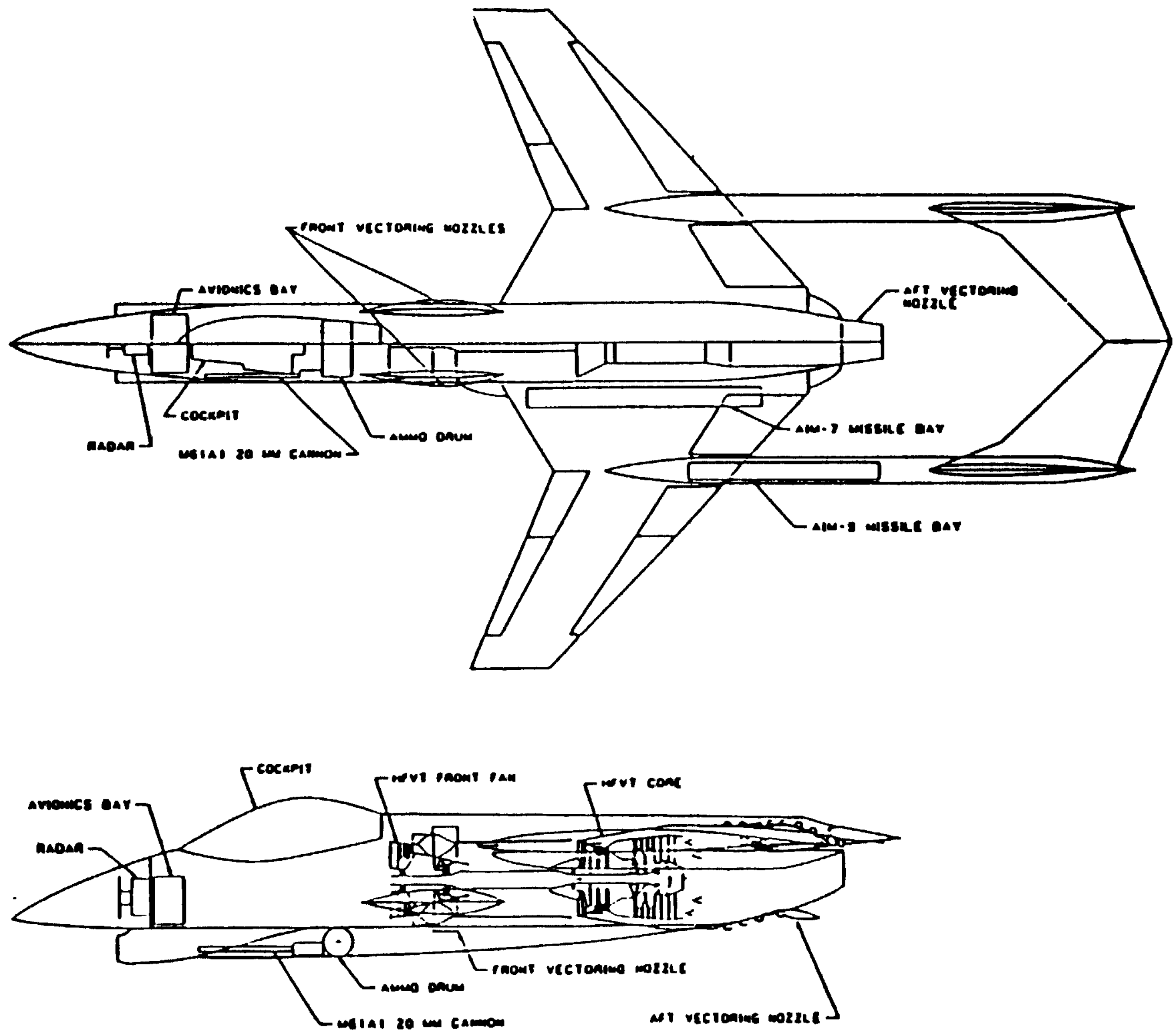


Figure 2.7 Hybrid Fan Vectored Thrust Aircraft Internal Layout

* Vibration and fatigue due to excessive noise of engine exhaust flow.
* Scrubbing drag from the engine exhaust impingement on the boom structure may be a problem.

The booms do, however, produce a shielding effect on the exhaust, reducing the infrared signature of the aircraft.

An aft swept inverted vertical tail is used for the Viper. This design aids in the stealth characteristics of the aircraft as well as act as a structural tie between the booms. The control surface of a V-tail must perform both of the jobs of a conventional elevator and rudder. Since a forward swept wing configuration lends itself to inherent longitudinal instability, a fly-by-wire system is needed.

For both counter air and battlefield air interdiction missions, a M61A1 20 mm cannon is used with 400 rounds of ammunition. The gun is located under the fuselage on the port side. The counter air weapons are carried internally. The AMRAAM's are located in fairings at the wing root/fuselage intersection. The ASRAAM's are stored inside the booms. The battlefield air interdiction weapons are carried externally. There are six hardpoint locations to provide for this weapon capability:

* two beneath the fuselage, between the nose and main landing gear doors,
* two beneath the boom, where the boom intersects the wing,
* and two beneath the wing, outboard of the boom intersection.

The Viper uses a hybrid fan vectored thrust (HFVT) engine. The hybrid fan vectored thrust engine comprises a mixed augmented turbofan driving a remote front fan through a shaft. The HFVT has a dry thrust split of 0.6. The front fan is connected to the rest of the engine by an interduct, at the forward end of which is a diverter valve. There are two operating modes:

1. Parallel -- The front fan flow is diverted to a plenum and fed to two unaugmented, fully vectoring front nozzles. The core air is fed by a ventral auxiliary inlet behind the cockpit. The rear nozzle is vectorable to 110 degrees.

2. Series -- The auxiliary inlet and front nozzles are shut off with an annular inverter valve (AIV) that performs the miracle of flow shifting. The front fan air passes through the valve to the rest of the engine. This provides for maximum engine boost.

The parallel mode is used in short take off, vertical landing and subsonic cruise. In short take off, the two front nozzles and the main rear nozzle are both vectored down and aft to create a lifting force and a forward velocity. All nozzles are vectored down during vertical landing. Using the parallel mode in subsonic cruise with the front nozzles vectored fully aft will allow for a higher bypass ratio. This may, consequently, improve the specific fuel consumption in subsonic cruise.

The series mode is used for high performance and supersonic flight. In this mode, the front nozzles will be faired in by a retractable ramp to minimize drag.

A chin inlet is implemented in the Viper design. This is done for several reasons:

* moving the inlet as far as possible from the exhaust nozzles will reduce the hot gas re-ingestion (HGR) and foreign object damage (FOD),
* this position allows for good pressure recovery,
* and pilot vision is not affected.

The forward vectorable nozzles are located on the sides of the aircraft, just forward of the wing. This will allow for some lateral control by differential vectoring of these two front nozzles. However, a reaction control system (RCS) will still be required for complete control.

Although the problem has not yet been thoroughly investigated, it will be assumed at this point that engine removal will be accomplished by removing it out of the back of the aircraft. The structural arrangement of the Viper has not yet been determined, but engine removal will be a major concern. The very large front fan dimension may not allow for removal through the tail. As mentioned previously, the engine will be mounted to the wing main spar frame for structural synergism.

The largest contributor to drag in a supersonic flight regime is wave drag, often influencing the overall layout of an aircraft by dictating its cross sectional area distribution. The area distribution for the Viper is shown in Figure 2.8 along with the ideal Sears-Haack Type I and II curves. The Viper matches the Sears-Haack Type II curve well along the forward fuselage, except for the canopy. Good visibility dictates this irregularity. A large increase in cross-sectional area occurs where the wing and wing glove begin. Because the glove is relatively large, it virtually counteracts the favorable gradual area build up of the forward swept wing. This is an aspect that should receive further consideration in a future report. Coke-bottling the fuselage at this location may decrease the effect.

The maximum cross sectional area is attained at roughly the midpoint of the aircraft. According to Reference 2.10, this maximum should occur between 55-60% of the aircraft length. The cross-sectional area decreases rapidly along the aft portion of the fuselage which is undesirable from a wave drag point of view. The irregularity that occurs as a result of the empennage could be reduced through local coke-bottling of the booms.

Table 2.3 gives the weight summary for the Viper aircraft.

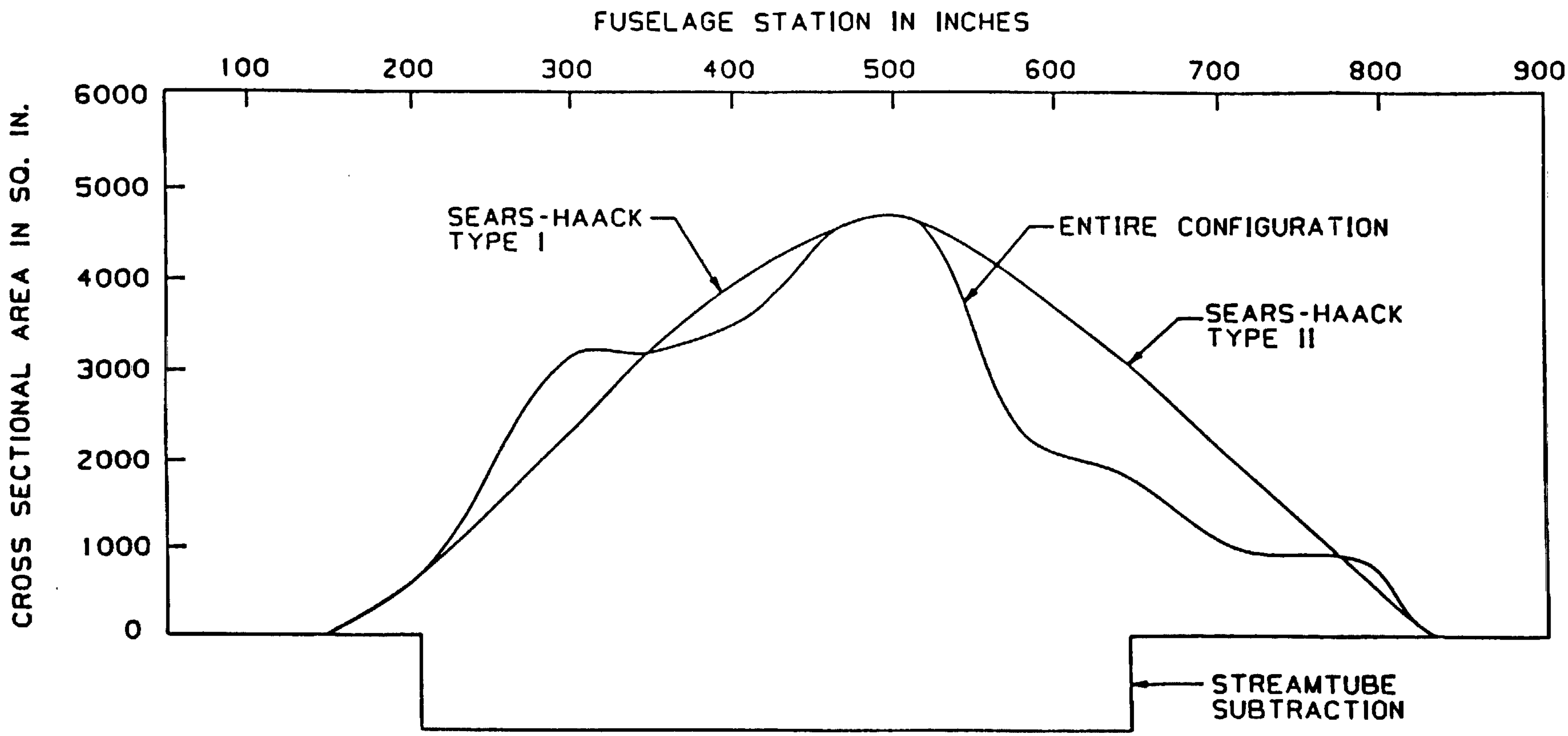


Figure 2.8 Hybrid Fan Vectored Thrust Aircraft Area Rule Plot

Table 2.3 Hybrid Fan Vectored Thrust Aircraft Weight Summary
(See Table 2.1 for CA, BAI#1, BAI#2, and BAI#3 Ordnance)

	MISSION			
	CA	BAI#1	BAI#2	BAI#3
STRUCTURE	(9055)			
Fuselage	5253			
Wing	2065			
Tails - Horizontal	324			
- Vertical	996			
Landing Gear - Main	996			
- Nose	176			
PROPULSION	(6802)			
Cruise Engine	5580			
- includes nozzles				
Air Induction	475			
Fuel Bladder	490			
Fuel Dumping	25			
Engine Controls	22			
Starting System	189			
Water Injection	21			
FIXED EQUIPMENT	(4591)			
Flight Control	1047			
Avionics	1164			
Electrical System	551			
Air Conditioning	254			
Oxygen System	17			
APU	278			
Furnishings	278			
Gun and Provisions	630			
Auxilary Gear, Paint	372			
STOVL EQUIPMENT	(423)			
RCS Equipment	423			
TOTAL EMPTY WEIGHT	20871	20871	20871	20871
Crew	225	225	225	225
Total Fuel	10754	10754	10754	10754
Armament	(1290)	(3820)	(3820)	(4300)
ASRAAMS	400			
AMRAAMS	670			
HARM		3600		
Mk-82's			3600	
Maverick's				4080
Ammo - 200 rnds	220	220	220	220
TAKEOFF WEIGHT	33140	35670	35670	36150

2.4 MIXED FLOW VECTORED THRUST AIRCRAFT DESCRIPTION (NEMESIS)

A three view of the Nemesis aircraft, including geometric parameters, is given in Figure 2.8 and the internal layout is shown in Figure 2.9. The major aspects of the Nemesis configuration are discussed below.

The pilot's eye position is located to provide adequate visibility over the nose and sides of the aircraft. Additionally, the upper fuselage is carefully developed to avoid pilot "blind spots" behind the aircraft.

The large ducts needed for hover with the MFVT concept dictated the middle and aft fuselage width. The cockpit and radar sized the forward fuselage. Fuel volume considerations and the need for a long internal weapons bay for the AMRAAM sized the fuselage length. Volume beneath the engine inlet and ducts was dedicated to the main landing gear and ASRAAM missile storage.

Simple normal shock inlets were selected and sized to the Mach 1.6 supersonic dash requirement. A bifurcated inlet was selected so that the wide aft fuselage could be easily blended into the outside edges of the inlets. A chin inlet, e.g. F-16 Falcon, was not selected so as to avoid hot gas re-ingestion and FOD problems. The flat underside fuselage that developed from this integration should be beneficial in enhancing the fountain effects during hover.

A conventional aft swept wing was selected for the Nemesis. This was done so that a simply constructed wing with adequate performance could be developed. Strakes have been incorporated to improve aircraft lift and to maintain adequate airflow to the bifurcated inlet at high angles of attack.

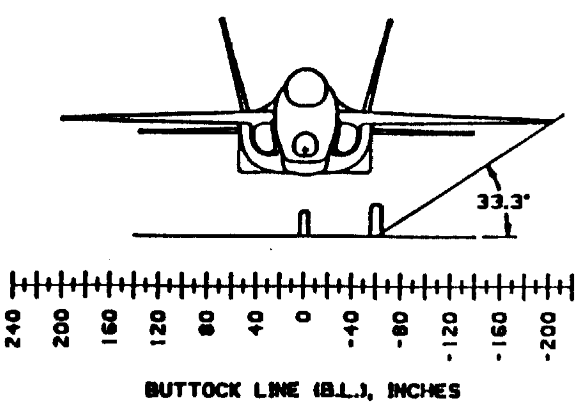
A tail aft configuration was selected for the Nemesis. This was done to keep the aerodynamic center near its originally estimated location, above the hover thrust location. Additionally, the MFVT propulsion system had already created a wide aft fuselage with adequate structural allowances for all moving stabilators. Twin vertical tails were selected to provide adequate directional stability throughout the flight envelope.

The area ruling plot for the Nemesis appears in Figure 2.11. The constant cross-sectional areas of the fuselage ahead of the wing and in the vicinity of the propulsion system kept the area distribution of the configuration from matching the ideal Sears-Haack shapes.

A weight summary for the Nemesis is shown in Table 2.4.

NEMESIS GEOMETRY

Length:		56.00 ft.
Height:		15.40 ft.
Main Gear Wheel Track		10.00 ft.
Wing		
	Span:	33.55 ft.
	Area:	321.60 ft^2
	Leading edge Sweep Angle:	40.00 deg.
	MGC:	11.67 ft.
	Airfoil:	NACA 64-206
	Aspect Ratio	3.5
Vertical Tail		
	Area	70.00 ft^2
	Span	7.25 ft.
	Leading edge Sweep Angle:	45 deg.
	Airfoil:	NACA 0004
	Aspect Ratio	1.50
Horizontal Tail		
	Area:	260.13 ft2
	Span:	14.00 ft.
	Leading Edge Sweep Angle:	30 deg.
	Airfoil:	NACA 0004
	Aspect Ratio	3.26



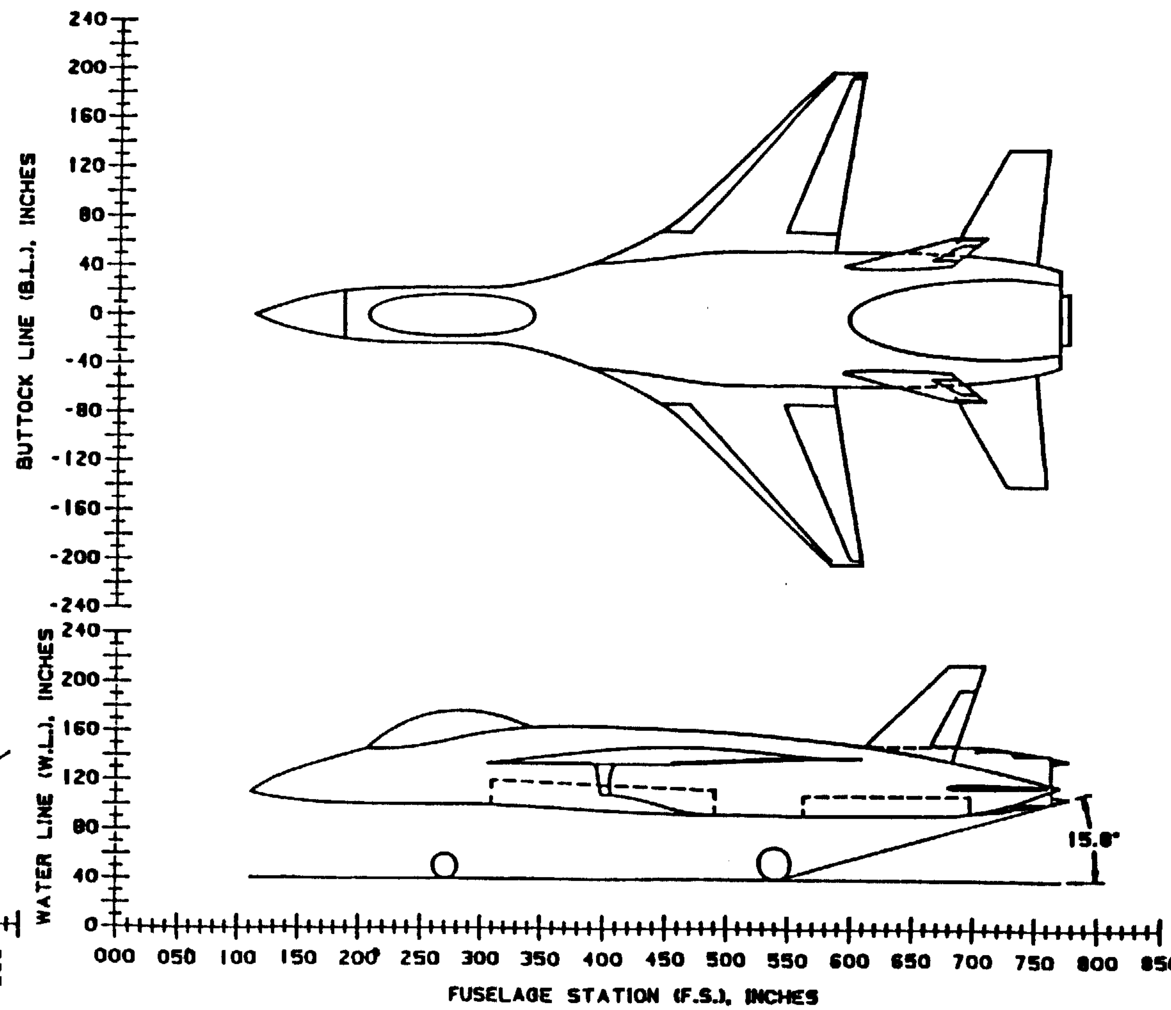


Figure 2.9 Mixed Flow Vectored Thrust Aircraft Three View

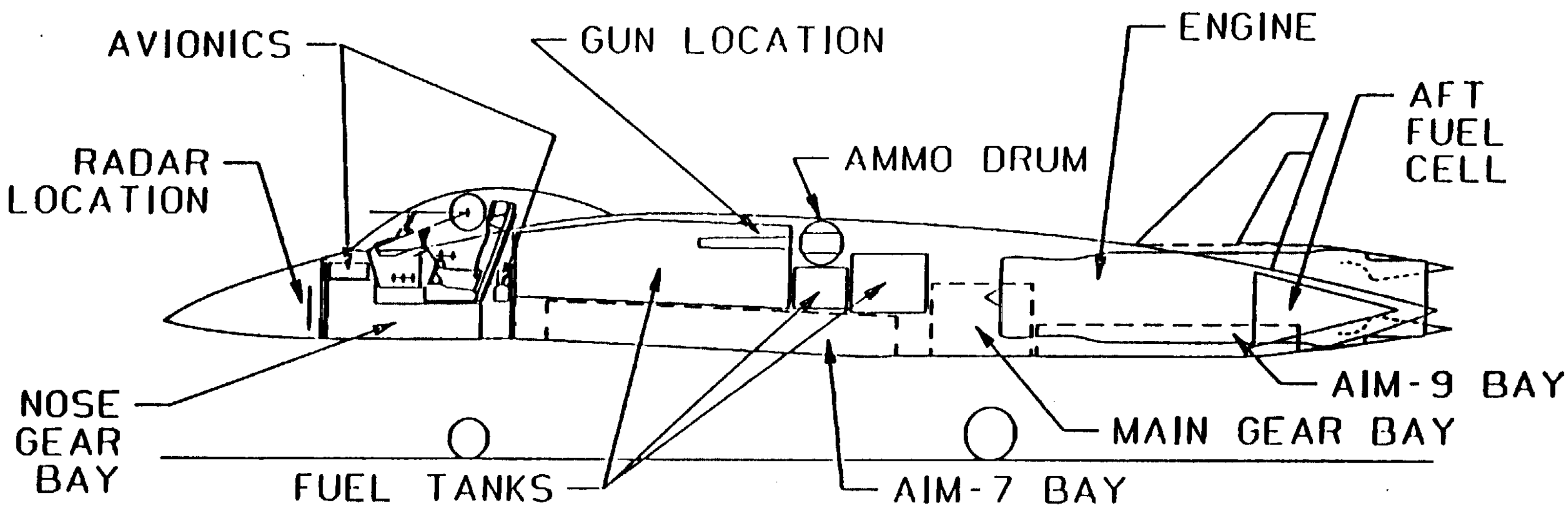


Figure 2.10 Mixed Flow Vectored Thrust Aircraft Internal Layout

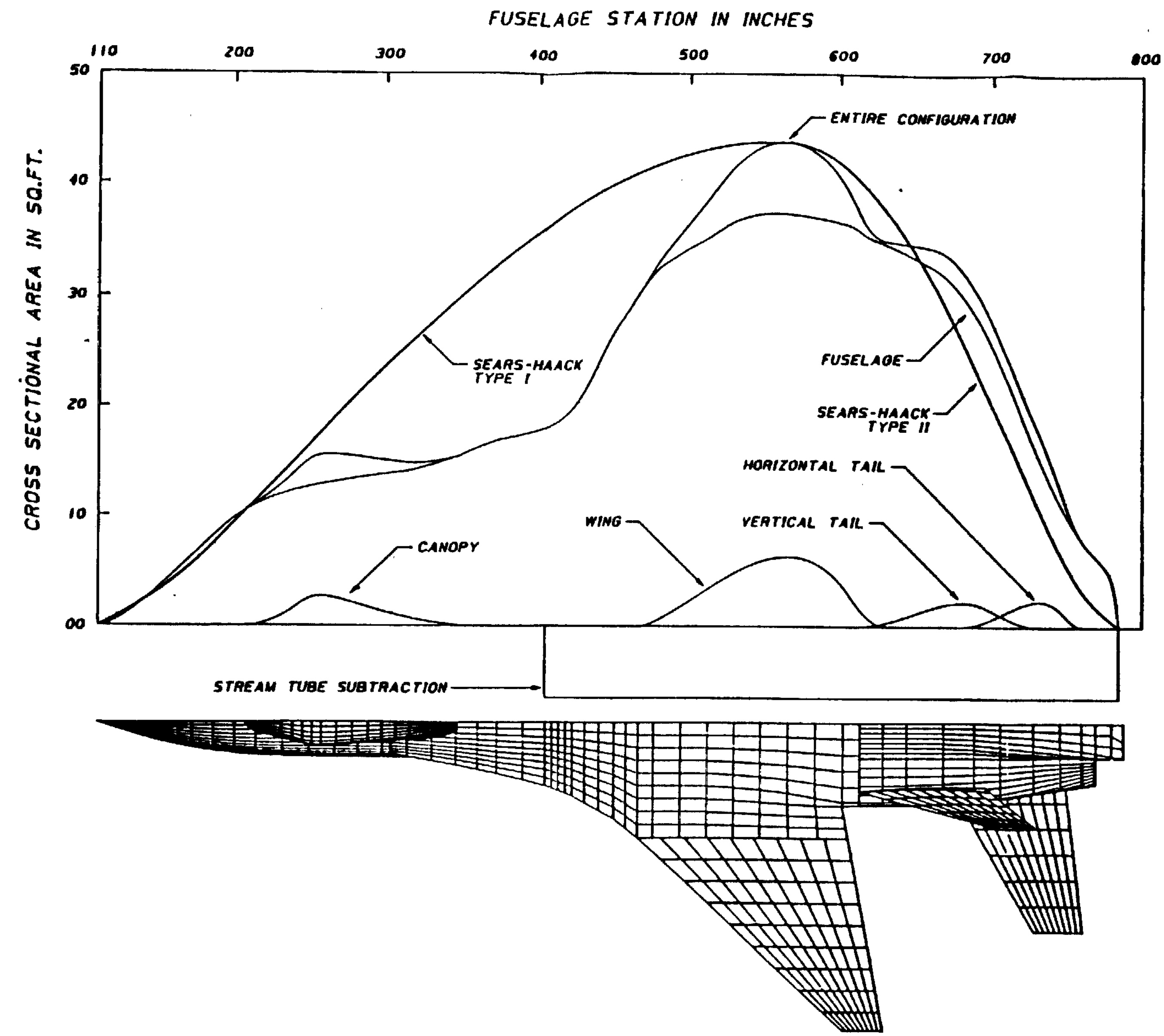


Figure 2.11 Mixed Flow Vectored Thrust Aircraft Area Rule Plot

Table 2.4 Mixed Flow Vectored Thrust Aircraft Weight Summary
(See Table 2.1 for CA, BAI#1, BAI#2, and BAI#3 Ordnance)

	MISSION			
	CA	BAI#1	BAI#2	BAI#3
STRUCTURE	(7982)			
Fuselage	4237			
Wing	1741			
Tails - Horizontal	302			
- Vertical	379			
Landing Gear - Main	1150			
- Nose	173			
PROPULSION	(5983)			
Cruise Engine	4728			
Air Induction	536			
Fuel Bladder	484			
Fuel Dumping	26			
Engine Controls	26			
Starting System	162			
Water Injection	21			
FIXED EQUIPMENT	(4560)			
Flight Control	1033			
Avionics	1164			
Electrical System	551			
Air Conditioning	254			
Oxygen System	17			
APU	272			
Furnishings	277			
Gun and Provisions	630			
Auxilary Gear, Paint	363			
STOVL EQUIPMENT	(1365)			
Remote lift ducts	305			
Insulation	160			
Block and Turn Nozzle	450			
Clamshell Nozzle	450			
TOTAL EMPTY WEIGHT	19890	19890	19890	19890
Crew	225	225	225	225
Total Fuel	10575	10575	10575	10575
Armament	(1290)	(3820)	(3820)	(4300)
ASRAAMS	400			
AMRAAMS	670			
HARM				
Mk-82's		3600		
Maverick's			3600	4080
Ammo - 200 rnds	220	220	220	220
TAKEOFF WEIGHT	31980	34510	34510	34990

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REFERENCES FOR CHAPTER 2

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2.3 Cox, Brian, et. al., Preliminary Design of a Supersonic STOVL Aircraft Using a Mixed Flow Vectored Thrust Engine Cycle, University of Kansas, AE 621, December 1989.

2.4 Park, P.H., Fouga, S.G., and Kitowski, J.V., "Synthesis of the Next USAF Fighter - An Overview of Possible Requirements For an Advanced Multirole Fighter", AIAA/AHA/ASEE Aircraft Design, Systems and Operations Conference, Seattle, WA, August, 1989.

2.5 Cox, Brian, et al, Battlefield Arena in 1995-2000, University of Kansas, AE 621, September, 1989.

2.6 Cox, Brian, et al, Preliminary Performance Assessment for a Supersonic STOVL Fighter, University of Kansas, AE 621, September, 1989.

2.7 USAF Wright Research and Development Center Presentation, Presenter: Second Lieutenant Gerald A. Swift, University of Kansas, October, 1989.

2.8 Hunt, B., and Titiriga, A., F-20 Evolution, AIAA-86-2612, Northrop Corporation, Hawthorne, CA, October, 1986.

2.9 Dollyhigh, S.M., et al, Development and Analysis of a STOL Supersonic Cruise Fighter Concept, NASA Technical Memorandum, 85777, March 1984.

2.10 Nelson, B.D., Design Scope for Student Supersonic Projects, AIAA/AHS/ASEE Aircraft Systems, Design and Technology Meeting, October, 1986.

3. PHASE I AIRCRAFT COMPARISONS AND SELECTION OF PHASE II AIRCRAFT

The purpose of this chapter is to compare the three STOVL aircraft from Phase I of the study and based on this comparison select the aircraft for Phase II. The aircraft are compared using the following parameters:

3.1 Aircraft Weights and Cost,
3.2 Aircraft Performance and Mission Capability
3.3 Area Rule and Drag Characteristics
3.4 Aircraft Components Required for STOVL Capability

The selection of the Phase II aircraft is discussed in Section 3.5.
It is important to note that the configurations presented here are not converged designs due to lack of time in the Phase I study. Nevertheless, it is felt that the comparisons made here are still valid for preliminary design purposes.

3.1 AIRCRAFT WEIGHTS AND COST

A comparison of the Phase I aircraft weights for the counter air mission are given in Table 3.1.

Table 3.1 Phase I Aircraft Weight Comparison (Counter Air Mission)
(all weights in pounds)

	LIFT	HFVT	MFVT
W_{TO}	30581	33140	31980
W_E	18758	20871	19890
W_{struct}	7349	9055	7982
W_{prop}	6235	6802	5983
W_{feq}	4484	4591	4560
W_P	10308	10754	10575

The HFVT aircraft is the heaviest due to its propulsion system and the boom arrangement. The LIFT aircraft is the lightest configuration which is consistent with data presented in Reference 3.1.

The aircraft cost is summarized in Table 3.2. These cost estimates, in 1995 dollars, are based on 1,000 aircraft operating 350 flight hours per year for 20 years.

Table 3.2 Phase I Aircraft Cost Comparison (Billions of 1995 Dollars)

	LIFT	HFVT	MFVT
Life Cycle Cost	59.42	59.72	59.26
Research, Test, Development and Evaluation	3.45	3.45	2.84
Acquisition Cost	19.46	19.76	20.21
Operating Cost	35.91	35.91	35.62
Disposal Cost	0.60	0.60	0.59
Cost per Aircraft (millions)	22.90	23.20	23.10

Although LIFT aircraft is the lightest configuration, its added cost for the lift engine makes its cost as much as the MFVT and HFVT configurations.

3.2 AIRCRAFT PERFORMANCE AND MISSION CAPABILITY

The performance requirements from the missions specifications were verified and are shown for the three aircraft in Table 3.3. The aircraft meet the required performance except for the time to climb and specific excess energy for the LIFT and HFVT configurations. The lack of adequate performance shown is due to optimistic estimation of the wave drag in the preliminary sizing of the aircraft. The MFVT aircraft, which requires dry thrust for vertical operations, met the requirements since its engine was oversized for hover.

The mission capability was measured by estimating the fuel required to meet the design missions (see Figures 2.1 and 2.2). Table 3.4 shows the mission fuel burn for the aircraft. All three configurations can meet the mission ranges with the MFVT aircraft using the least amount of fuel.

Table 3.3 Phase I Study Aircraft Point Performances

Requirement					
H	M	Value	LIFT	HFVT	MFVT
0	0	0 to 40k, 2 min	2.51	2.18	2.00
30000	0.9	500 ft/sec	417	516	510
10000	0.9	1000 ft/sec	933	980	1120
15000	0.8	15 deg/sec	16.34	16.19	17.90
30000	0.9	9 deg/sec	9.22	9.85	10.60
30000	1.2	8 deg/sec	10.10	7.92	7.80
30000	1.6	4.5 g	8.67	6.88	6.50
15000	0.9	6.5 g	8.25	8.18	9.60
30000	0.9 to 1.6 in 70 sec		40.20	38.10	34.70
30000	0.5 to 1.4 in 80 sec		57.70	52.40	49.30
10000	0.3 to 0.9 in 22 sec		17.80	16.80	16.10

Table 3.4 Phase I Aircraft Mission Fuel Burn

	LIFT	HFVT	MFVT
CA Mission	7509 lbs	8062 lbs	6917 lbs
BAI Mission	9695 lbs	10299 lbs	7995 lbs

3.3 AREA RULE AND DRAG CHARACTERISTICS

The area rule plots for the three configurations were shown in Chapter 2. A comparison of the area rule and drag characteristics is given in Table 3.5.

Table 3.5 Phase I Area Rule and Drag Characteristics

	LIFT	HFVT	MFVT
Match with Sears-Haack	Fair	Unacceptable	Unacceptable
Maximum Area	4073 in^2	4709 in^2	6303 in^2
Wave Drag Increment at M=1.6	0.012	0.016	0.016
Aircraft Skin Friction Coeff. (M=0.8, H=30000 ft)	0.0036	0.0051	0.0030

The unconventional fuselage shaping for the propulsion systems of the HFVT and MFVT concepts caused unacceptable area rule plots and also large maximum cross sectional areas, both of which increase wave drag.

3.4 AIRCRAFT COMPONENTS REQUIRED FOR STOVL CAPABILITY

The weight and volume for the components required for STOVL capability are presented here. Table 3.6 shows the components required for each aircraft along with their weights and volumes.

Table 3.6 Weight and Volume Data for STOVL Components

	Volume (ft^3)	Weight (lbs)
LIFT		
* Lift Engine	21	647
* Ventral Nozzle and Turning Vanes	*	300
* RCS System	8	390
Total	29	1337
HFVT		
* Flow Switching Mechanism and Extended Power Shaft	83	1351
* Front Vectoring Nozzles	2	*
* Rear Vectoring Nozzle	*	*
* Penalty for Booms	117	1112
* RCS System	6	423
Total	208	2886
MFVT		
* Block and Turn Nozzle	*	450
* Transfer Ducts	92	465
* Front Clamshell Nozzles	2	450
Total	94	1365

The HFVT configuration suffers the most from the STOVL equipment for two reasons. First, the engine components required for flow shifting are heavy and require a large volume. Second, the engine thrust split requires the engine to be at the center of the aircraft and thus some sort of boom configuration. The LIFT and MFVT configurations have similar weight penalties but the MFVT has a larger volume penalty due to the transfer ducts.

3.5 SELECTION OF PHASE II AIRCRAFT

The lift+lift/cruise configuration was selected for the Phase II aircraft study. The reasons for this selection were:

1) The LIFT configuration exhibited the most promising area rule distribution.
2) The technology required for this configuration is the most consistent with the 1995 Technology Availability Date (TAD) assumed for the study.
3) The LIFT aircraft was the lightest configuration.

At the start of the Phase II study, the LIFT configuration was iterated to reflect comments made about the design from References 3.1-3.3. The following were the drivers for the iteration:

It is good if:
* the aircraft center of gravity moves aft
* the CA and BAI mission cg's in hover are aligned
* the rear thrust post is moved forward
* the aircraft has three posts instead of two
* the lift engine is small

With these considerations, the following modifications were made to the design:

1) The aircraft has a horizontal tail, not a canard. The purpose of this iteration was:
   * to move the cg further aft
   * to reduce the complexity in the main inlet region
   * to have more favorable stability margins

2) The aircraft has three posts instead of two. This was done to allow for:
   * reduced suckdown in ground effect
   * hover roll control through differential area change

3) The wing was shifted forward 10 inches to achieve a smaller positive stability margin in supersonic flight.

4) The avionics were moved aft in the aircraft behind the internal weapons bay to move the hover cg rearward, thus decreasing the size of the lift engine.

5) The BAI mission payloads were changed to reflect more realistic missions according to Reference 3.1 and 3.3. The mission payloads are now configured to allow carrying radar guided weapons (Mavericks and HARM's) along with unguided weapons (Mk 82), thus having the aircraft capable to deliver munitions even if the target shuts off its

radar.

The BAI missions (two of them) were changed to:

* BAI Mission #1 - Four Mk-82's and two HARM's
* BAI Mission #2 - Four AGM-65 and two Mk-82's

6) The short range missiles were placed on the wing tip for two reasons:
   * the target field of view of the missile is greatly enhanced
   * the missile must have "lock-on" before it is launched, and external carriage allows more operational freedom.

7) The design missions were scaled down to get a more realistic fuel fraction according to Reference 3.1. The counter air mission was scaled down to a 100 nm subsonic cruise and a 50 nm supersonic cruise. The battlefield air interdiction mission was scaled down to a 200 nm subsonic high level cruise and an 80 nm low level dash.

8) Actual data of the General Dynamics F-16 and Grumman F-14 wave drag increments were used to estimate the wave drag of the configuration.

9) The weights of the following components were adjusted based on previous industry and government aircraft studies and actual aircraft:
   * cruise engine
   * installed avionics
   * reaction control system
   * rear and ventral nozzles
   * internal weapons launching mechanisms

10) The landing gear was re-sized for soft ground capability.

11) The wing thickness ratio is 4.5 percent for more favorable area rule characteristics.

The result of these design modifications is described in Chapter 4.

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3.3 NASA Ames Research Center Presentation, Presenter: Andrew Hahn, Powered Lift Technology Branch, University of Kansas, February, 1990.

4. CONFIGURATION DESCRIPTION

The purpose of this chapter is to give the configuration description of the Monarch lift+lift/cruise supersonic STOVL aircraft. A three view of the aircraft with a table of geometric parameters is shown in Figure 4.1. The internal layout is shown in Figure 4.2.

The requirements that had a major impact on the Monarch design are:

* the short takeoff and vertical landing capability,
* the supersonic cruise and combat conditions,
* and the internal volume for medium range missiles.

The Monarch configuration decouples the short takeoff and vertical landing capability from the supersonic requirements by employing a lift+lift/cruise engine cycle. The lift engine, sized for the hover flight condition, allows the mission performance requirements to size the lift/cruise engine, thus making the propulsion system integration of the Monarch a more conventional integration than other STOVL concepts. The Monarch has a pitch and yaw vectoring nozzle system to allow for enhanced maneuvering a post stall conditions and, in the yaw axis, to augment the directional control.

The supersonic cruise and combat conditions required the Monarch to have a smooth area rule distribution that matched the ideal Sears-Haack shape. Figure 4.3 shows that the Monarch met this requirement. The internal volume required for the medium range missiles was offset by the wing thickness selection. The Monarch uses a 4.5 percent thickness to chord ratio for its aft swept wing. The strake on the wing was included to provide for delayed wing stall at high angle of attack and for vortex lift in maneuvering. The empennage of the Monarch consists of a single vertical tail and all moving horizontal stabilators. The size of the vertical tail was reduced and the rudder removed by using the yaw vectoring nozzle. The size and placement of the stabilators were selected with the desire for the Monarch to have minimal trim drag throughout the flight envelope.

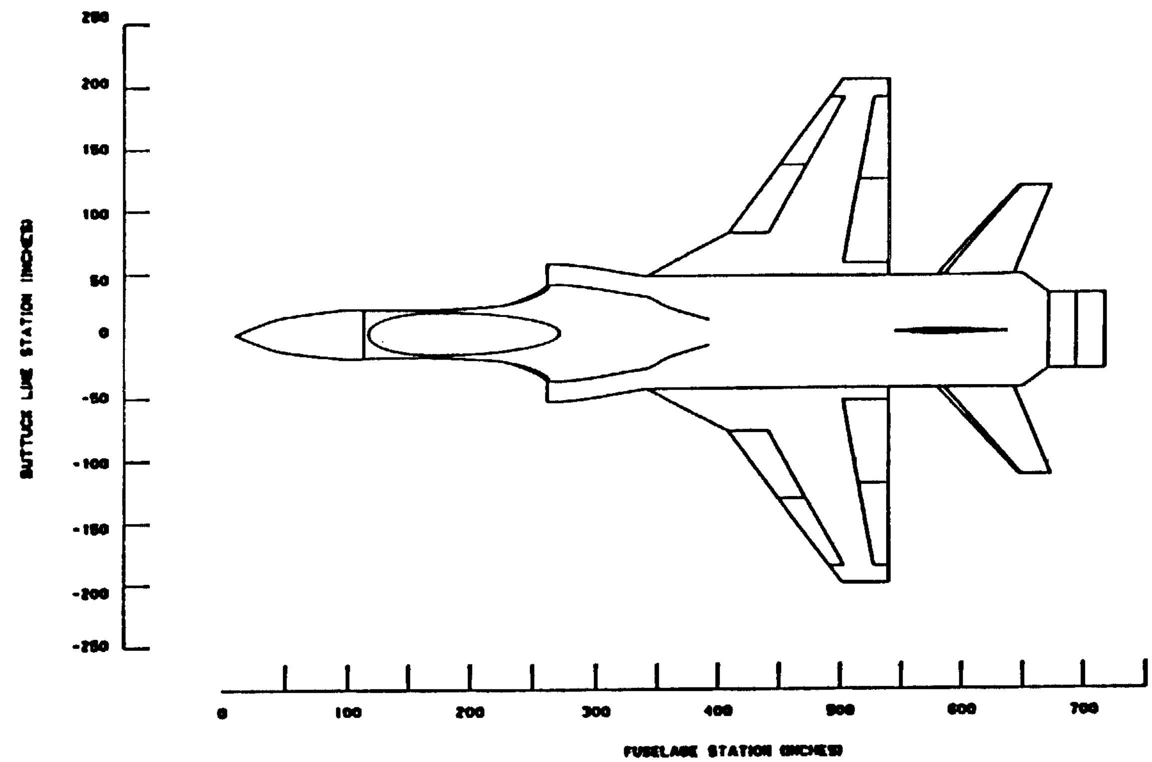
The high inlet placement was the result of two requirements. First, a low inlet placement would have required the inlet to shape itself around the internal weapon bay which was not desirable since this would have distorted the flow. Second, a higher inlet placement leads to less severe hot gas reingestion and foreign object damage problems.

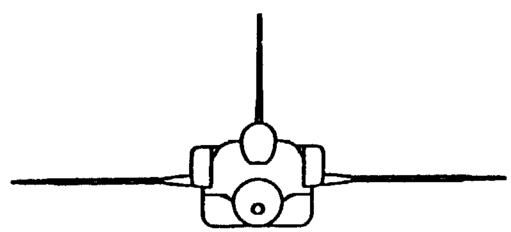
The Monarch carries two medium range missiles internally and two short range missiles on the wing tips for the counter air mission. For the fallout battlefield air interdiction mission, the Monarch carries a combination of guided and unguided munitions on wing pylons. Wing pylons were selected for two reasons. First, pylon mounted stores allowed more flexibility in maintaining a constant hover center of gravity, which is important to the lift engine sizing. Second, stacking munitions underneath the fuselage interfered with the internal weapons bay doors, eliminating the possibility of a combined counter air and battlefield air interdiction mission.

The landing gear tires for the Monarch are oversized for a fighter due to the fact that a STOVL type aircraft may often find itself in an austere battlefield scenario with soft field landing and takeoff conditions.

MONARCH GEOMETRY

Length:	56.0 ft	
Height:	6.8 ft	
Main Gear Wheel Track:	7.5 ft	
Wing		
	Equivalent	Actual
Span:	33.67 ft	33.67 ft
Area:	347.9 ft^2	322.0 ft^2
LE Sweep:	40.3 deg	37.8 deg
MGC:	12.0 ft	9.0 ft
Airfoil t/c:	0.045	0.045
Aspect Ratio:	3.26	3.50
Vertical Tail		
Span:	8.4 ft	
Area:	43.6 ft^2	
LE Sweep:	40.5 deg	
MGC:	5.56 ft	
Airfoil t/c:	0.045	
Aspect Ratio:	1.61	
Horizontal Tail		
Span:	11.5 ft	
Area:	40.0 ft^2	
LE Sweep:	42.8 deg	
MGC:	3.71 ft	
Airfoil t/c:	0.06	
Aspect Ratio:	3.25	





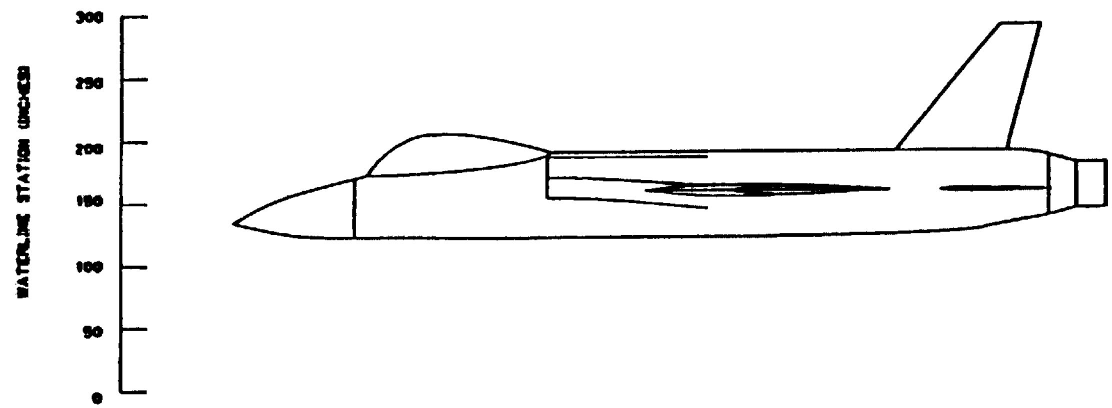


Figure 4.1 Monarch Lift+Lift/Cruise Aircraft Three View

1. Radar and Equipment
2. Nose Landing Gear Bay
3. ACES II Zero/Zero Ejection Seat
4. Dedicated Lift Engine
5. Fuel Pump
6. On Board Oxygen Generating System
7. Heat Exchanger and Air-Conditioning Unit
8. Forward Fuselage Fuel Tank
9. Internal Weapons Bay for AIM-7s
10. Ammunition Drum
11. Hydraulic Reservoir
12. Aft Fuselage Fuel Tank
13. Avionics Bay
14. Main Landing Gear Bay
15. Cruise Engine
16. Jet Fuel Starter
17. ECM Pod
18. Chaff and Flare Pod

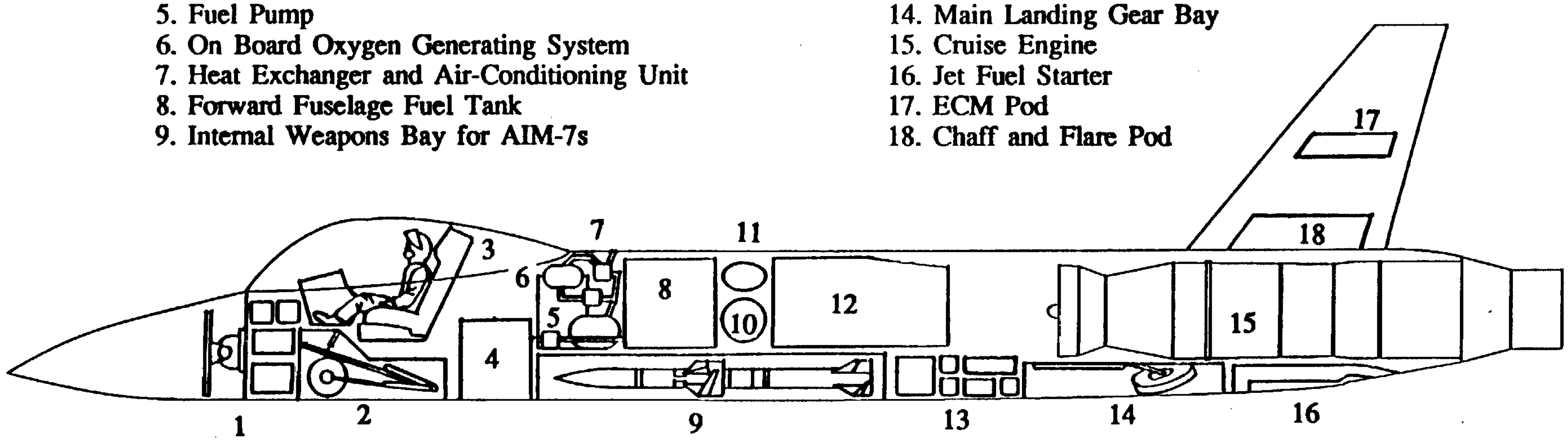


Figure 4.2 Monarch Lift+Lift/Cruise Aircraft Internal Layout

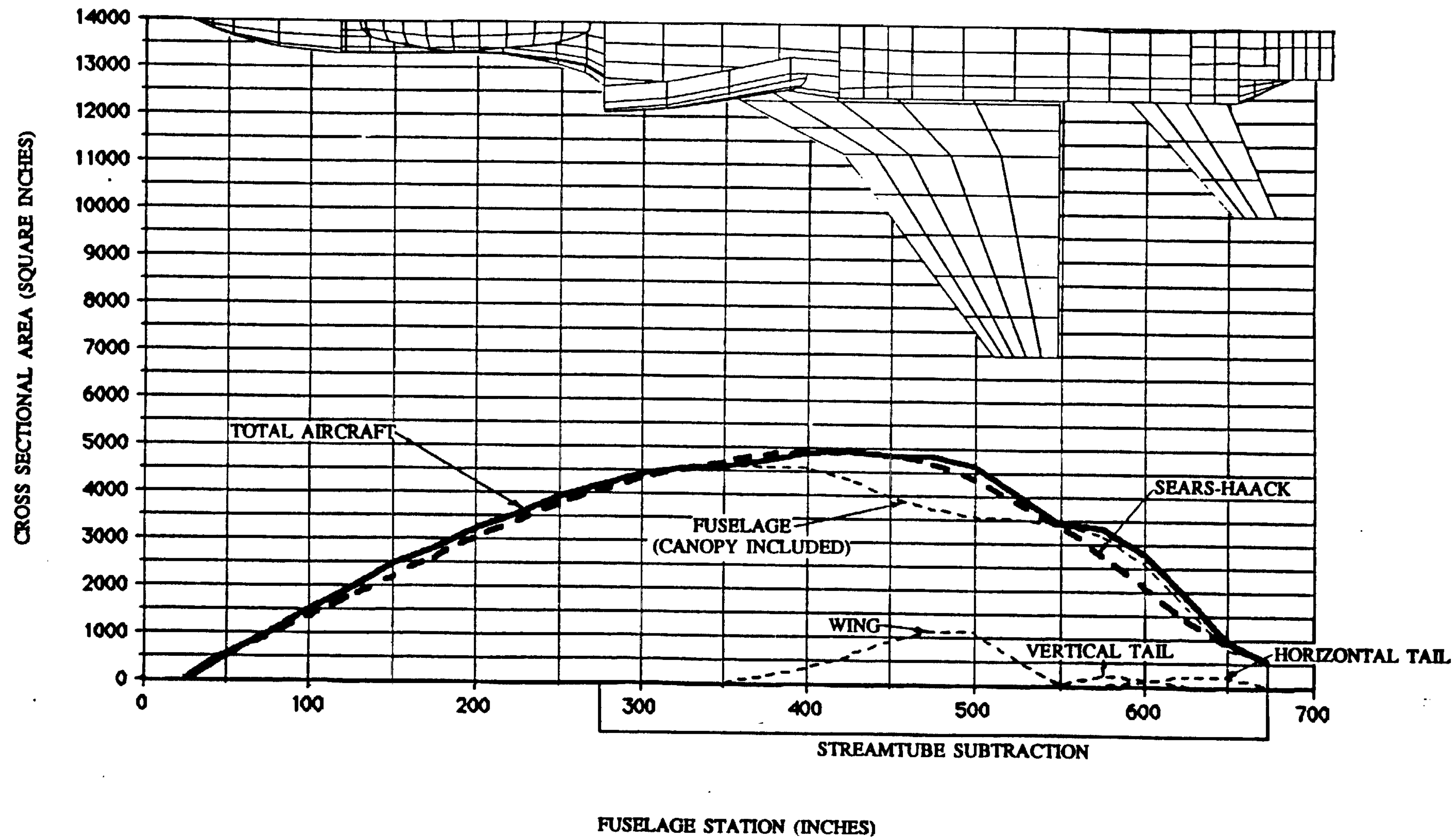


Figure 4.3 Monarch Lift+Lift/Cruise Aircraft Area Rule Distribution

5. WEIGHT AND BALANCE

The purpose of this chapter is to present the Monarch weight and balance results. The weight and balance method is first presented, followed by the weight and balance data.

Figure 5.1 shows the weight and balance flow chart used for the design. As shown, the three primary drivers for the weight and balance are having:

* the hover cg and thrust center balanced,
* the inflight cg travel acceptable,
* and an acceptable static margin.

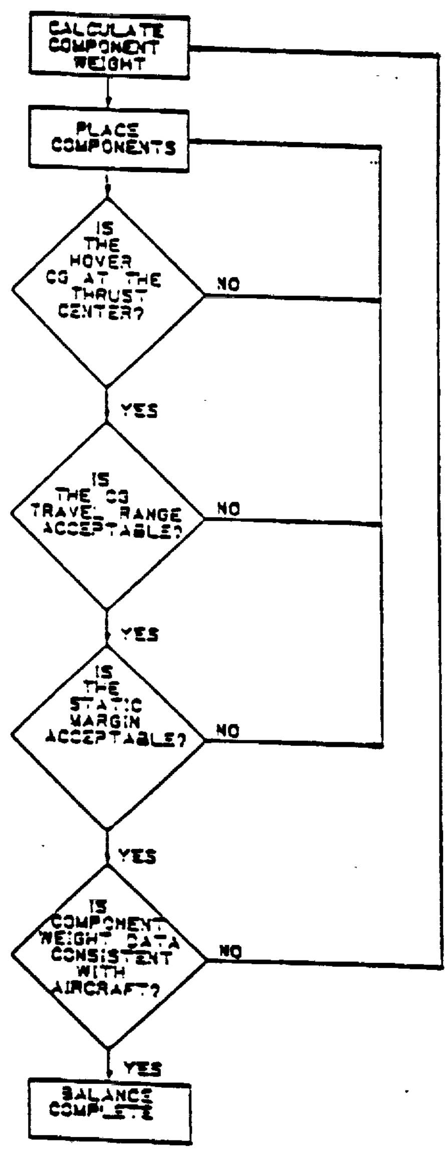


Figure 5.1 Weight and Balance Flow Chart

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A secondary driver is to assure that the weight data and placement of components are reasonable. The weight data were estimated using empirical weight equations of Reference 5.1 and actual weights from operational aircraft. The weight and balance calculations are shown in Appendix 1. The final weight statement for the Monarch is shown in Table 5.1.

The center of gravity excursion diagrams for the counter air and battlefield interdiction missions are shown, respectively, in Figure 5.2 and Figures 5.3 and 5.4. The inflight center of gravity travel is within the acceptable range given in Reference 5.2.

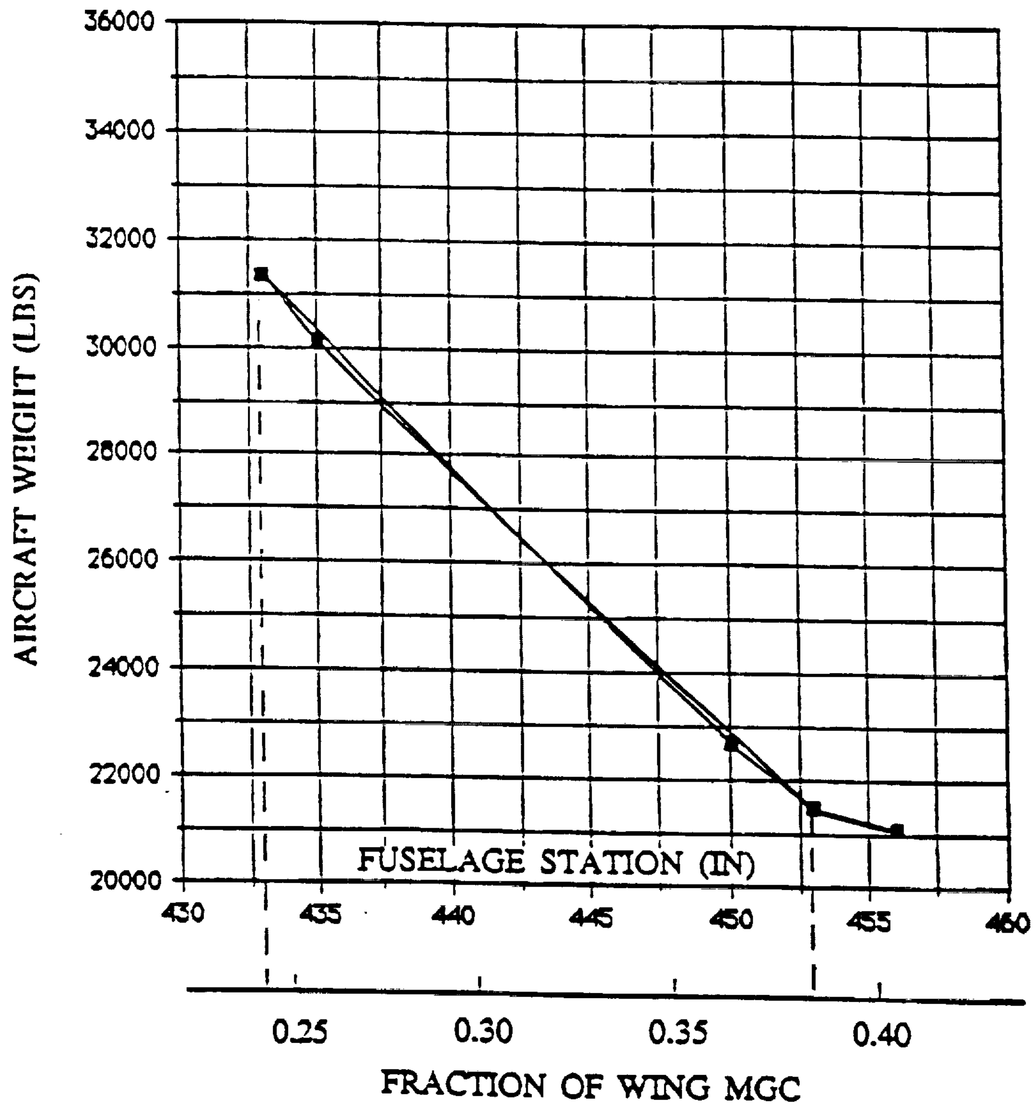

Figure 5.2 Counter Air Mission GC Excursion Diagram
(See Table 5.1 for CA, BAI#1, and BAI#2 Ordnance)

Table 5.1 Weight Summary of the Monarch Aircraft

	CA	BAI#1	BAI#2
STRUCTURE	(9498)		
Fuselage	4385		
Wing	2490		
Tails - Vertical	256		
- Canard	295		
Landing Gear - Main	1249		
- Nose	220		
Launch Mechanims (Int. Weap)			
ASRAAM	40		
AMRAAM	262		
Ventral Clamshell Nozzles	300		
PROPULSION	(6139)		
Cruise Engine	3557		
Lift Engine	480		
Cruise Engine Tailpipe Ext	300		
Cruise Engine Nozzle	420		
Air Induction	773		
Fuel Bladder	415		
Fuel Dumping	24		
Engine Controls	45		
Starting System	125		
FIXED EQUIPMENT	(5480)		
Flight Control	1021		
Avionics	1517		
Electrical System	596		
Air Conditioning	301		
Oxygen System	17		
APU	298		
Furnishings	277		
Gun and Provisions	630		
Auxilary Gear, Paint	418		
RCS Ducting and Nozzles	405		
TOTAL EMPTY WEIGHT	21117	21117	21117
Crew	225	225	225
Total Fuel	8642	8642	8642
Armament	(1196)	(4074)	(3316)
ASRAAMS	322		
AMRAAMS	654		
HARM		1614	
Mk-82's		2240	1120
Maverick's			1976
Ammo - 200 rnds	220	220	220
TAKEOFF WEIGHT	31336	34400	33642

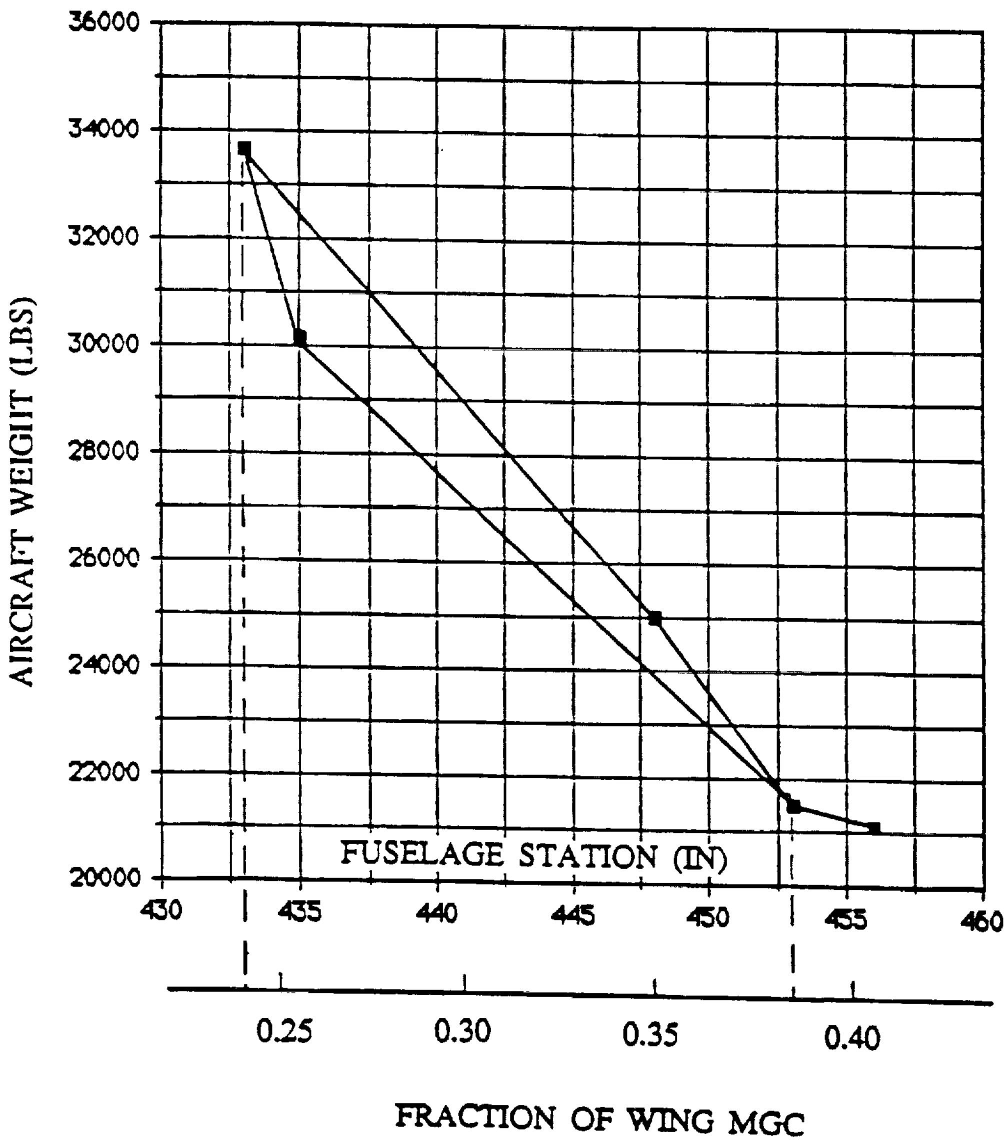

Figure 5.3 BAI #1 Mission CG Excursion Diagram
(See Table 5.1 for CA, BAI#1, and BAI#2 Ordnance)

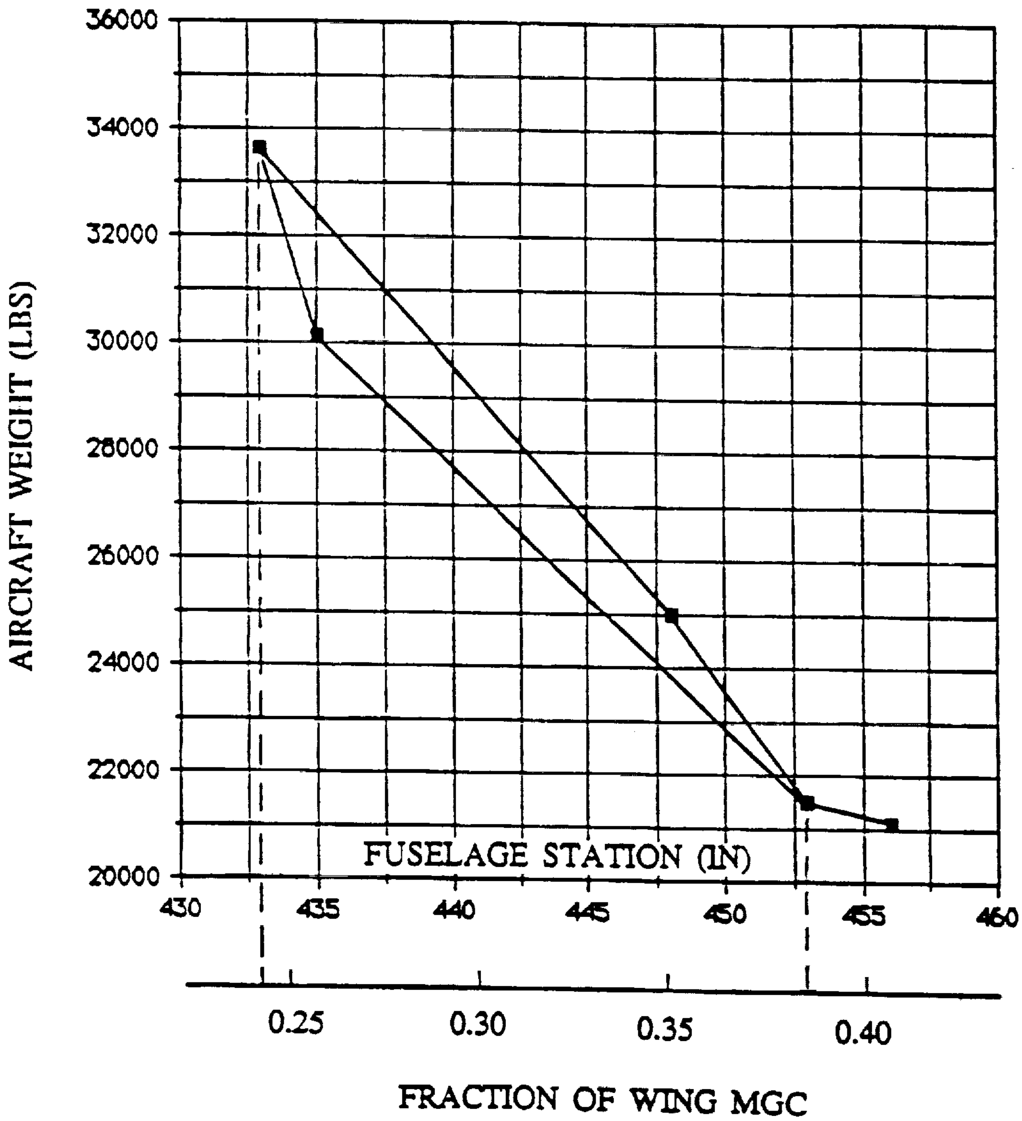

Figure 5.4 BAI #2 Mission CG Excursion Diagram
(See Table 5.1 for CA, BAI#1, and BAI#2 Ordnance)

REFERENCES FOR CHAPTER 5

5.1 Roskam, Jan, Part V, Component Weight Estimation, Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, KS, 66067, 1985.

5.2 Roskam, Jan, Preliminary Configuration Design and Integration of the Propulsion System, Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, KS, 66067, 1985.

6. PROPULSION SYSTEM INTEGRATION

The purpose of this chapter is to describe the integration of the Monarch propulsion system. Section 6.1 describes the cruise engine and 6.2 describes the lift engine. Each section describes the engine as well as the inlets and nozzles associated with the engine. Figure 6.1 shows the complete propulsion system as it is integrated in the airframe.

6.1 CRUISE ENGINE

The cruise engine that is used is based on an engine provided by Reference 6.1. The engine is designed to operate in both the hovering as well as the cruise/maneuver flight conditions. The following sub-sections will describe the engine as well as the inlets and nozzles.

6.1.1 Cruise Engine Description and Performance

The cruise engine was sized for both the hover and conventional wing-borne flight conditions. The total dry thrust required from the cruise engine during hover must be 1.30 times the weight of the aircraft in hover which is 24744 lbs. This factor is based on the following:

1) The total vertical thrust during hover must be sized to include the following factors:

   a) 1.0g is to provide a force to counter the weight of the aircraft.
   b) 0.1g is to enable the aircraft to counter a tenth of a g sink rate.
   c) 0.03g is for out-of-ground suckdown (assumed)
   d) 0.1g is for in ground effect suckdown (assumed)

2) The cruise engine must also be able to support the RCS which is 0.07g.

Based on these parameters, the required thrust from the cruise engine is 19,800 lbs dry. A point performance determined that a takeoff thrust-to-weight ratio of 1.15 was required for a maneuver flight condition; therefore, the engine must produce 35,450 lbs of thrust augmented. This means that the maneuver condition is more critical and determines the size of the engine. The base engine was resized using the following scaling laws from Reference 6.2.

$$\text{New Length} = \text{Base Length}^*(\text{New Thrust/Old Thrust})^{0.4}$$

$$\text{New Radius} = \text{Base Radius}^*(\text{New Thrust/Old Thrust})^{0.5}$$

$$\text{New Inlet Airflow} = \text{Base Airflow}^*(\text{New Thrust/Old Thrust})$$

Table 6.1 gives the Monarch engine parameters and the engine dimensions are shown in Figure 6.2. The engine weight includes the engine, fuel and oil systems, gear box, necessary plumbing, and mounting hardware. The performance plots for the installed cruise engine are shown in Figures 6.3 through 6.5 for three engine ratings: maximum augmented, maximum unaugmented, and a partial throttle setting. Figure 6.3 shows the mass flow rate for the cruise engine at various altitudes and mach numbers. Figures 6.4a

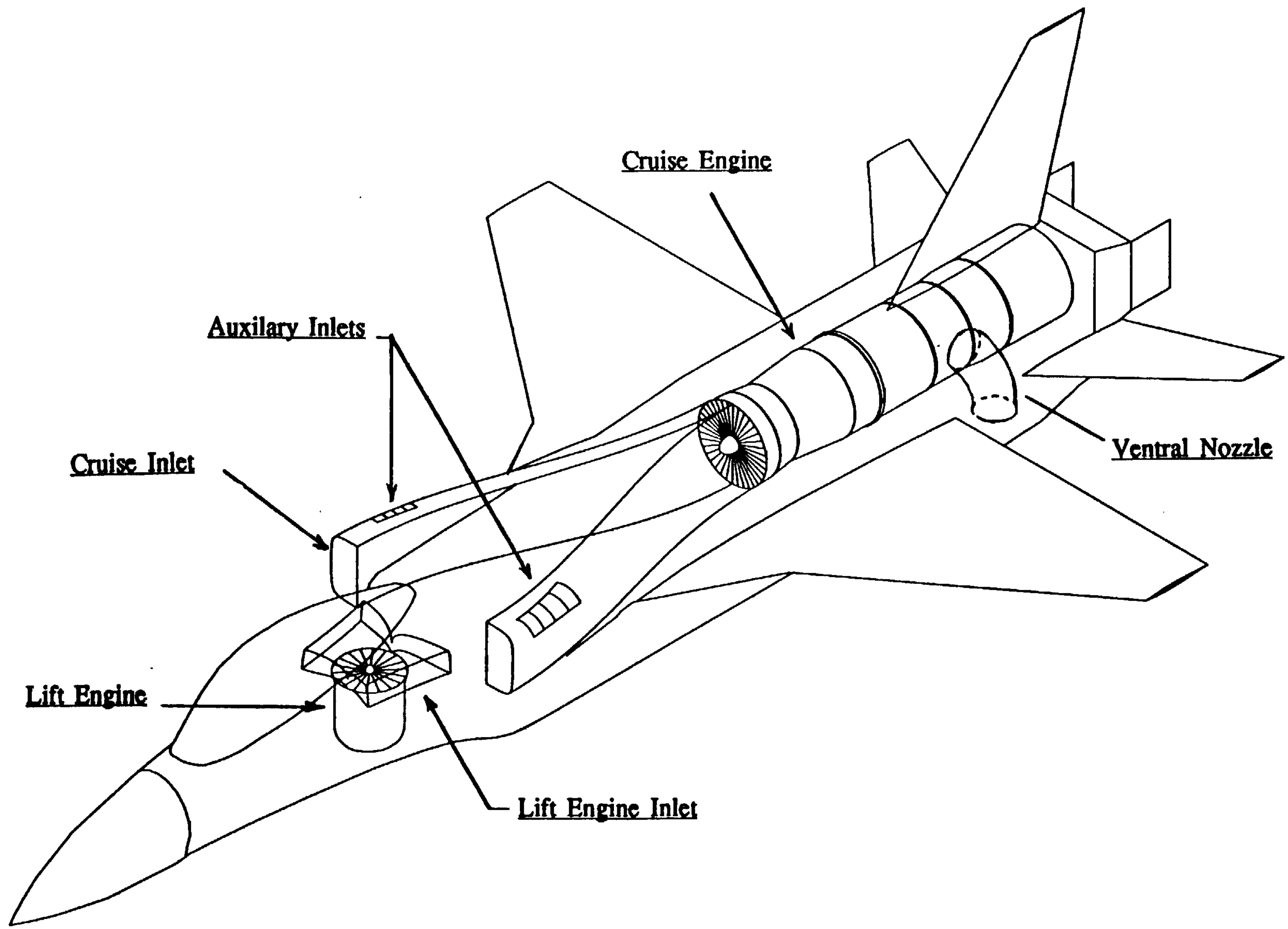

Figure 6.1 Monarch Engine Integration

through 6.4c shows the specific fuel consumption at partial throttle, maximum unaugmented thrust, and maximum augmented thrust, respectively. Figures 6.5a through 6.5c shows the thrust at partial throttle, maximum unaugmented thrust, and maximum augmented thrust, respectively.

Table 6.1: Cruise Engine Parameters

Rating	Max Dry Thrust	Max Aug. Thrust
Condition	SLS 90°F day	SLS 90°F day
Mass Airflow	319.64 lbm/sec	319.64 lbm/sec
Nozzle Throat Area	3.431 ft^2	5.268 ft^2
Bypass Ratio	0.80	0.80
Nozzle Pressure Ratio	3.268	3.096
Net Thrust	24,673 lbs	35,573 lbs
Diameter	44 in.	44 in.
Length	184 in.	184 in.
Weight	3557 lbs	3557 lbs

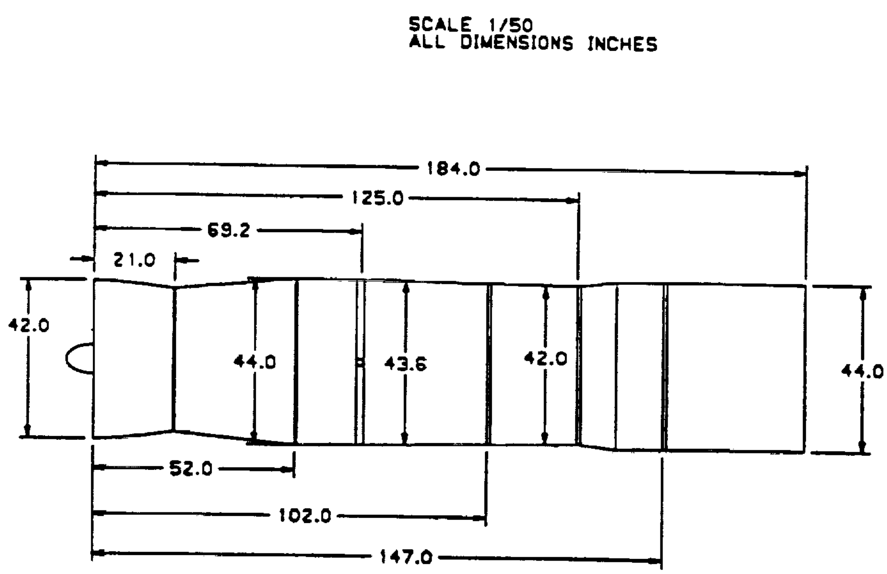

Figure 6.2 Cruise Engine for the Monarch (nozzles not included)

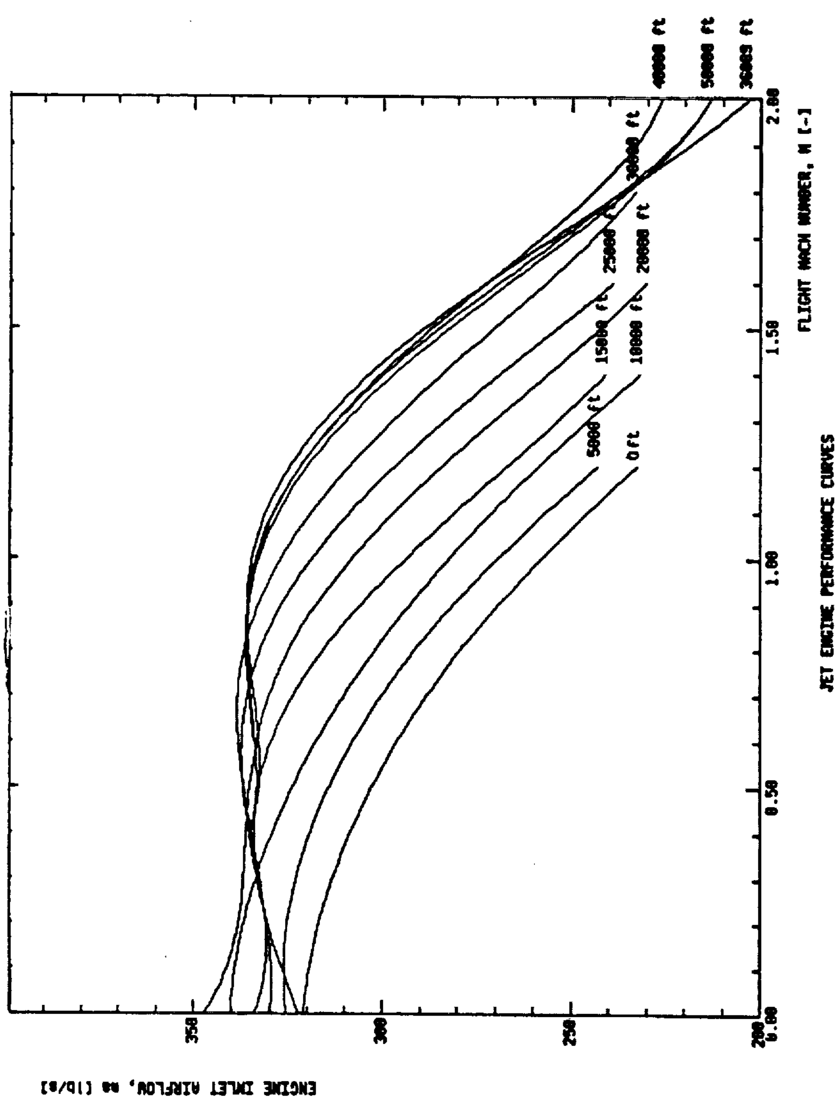

Figure 6.3 Mass Flow Rate for the Monarch

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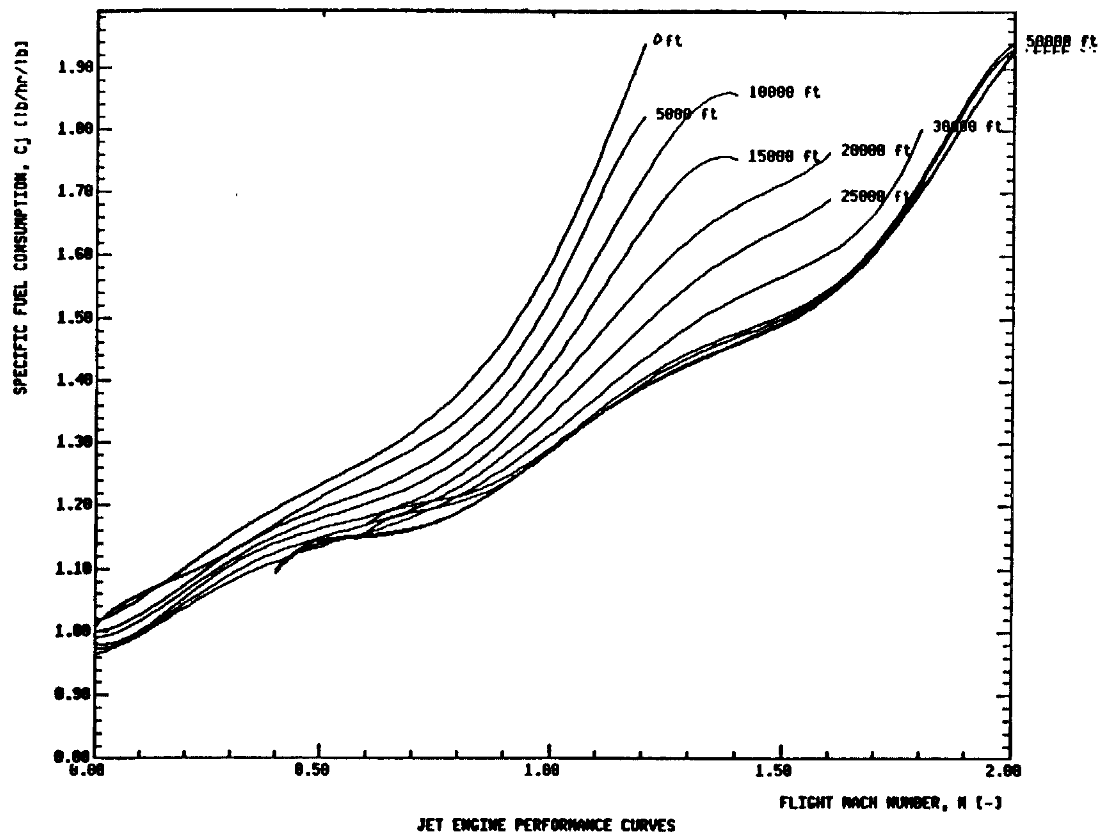

Figure 6.4a Specific Fuel Consumption at a Partial Throttle Setting for the Monarch

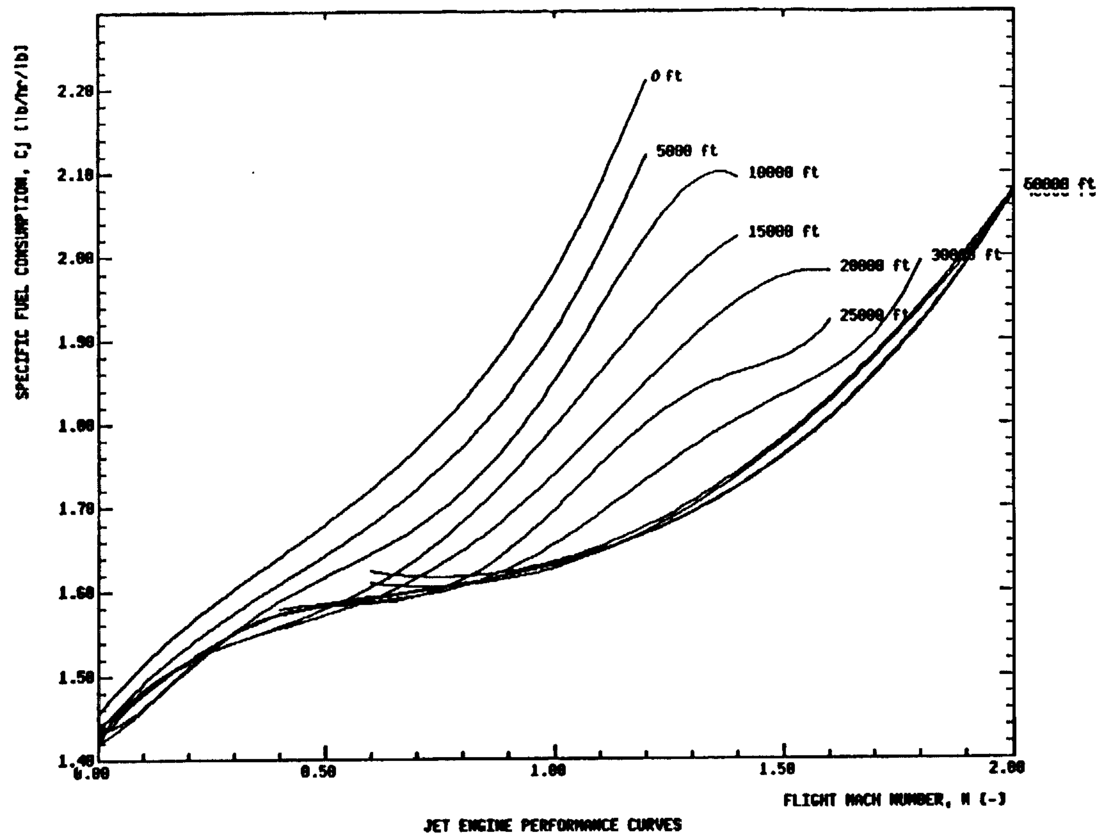

Figure 6.4b Specific Fuel Consumption at Maximum Unaugmented Thrust for the Monarch

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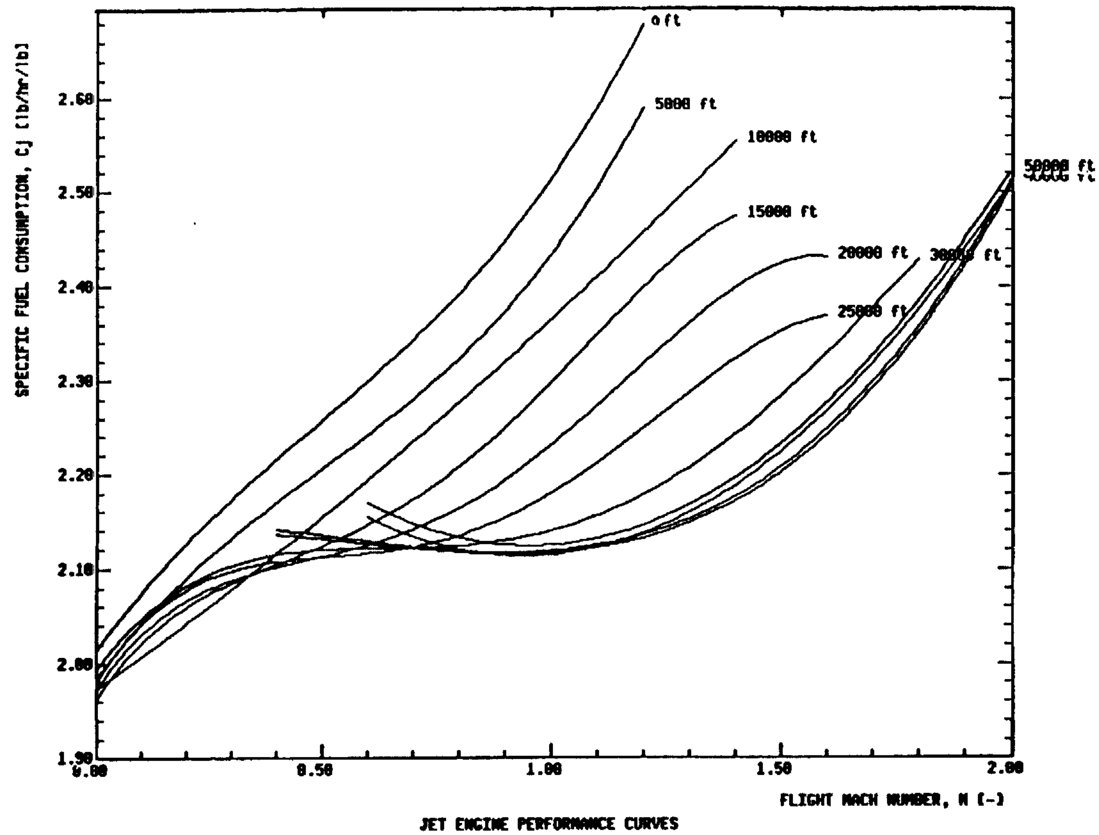

Figure 6.4c Specific Fuel Consumption at Maximum Augmented Thrust for the Monarch

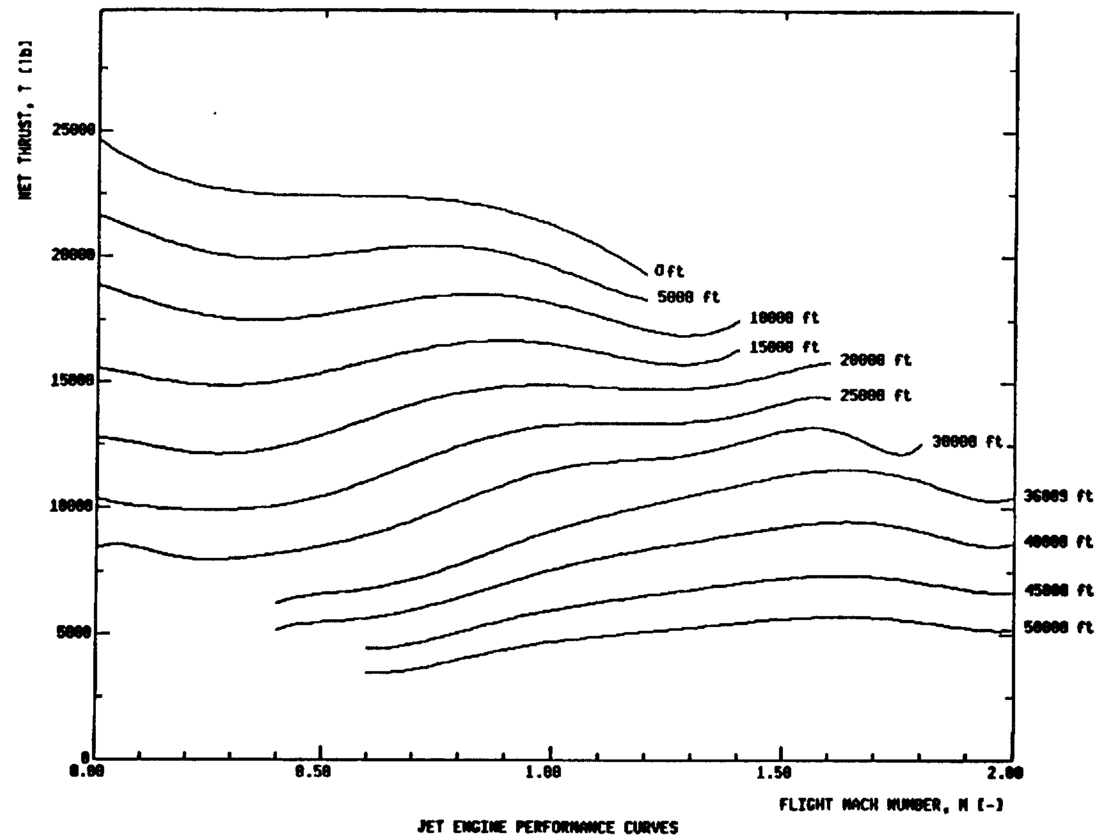

Figure 6.5a Partial Throttle Setting Thrust for the Monarch

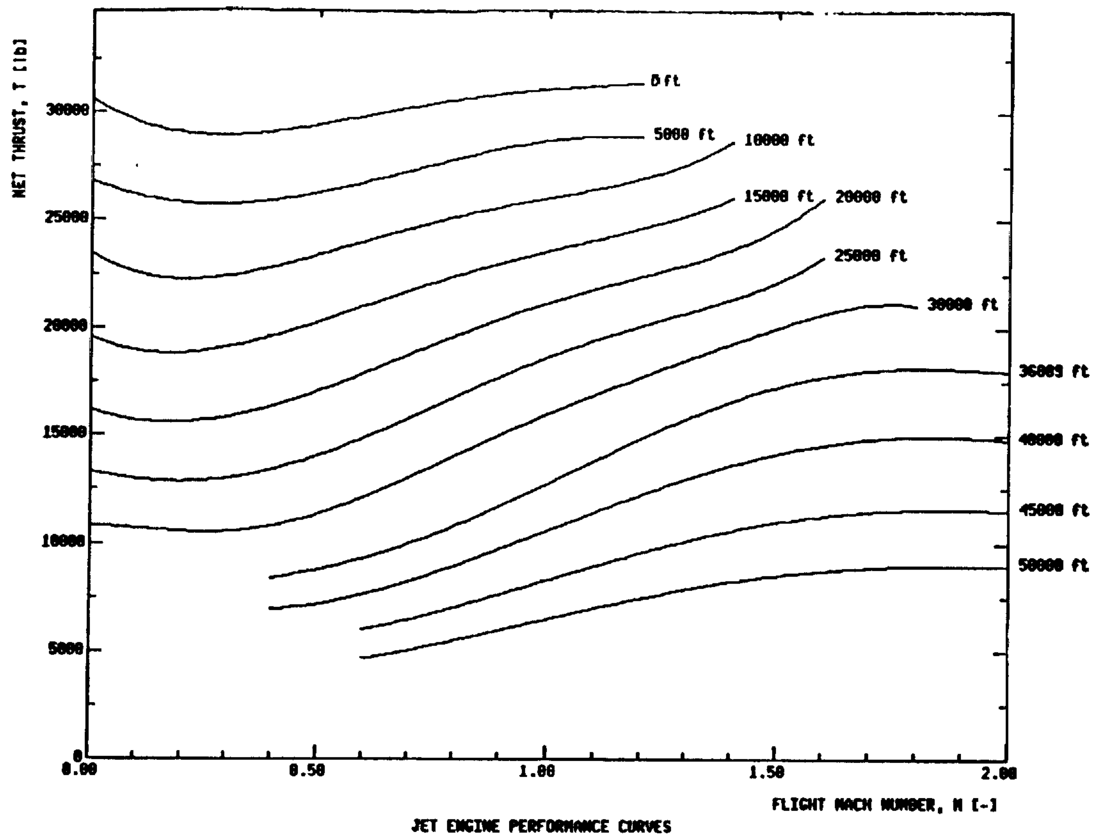

Figure 6.5b Maximum Unaugmented Thrust for the Monarch

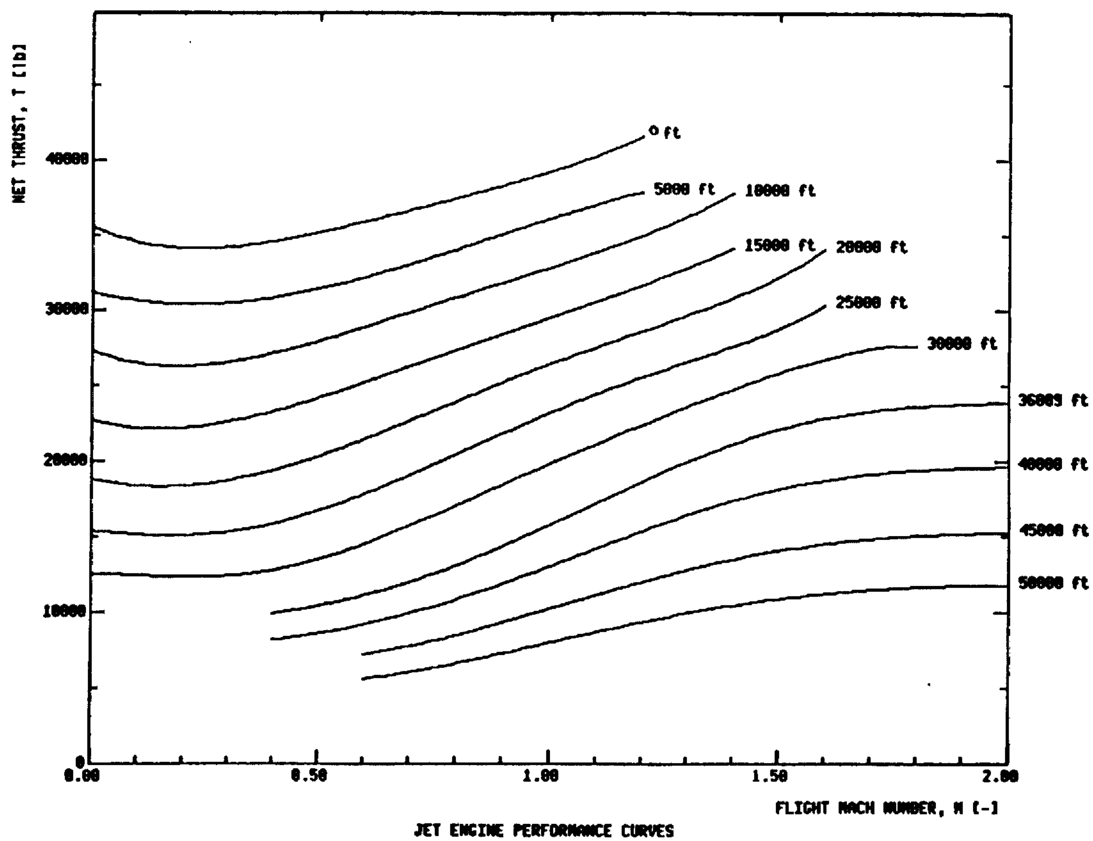

Figure 6.5c Maximum Augmented Thrust for the Monarch

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6.1.2 Air Induction System

The following list shows the design considerations used for the design of the inlet:

* supersonic operation,
* small inlet losses,
* high angle of attack operation,
* hot gas reingestion and FOD,
* avoiding system conflicts,
* and fuselage area ruling.

Since the maximum operating speed of the aircraft does not dictate the use of a variable geometry inlet, a normal shock inlet is used, and as discussed in Chapter 11, a bifurcated inlet is better than a chin inlet for alleviating hot gas reingestion. Therefore, a bifurcated normal shock inlet is used on the Monarch. According to the methods of Reference 6.3, and using the engine data from Reference 6.1, the total capture area of the bifurcated inlet is calculated to be 6.89 ft^2. This capture area size is based on the following assumptions.

* The inlet sizing point is the supersonic operation at M = 1.6 and 30,000 ft altitude.
* The current engine mass flow rate for the given flight condition is 319.64 lbm/sec.
* The ratio of secondary air flow to engine air flow (Ms/Me) is assumed to be 0.2 (Reference 6.3).
* The mass flow of the boundary layer bleed is 3 % of the inlet capture area (Reference 6.3).

A dimensioned front view of the inlet lip showing the capture area and shape is shown in Figure 6.6

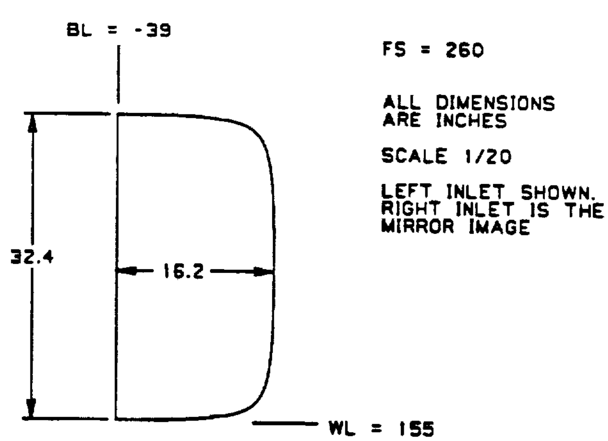


Figure 6.6 Cruise Inlet Lip Shape

The inlet lips are placed on each side of the fuselage as high as possible to avoid FOD and HGR (see Chapter 7). They are also placed behind the cockpit for improved pilot visibility. The exact layout of the inlet from the mouth to the compressor is designed to avoid conflicts with any systems while attempting to maintain the highest inlet efficiency. The inlet layout is shown in Figure 6.7.

A channel type boundary layer splitter is used. According to Reference 6.3, the width of the boundary layer at the inlet can be assumed to be 1% - 3% of the length of the fuselage ahead of the inlet. 2% is used for the Monarch which results in a width of 5 inches. Therefore the boundary layer splitter is placed 5 inches from the fuselage.

It is necessary to insure that the cruise engine has sufficient airflow at all times including low speed and hover flight conditions. The bifurcated inlet described must be designed for the supersonic flight conditions which means that it does not have sufficient capture area at the low speed conditions. Therefore, auxiliary inlets will be place on top of the main inlets and will operate only during the low speed flight conditions. According to Reference 6.3, the ideal inlet during hover is a bellmouth since there are no ram effects. Geometric constraints make this impossible, so it is assumed for preliminary design that a capture area of 1.15 times the compressor diameter is sufficient. The compressor area is 10.56 ft^2, which means that the total inlet capture area must be 1.15 times greater or 12.14 ft^2. As previously stated, the cruise inlet capture area is 6.89 ft^2. Therefore, the total auxiliary inlet capture area is 5.25 ft^2.

The location of the auxiliary inlets should be such that the air from them sufficiently mixes with the air from the main inlet openings before reaching the engine face. They should also be located such that the total air flow is accelerated to approximately Mach 0.5. The size, shape, and location of these inlets are shown in Figure 6.6. A permanent screen will be placed over the auxiliary inlets to prevent FOD. The losses due to this screen are considered negligible due to the low speed. Based on Reference 6.5, a set of horizontal louvers will be over the inlets to seal them during cruise flight and open during low speed flight. Louvers are viewed as being the easiest to mechanically operate plus they should act as flow turning veins when opened. Since they will only be operated during very low speed, it is not believed that the most forward louver will block the flow into the aft louvers. An electromechanical actuator will be used to operate the louvers, and will be place in the inlet boundary layer splitter. Figure 6.7 also shows the location of the auxiliary inlets as well as a schematic of the actuation.

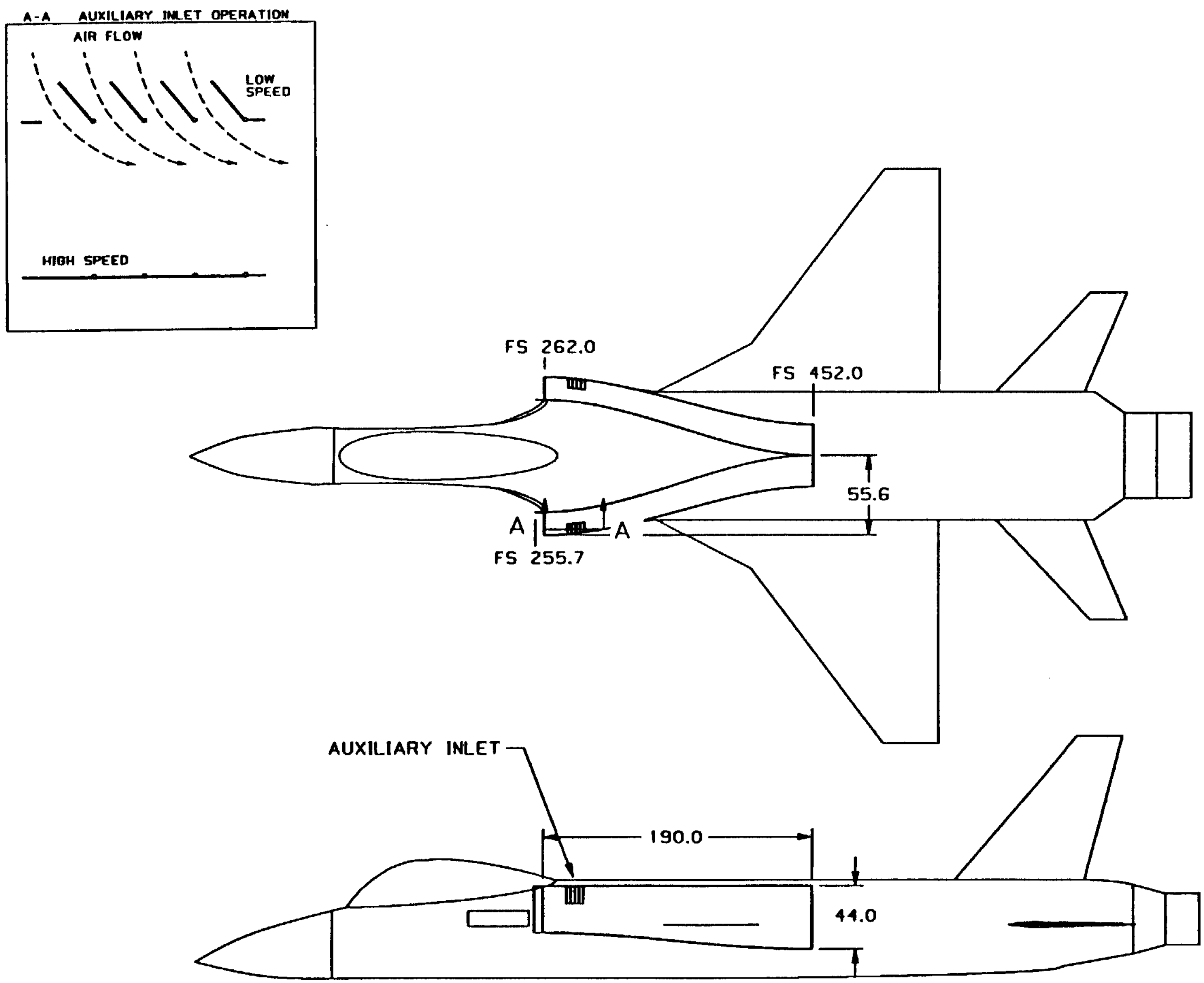


Figure 6.7 Cruise Inlet for the Monarch

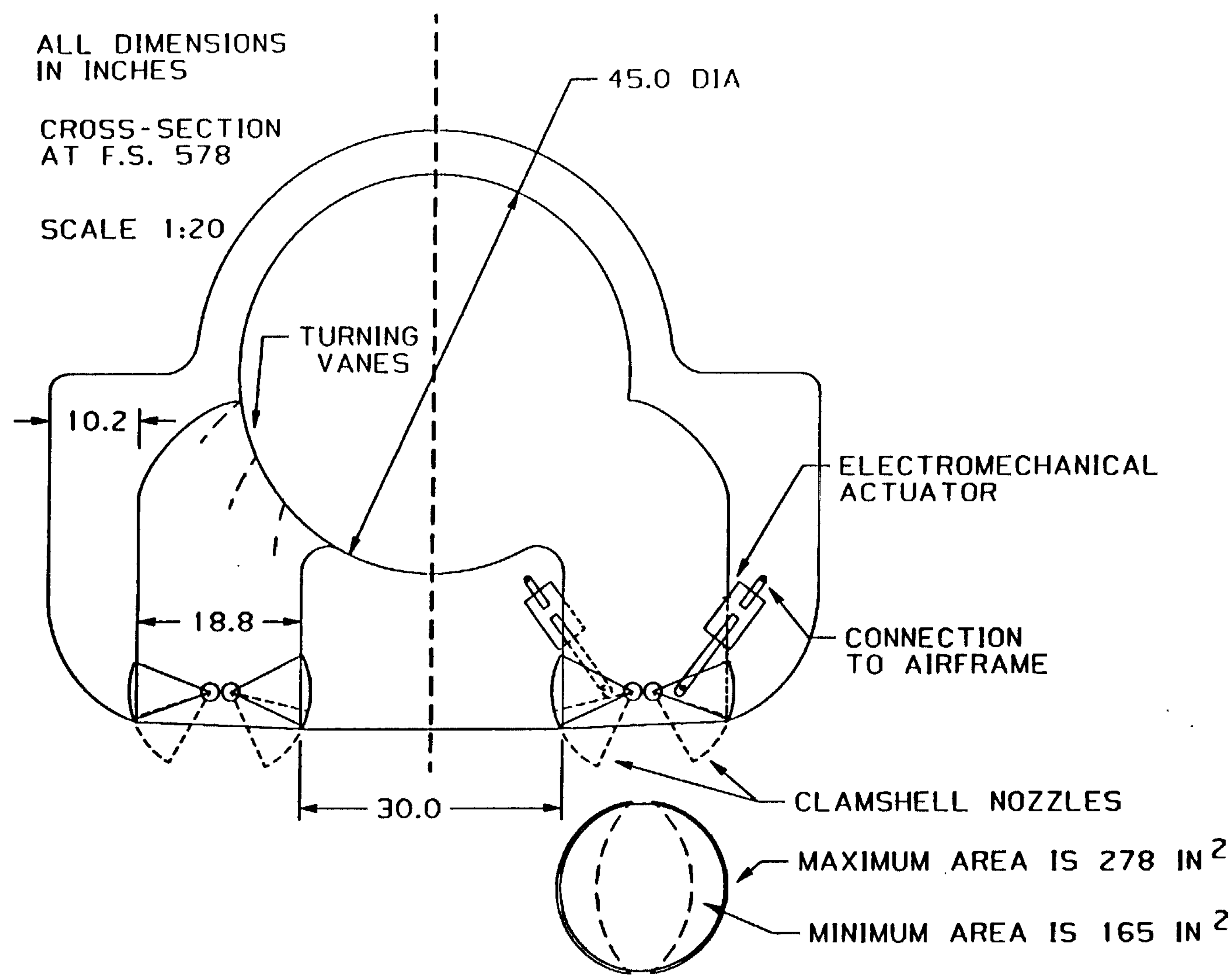
6.1.3 Cruise Engine Nozzles

The cruise engine has two types of nozzles. One nozzle is the rear nozzle providing thrust vectoring which is shown in Figure 6.11. The other type of nozzle is a pair of ventral nozzles providing hover capabilities which are shown in Figure 6.8.

The design driver for the ventral nozzles was that the nozzles must have variable area capabilities along with thrust vectoring of 15 degrees about the x-axis to allow for translation. The ventral nozzles were sized by getting the throat area from Reference 6.1 and converting it to an equivalent area for each ventral nozzle. Therefore, the ventral nozzles will have the required throat area to keep the flow "choked" as the rear nozzle blocks the airflow.

The ventral nozzles are shown in Figure 6.8. The clamshell nozzle is a low weight, a low complexity, and a variable area nozzle. The other nozzle considered was one with turning vanes. The primary problem with the turning vane nozzle is that the flow must be vectored to reduce the throat area. This is not acceptable for the ventral nozzles because the nozzles will be used for roll control by differential thrust of the two ventral nozzles, which require variable area capabilities without loss of thrust along the z-axis. The clamshell nozzles will be retracted for up and away flight. Fuselage doors will be used to reduce drag that would be caused by the exposed ventral nozzles.

The turning vanes, as shown in Figure 6.8, help to alleviate pressure losses when turning the flow 90 degrees. The sizing of the duct was calculated assuming 5 percent loss in pressure in the duct. The ventral nozzle ducts must be detachable from the main engine so that expedient engine removal is possible.



<u>Figure 6.8 Ventral Nozzle for the Monarch</u>

The rear nozzle must provide pitch and yaw vectoring capabilities as discussed in Chapter 4. The primary driver for a pitch and yaw vectoring nozzle is to provide enhanced maneuvering capabilities and allow for removal of the rudder. The following rear nozzle designs were looked at:

1) 2D convergent-divergent nozzle with 20 degree pitch and 15 degree yaw capabilities. The nozzle could also block and turn the flow, which is required for hover. The primary drawback to this nozzle is the complexity. The secondary drawback is that to produce a side force the nozzle had to be spoiled (similar to thrust reversing, but not as extreme), causing large losses in axial thrust. This nozzle is shown in Figure 6.9.

2) An axisymmetric nozzle with 20 degree pitch vectoring and block and turn capabilities. This nozzle would have been used if nozzle #3 did not produce the side-force required to remove the rudder. The reason this nozzle would have been used is because of its low weight relative to a 2D nozzle. This nozzle is shown in Figure 6.10. Because nozzle #3 provides the required side-forces this nozzle was excluded.

3) 2D convergent-divergent nozzle with 20 degree pitch and 25 degree yaw capabilities. This nozzle can also block and turn the flow, which is required for hover. The reason that this nozzle is better than nozzle #1 is that the yaw vectoring occurs after the nozzle. Therefore, the axial thrust loss is reduced. The drawback to this nozzle is its size and weight are larger than nozzle #2. Nozzles #3 and #1 are similar in size and weight. This nozzle produces enough side-force to eliminate the rudder as discussed in Chapter 10. This nozzle is shown in Figure 6.11.

Nozzle #3 was chosen for the Monarch because of the capability to remove the rudder as discussed in Chapter 9. The primary drawback to nozzle #3 is that the weight is 20% greater than the other nozzle options. Nozzle #3 is shown in Figure 6.11.

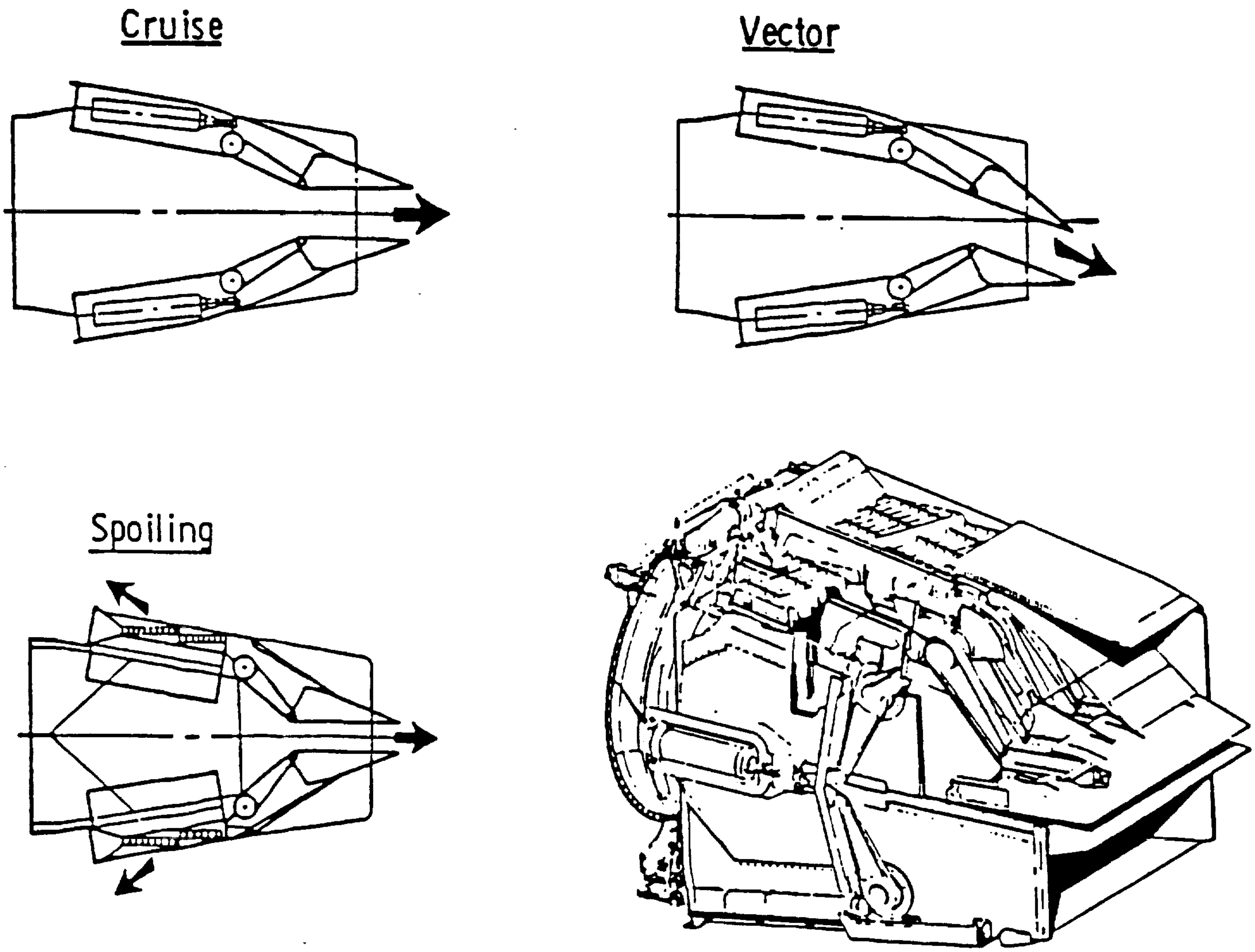


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Figure 6.9 2-Dimensional Convergent-Divergent Nozzle

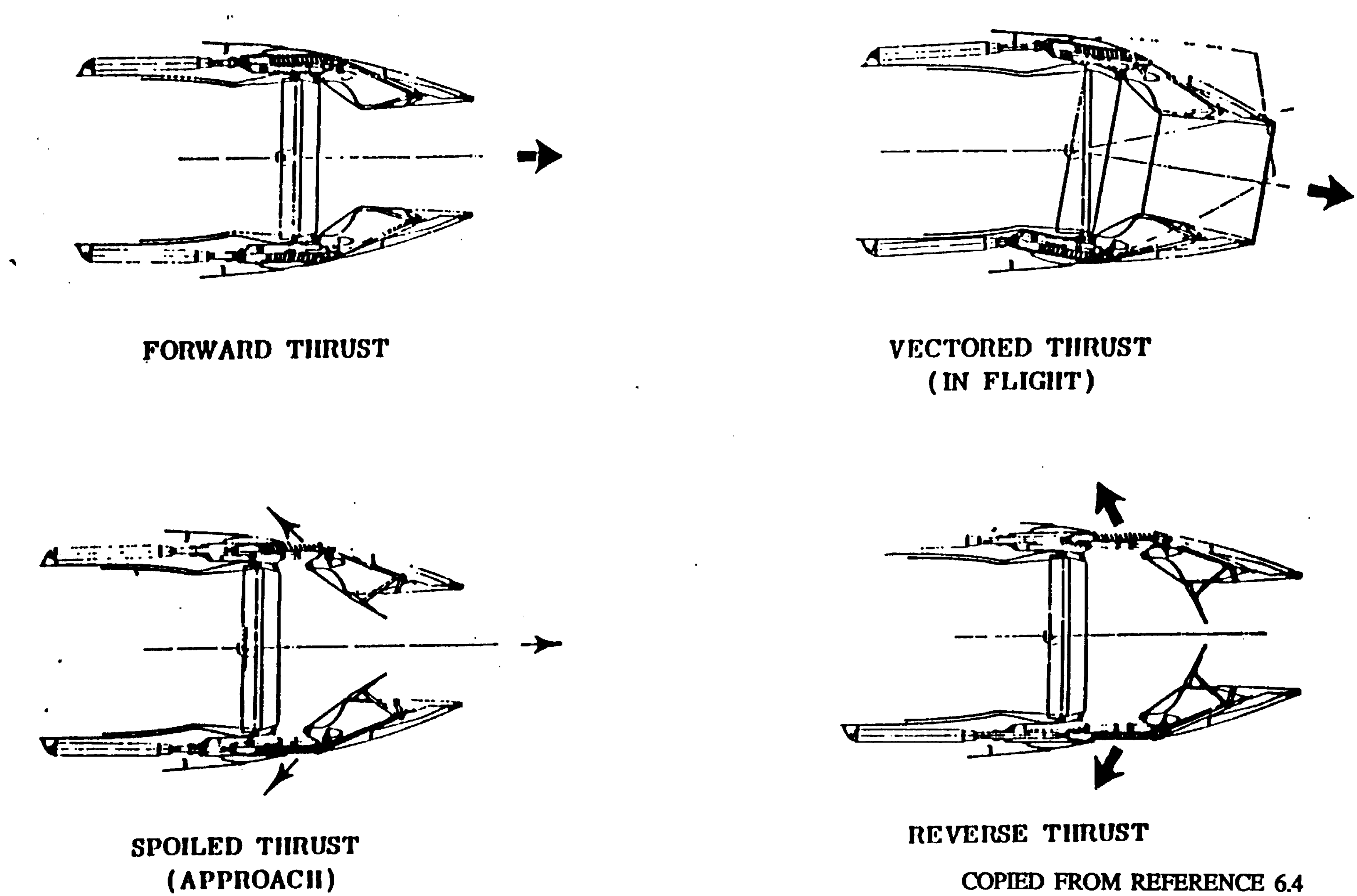

Figure 6.10 Axisymetric Nozzle With Pitch Vectoring (copied from Reference 6.4)

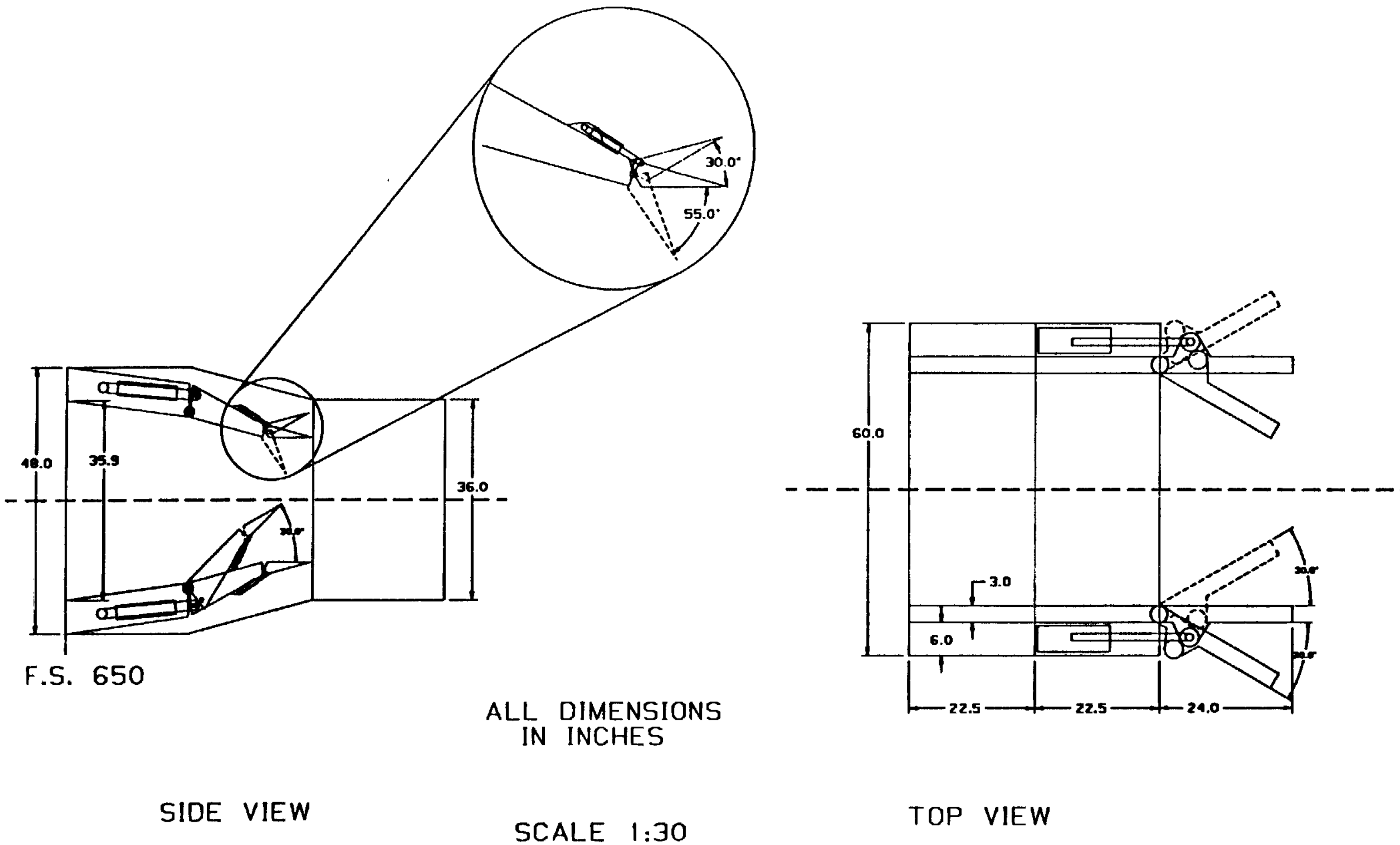

Figure 6.11 Rear Nozzle for the Monarch

6.2 LIFT ENGINE

The lift engine used for the Monarch is based on the Rolls-Royce direct lift engine. The engine parameters for this engine are taken from Reference 6.5. It is an unmixed turbofan designed to provide vertical thrust for a STOVL aircraft. The technology standard assumed for this design is consistent for an initial operational capability of 2005.

According to Reference 6.5 the Rolls-Royce engine was designed for vertical mounting and includes a vectoring exhaust nozzle. The engine has a large amount of parts made with advanced composites, which enables the uninstalled thrust to weight ratio to reach 28. To achieve the lightest possible solution while maintaining acceptable jet exhaust conditions, a relatively high bypass ratio is implemented. A higher bypass ratio results in a higher engine volume. A smaller diameter engine with a higher specific thrust could be used to decrease the required engine volume. However, this will lead to an increase in engine weight and/or more severe exhaust conditions.

6.2.1 Engine Description and Performance

The size required for the lift engine was determined solely by the thrust requirements of hover. As mentioned in subsection 6.1.1, the total thrust required during hover is 1.23 times greater than the hover weight of the aircraft. The amount of thrust from the lift engine was determined by balancing the thrust from both engines about the center of gravity of the aircraft. The thrust balance at hover for the Monarch is shown in Figure 6.12. The original Rolls-Royce engine was resized using the same scaling laws that were used for the cruise engine. The resized engine parameters are listed in Table 6.2.

Table 6.2: Engine Parameters

Rating	Max Dry Thrust
Condition	SLS 90°F day
Mass Airflow	266.37 lbm/sec
Nozzle Throat Area	3.431 ft^2
Bypass Ratio	1.5
Nozzle Pressure Ratio	3.268
Maximum Installed Thrust	12,105 lbs
Diameter	32.8 in.
Length	35.1 in.
Weight	480 lbs

6.2.2 Engine Air Induction System

The lift engine inlet is positioned at fuselage station 230. Due to the close proximity of the engine to the cockpit, a bifurcated inlet is used. Since this engine is only used during hover and transition, the total inlet capture area is assumed to be 1.15 times greater than the compressor area or 6.83 ft^2. Louvers will also be used to seal the inlet during wing-borne flight, and will operate similarly to the auxiliary inlets.

Figure 6.13 is a cross-sectional view of the lift engine and inlet including the louvers.

Counter Air Mission:

Takeoff Weight = 31,336 lbs
Hover Weight = 24,423 lbs
Hover Thrust = 31,750 lbs

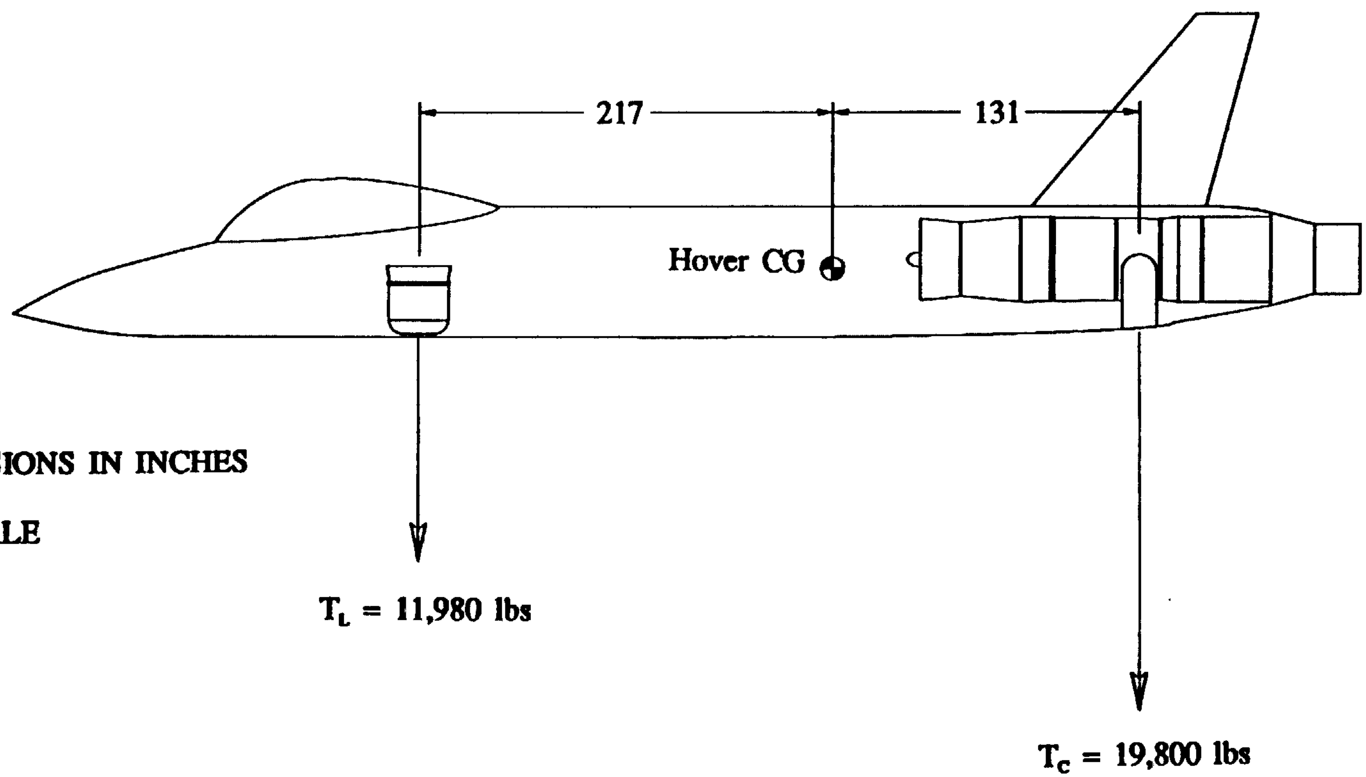

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Figure 6.12 Thrust Balance at Hover for the Monarch

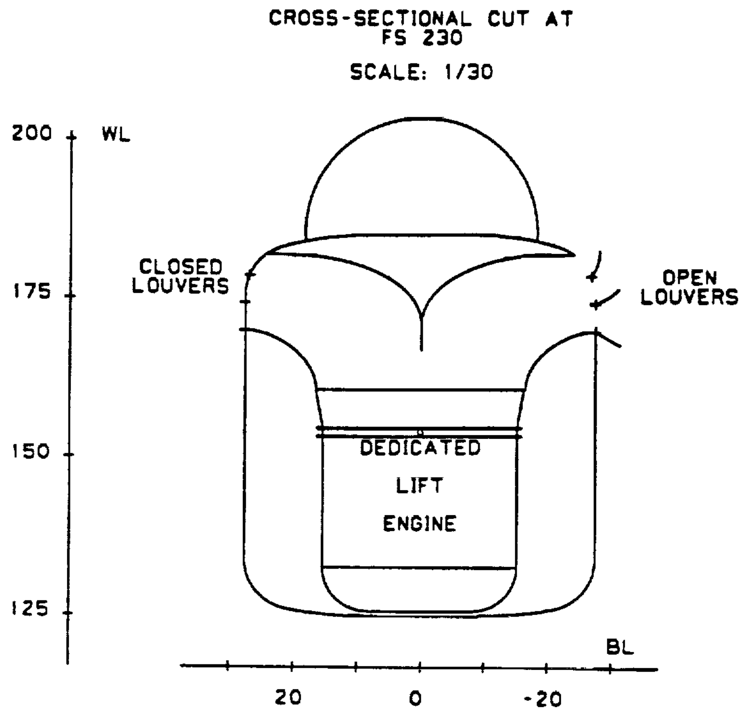

Figure 6.13 Cross-section View of Engine Inlet for the Monarch

6.2.3 Engine Nozzle

The lift engine nozzle was designed so that the thrust could be vectored 20 degrees forward and aft to allow for pitch control. The nozzle was designed so that during up and away flight the thrust vectoring vanes will close. Therefore, no fuselage doors are needed. This nozzle design is shown in Figure 6.13. The lift engine nozzle vectoring vanes are powered by two electromechanical, jack-screw, actuators. The vectoring may allow the lift engine to enhance the pitch control of the aircraft during hover and transition.

A gimballing nozzle, similar to that used for a rocket, was considered for the lift engine. The problem with the gimballing nozzle is greater complexity than the vectoring nozzle and also the need for a fuselage door. Another nozzle considered for the lift engine was a clamshell nozzle, similar to the ones on the ventral nozzles. The primary drawback of the clamshell nozzle is that the clamshell nozzles occupy more volume than the thrust vectoring vanes. The reason the clamshell design is used for the ventral nozzles is that the ventral nozzles are required to be variable area nozzles.

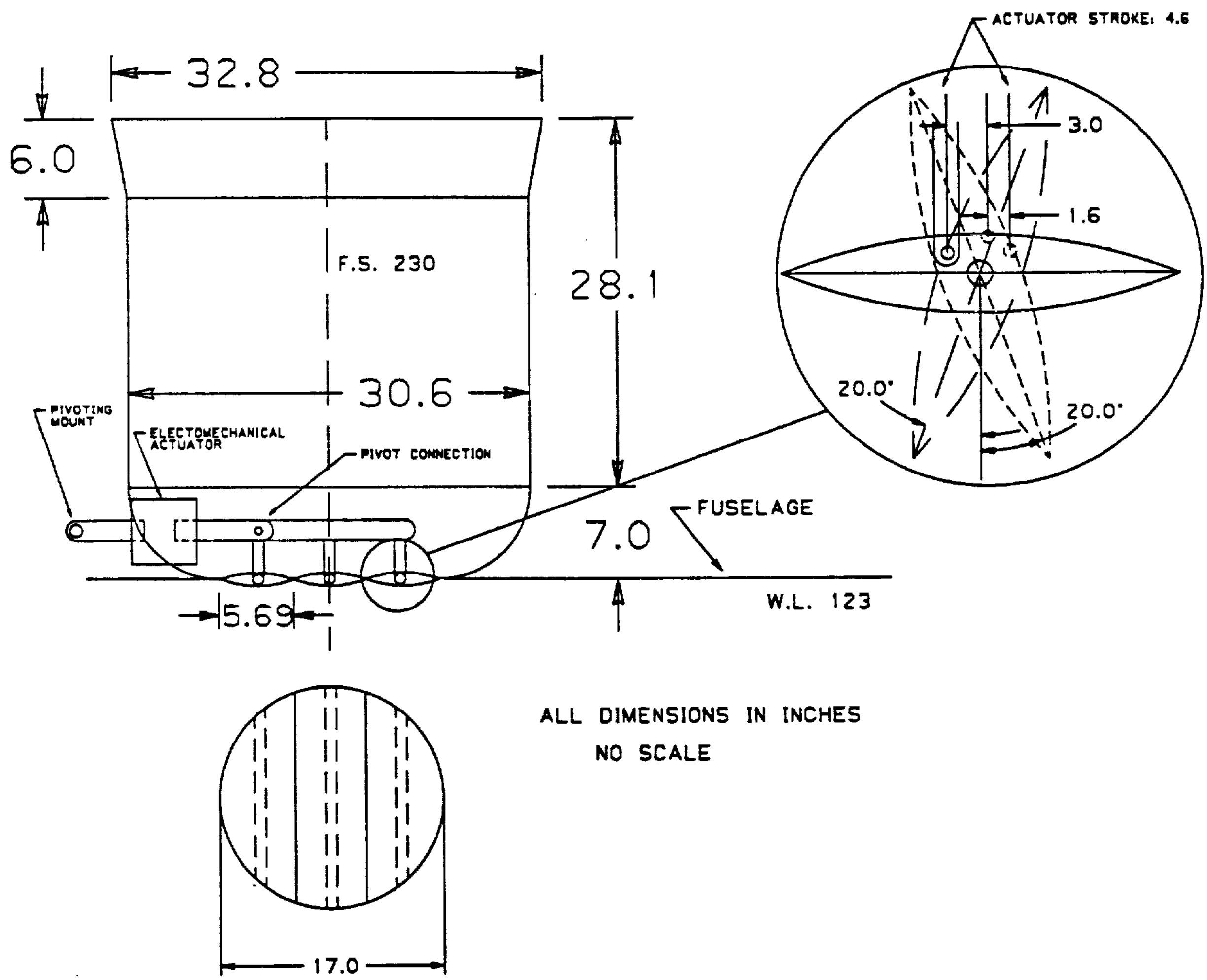

Figure 6.14 Lift Engine Nozzle for the Monarch

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7. TAKEOFF, HOVER AND TRANSITION ANALYSIS

The purpose of this chapter is to analyze some of the unique features of STOVL aircraft during operation below the velocity for wing-borne flight. The following topics are covered.

Section 7.1 TAKEOFF GROUND ROLL DETERMINATION
Section 7.2 TRANSITION ANALYSIS
Section 7.3 HOVER ANALYSIS
Section 7.4 PILOT WORKLOAD ANALYSIS

7.1 TAKEOFF GROUND ROLL DETERMINATION

The following step-by-step procedure is used by the Monarch for short takeoffs.

Step 1 With the airplane at the beginning of the runway, and the brakes on, the auxiliary inlets and the lift engine inlets are opened, the leading edge flap is deflected 20° and the trailing edge flap is deflected 40°. Then both engines are started.

Step 2 While keeping the brakes on, the Cruise engine is throttled up to maximum dry thrust and only the main nozzles is used. However, it is deflected 20° downward to balance out the moment created by the idling Lift engine.

Step 3 The brakes are released, the aircraft begins to move, and the Lift engine is throttled up

Step 4 When the airplane has accelerated enough that the wing provides sufficient lift for the wing and engines to lift the airplane, 16,312 lbs of thrust is diverted to the ventral nozzles of the Cruise engine. This thrust combined with the 11,514 lbs of thrust provided by the Lift engine and the wing will lift the airplane into the air.

Figure 7.1 shows the thrust vectors produced by the engines at critical stages of the takeoff as well as the equivalent thrust. The times and distances shown are for the Counter Air Mission. It should be noted that at all points during the takeoff the total thrust is balanced independently of the aerodynamic forces on the aircraft. Also, the aircraft has 2° of ground incidence but it does not rotate to takeoff. This was not desired since the ventral nozzles cannot be deflected aft; therefore, they would produce a component of drag.

As seen in Figure 7.1, following this procedure using precise thrust angles and magnitudes, the Counter Air Mission takeoff ground roll was determined to be 238 ft. This distance is determined using lift and pitching moments in ground effects and a friction coefficient of 0.2. The Lift engine is operating at full capacity at the point of takeoff and the Cruise engine is operating at maximum dry power with enough thrust vectored through the ventral nozzles to balance the thrust from the Lift engine. The remaining thrust is ducted through the main nozzle to accelerate the aircraft horizontally. Figure 7.2 shows a plot of takeoff ground roll distance as a function of the aircraft takeoff weight for all of

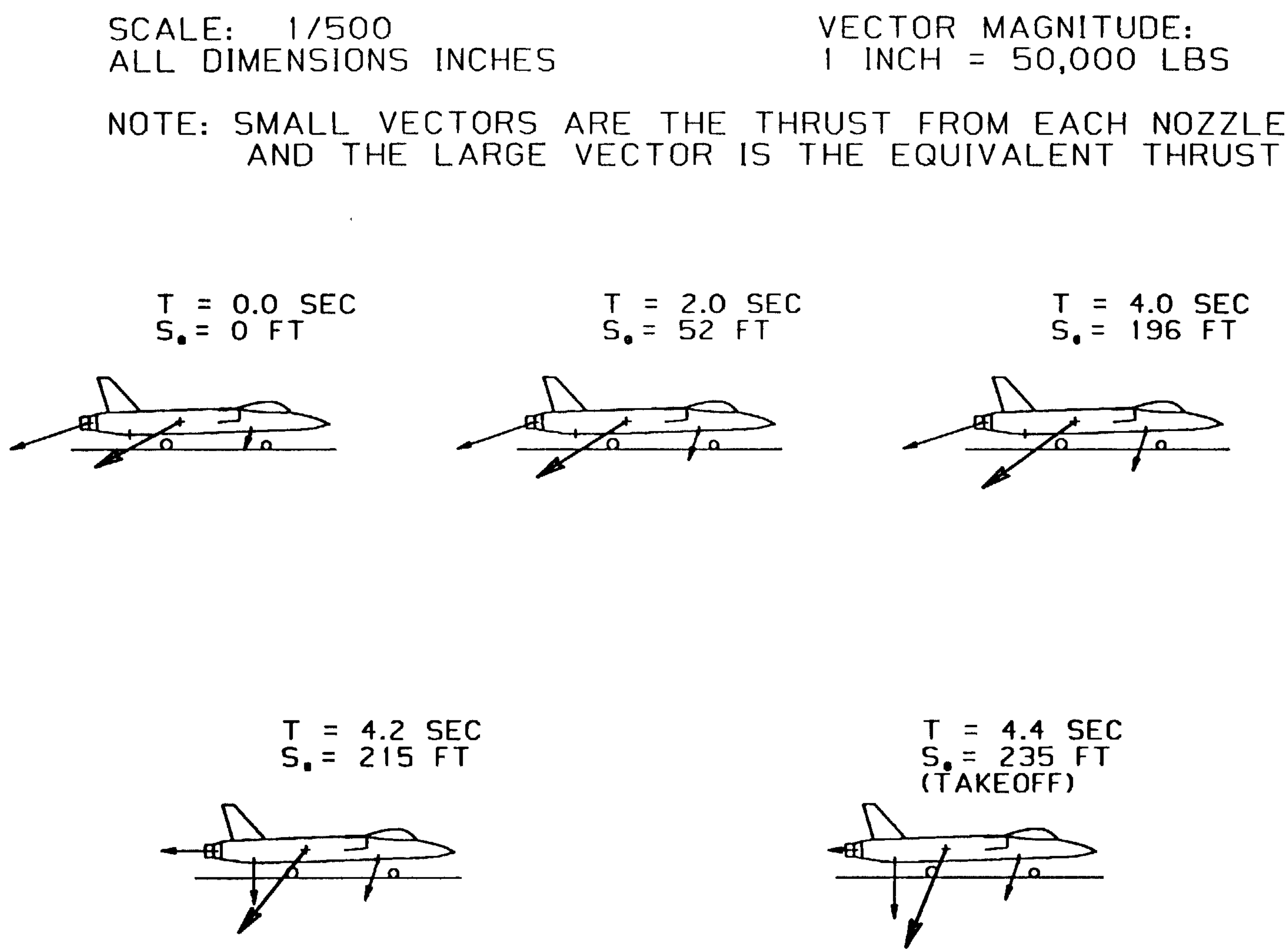

Figure 7.1 Takeoff Schematic for the Monarch

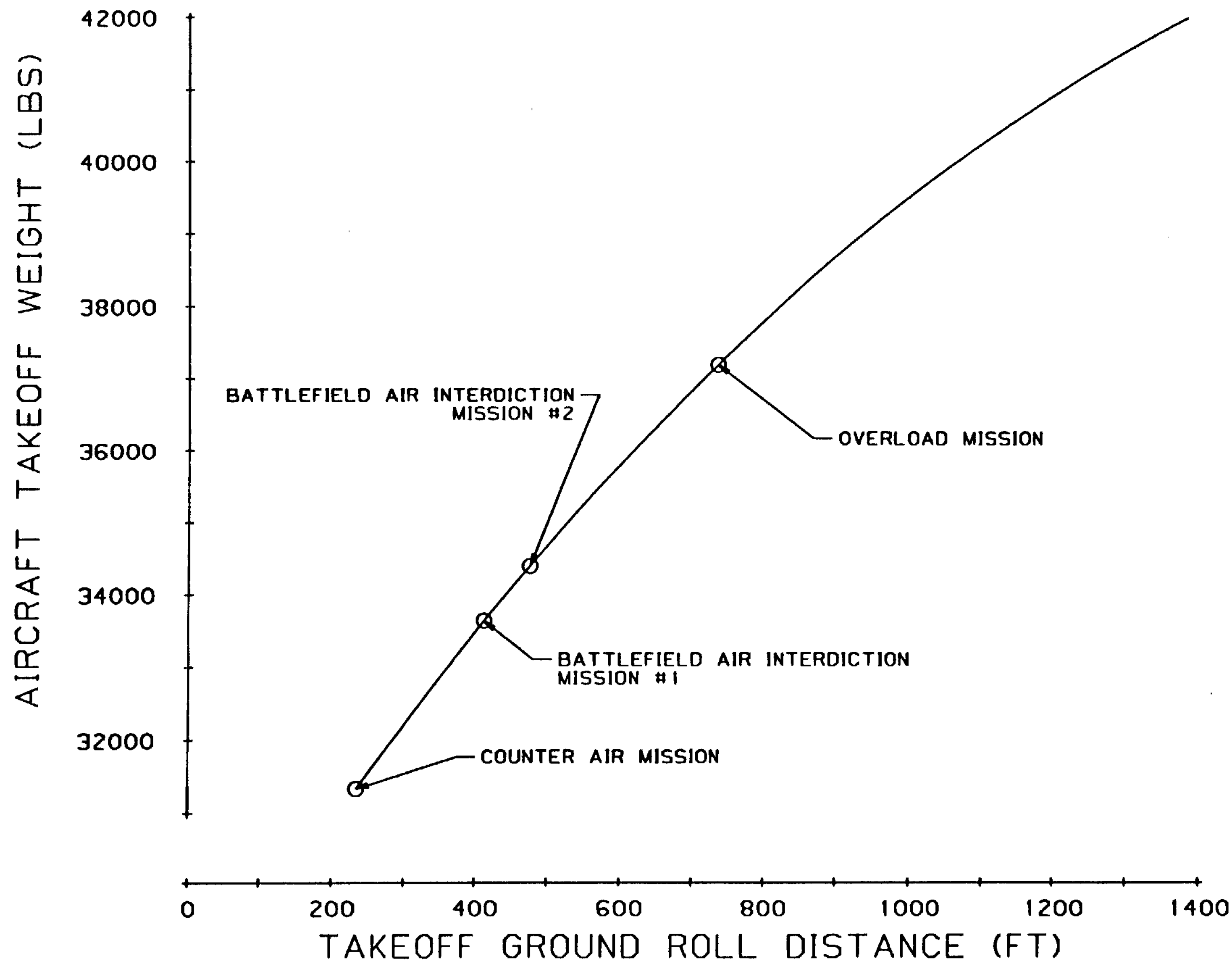

Figure 7.2 Takeoff Ground Roll for Various Weights of the Monarch

the various missions including the overload mission.

7.2 TRANSITION ANALYSIS

The transition from the point of takeoff to purely wing-borne flight begins with the engines left at the same operating condition as takeoff which is with maximum vertical thrust. This configuration is held until the aircraft reaches a desired height. When vertical acceleration is not desired, the lift engine is throttled down and the Cruise engine starts to slowly transfer more thrust from the ventral nozzles to the main nozzle keeping the total thrust balanced about the aircraft center of gravity. The rate at which this occurs is such that the airplane remains level because the decrease in vertical thrust can be made to equal the increase in wing lift providing no vertical acceleration. This process is continued until the Lift engine reaches its minimum throttle setting. At this point the lift engine must be shut down. During the spool-down of the Lift engine, it will still be providing some thrust but it will not be exactly the desired amount so the pilot will accelerate in the vertical direction or he will have to rotate the aircraft to a different angle of attack to alter the wing lift to compensate for the change in vertical engine thrust. This process is shown schematically in Figure 7.3. The numbers shown are for the counter air mission, and the pilot has chosen to level off at 100 ft. altitude.

The transition from wing-borne flight to hover follows nearly the same procedure only in the opposite direction. The aircraft is brought in at a given altitude and at approach velocity. The lift engine is started and the rear engine begins to transfer a portion of the flow to the ventral nozzles. If this maneuver is done at a high angle of attack the ventral nozzles will produce drag which will significantly slow the airplane. The lift engine begins to throttle up and the Cruise engine continues to transfer more flow to the ventral nozzles to balance the force from the Lift engine. The vertical acceleration is controlled by the pilot but it is desired to keep the aircraft high enough above the ground that HGR, suckdown, and ground erosion are avoided. Once the aircraft is positioned directly above the landing site, a constant vertical acceleration of approximately 3 ft/sec is established until the aircraft touches the ground. Then the Lift engine is immediately shut down and the Cruise engine is either shut down or the ventral nozzles are closed sending all of the thrust through the main nozzle for ground taxiing. This is done to reduce the amount of ground erosion. A schematic of this is shown in Figure 7.4 for the Counter Air Mission with the transition beginning at 100 ft. and the final descent to landing beginning at 50 ft.

The flight control system will need alterations due to the required control over the thrust vectoring. The flight control systems that are changed for this report are: pitch and altitude hold, the bank angle control. The dynamic pressure is required for the flight control system so that the automatic flight control system can determine whether to use the aerodynamic controls of the thrust vectoring. In the block diagrams, shown in Figures 7.5 through 7.7, the "yes" by the dynamic pressure block means that the dynamic pressure is high enough to use aerodynamic controls. If the dynamic pressure is not high enough for the aerodynamic controls, thrust vectoring and the RCS will be used. The symbol δ_T refers to the nozzle and throttle actuation. The reason for this is that if the nozzle deflections are

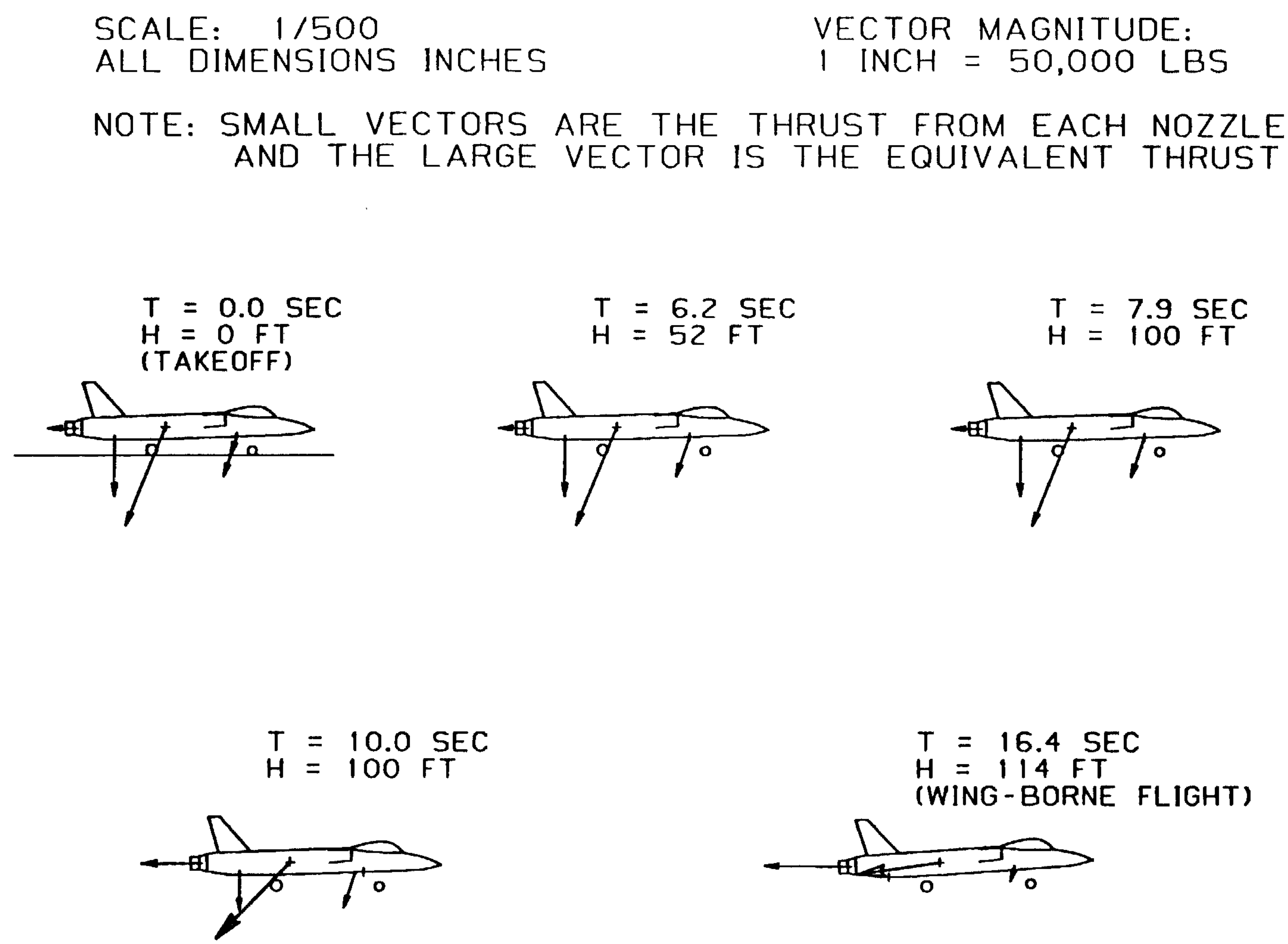

Figure 7.3 Hover to Wing-borne Flight Transition Schematic for the Monarch

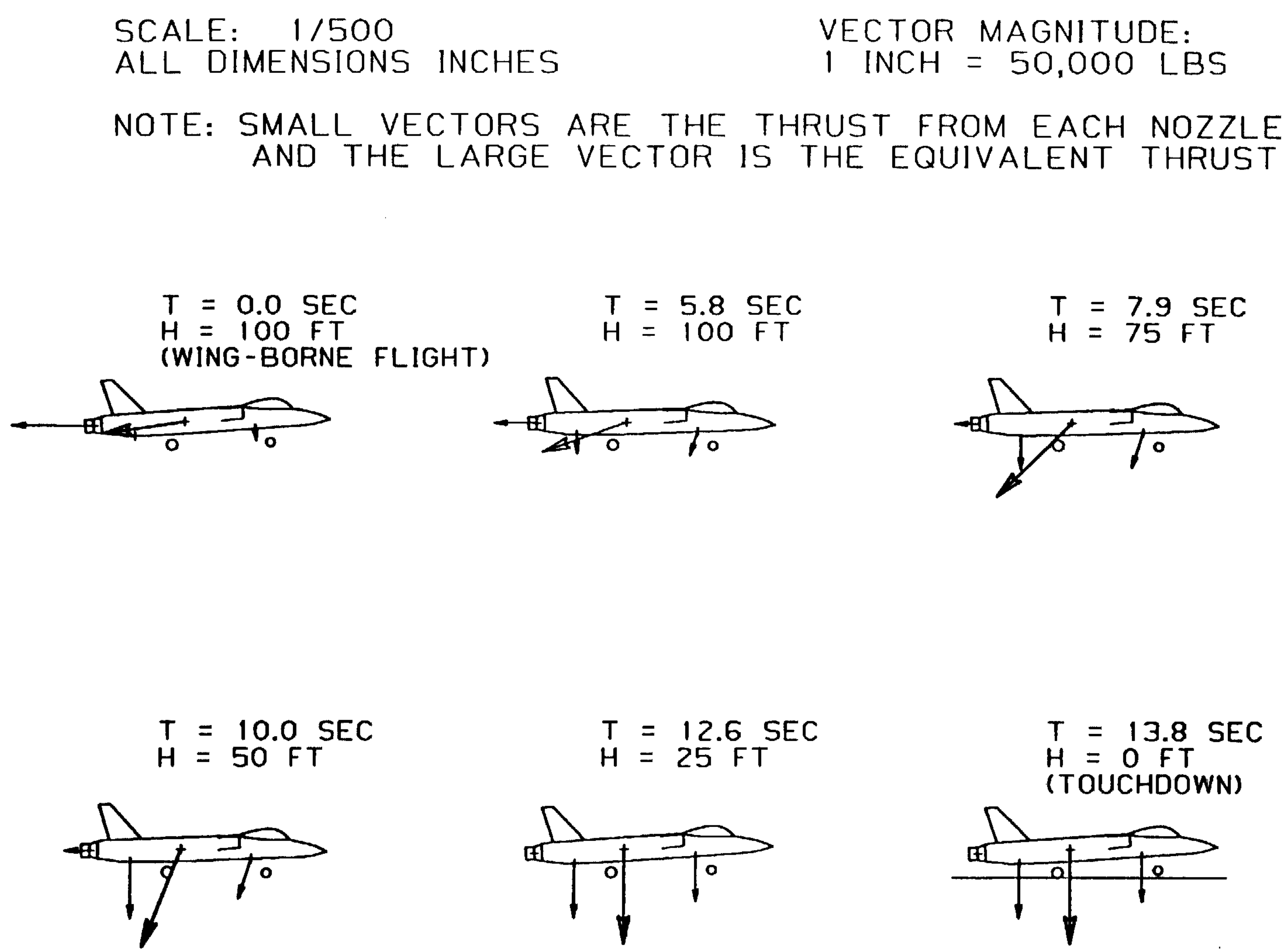

Figure 7.4 Wing-Borne Flight to Hover Transition Schematic for the Monarch

changed the throttle setting may need to be increased or decreased balance the moments and forces created by the thrust vectoring. Figure 7.5 shows the block diagram for the pitch attitude hold with inner loop pitch damping. Figure 7.6 shows the bank angle control system block diagram. Figure 7.7 shows the altitude control system block diagram for the Monarch.

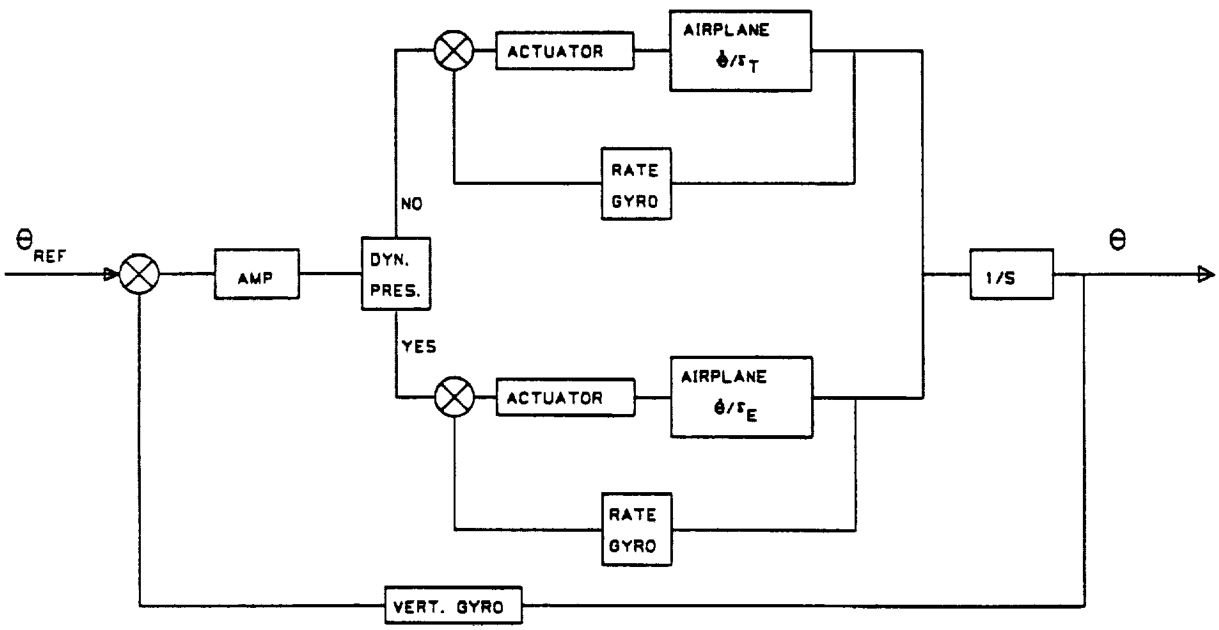

Figure 7.5 Pitch Attitude Hold AFCS for the Monarch

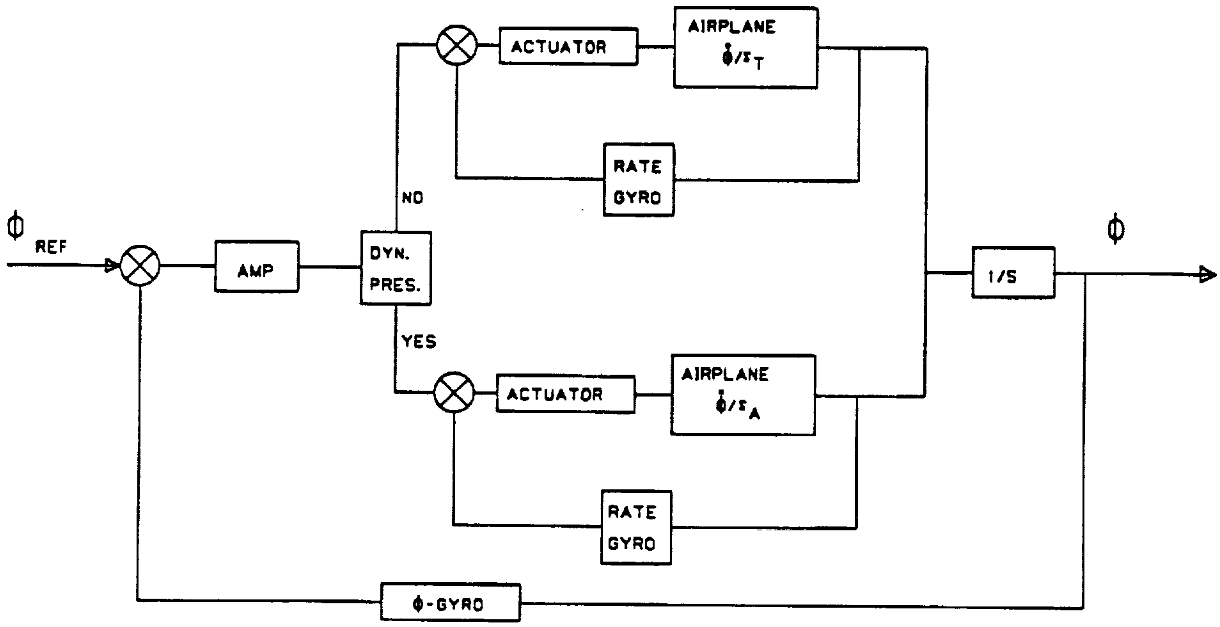

Figure 7.6 Bank Angle AFCS for the Monarch

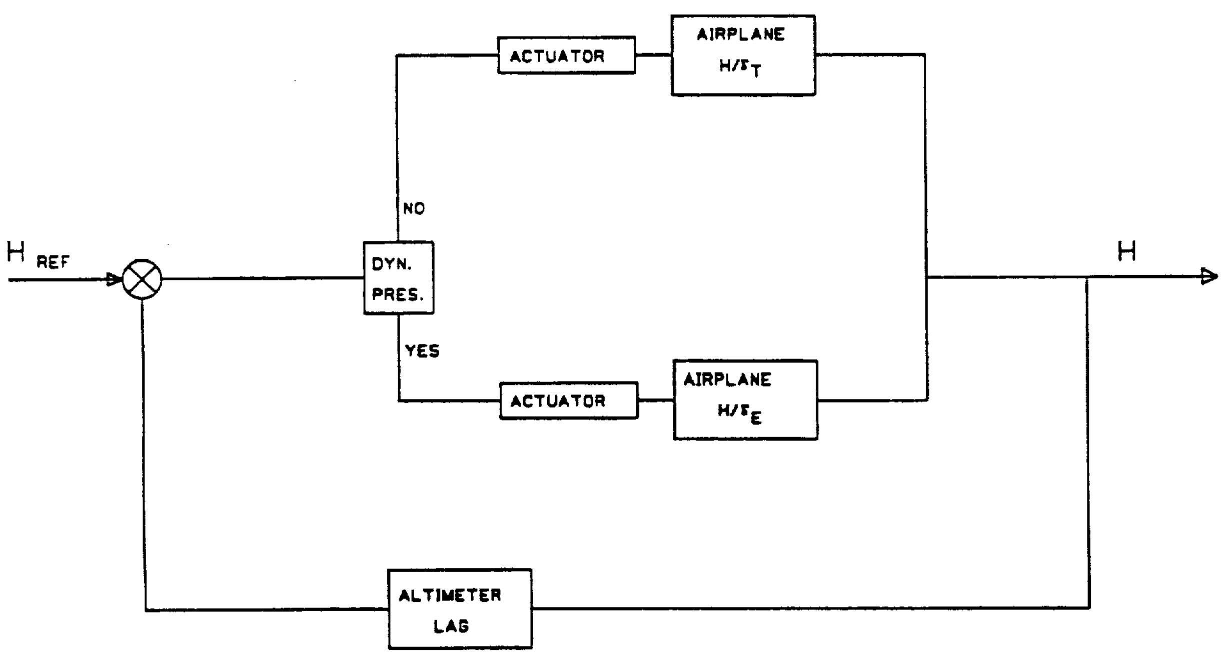

Figure 7.7 Altitude Hold AFCS for the Monarch

7.3 HOVER ANALYSIS

In this section the reaction control system, suckdown predictions, and hot gas reingestion are discussed for the Monarch.

7.3.1 Reaction Control System

The reaction control system maintains control about the aircraft axes in STOVL modes. It also assists the conventional control in transitional flight. Hot air is bled from the compressor of the engine and is fed to a butterfly valve which controls the flow to the four valve outlets. The butterfly valve is operated by an electromechanical actuator which is activated when the aircraft is at approach speed and below. The ducting is made from rolled and welded nickel-chromium alloy. The duct diameter was sized using Reference 7.1, the diameter varies from 4.5 inches in the fuselage to 3.5 inches at the reaction valves.

The amount of bleed air required from the engine is 2.0% mass flow of the cruise engine, this value was calculated using Reference 7.1. To calculate the amount of bleed air required for the RCS Reference 7.2 was used to find the control authority required in hover. The pitch and yaw control required for Level 1 flying qualities in hover is .3 rad/s^2 and .5 rad/s^2 respectively. The pitch and yaw control required for Level 1 flying qualities in transition is .2 rad/s^2 and .25 rad/s^2 respectively. The angular accelerations are converted into thrust by the following equation:

$$\text{Treq} = (\text{I x psi double dot})/\ 1 \qquad (7.1)$$

With (I) being the airplane moment of inertia about the z-axis, (l) being the distance from the reaction control valve to the z-axis, and psi double dot being the yaw control required. From the thrust it is possible to calculate the required mass flow using Reference 7.1.

The amount of bleed air for the pitch control is 1.2% mass flow of the cruise engine, and for yaw control 0.8% mass flow of the cruise engine, therefore, the total RCS bleed is 2.0%. The maximum temperature and pressure at the valves are approximately 1350 R. and 236 psi.

Roll control is provided by using variable area ventral nozzles which generate the required roll control authority for Level 1 flying conditions. The yaw control will be provided using two reaction control valves at the aft section of the fuselage. This is shown in Figure 7.8. The pitch control will be provided by using two reaction control valves at the aft section and the forward section of the fuselage. The aft section of the pitch reaction control valve is shown in Figure 7.8. The overall layout of the RCS system for the Monarch is shown in Figure 7.9.

The roll, pitch, and yaw reaction control system will be controlled by pilot-stick movement or hover SAS. The hover SAS will allow the plane to remain stable throughout hover and transition.

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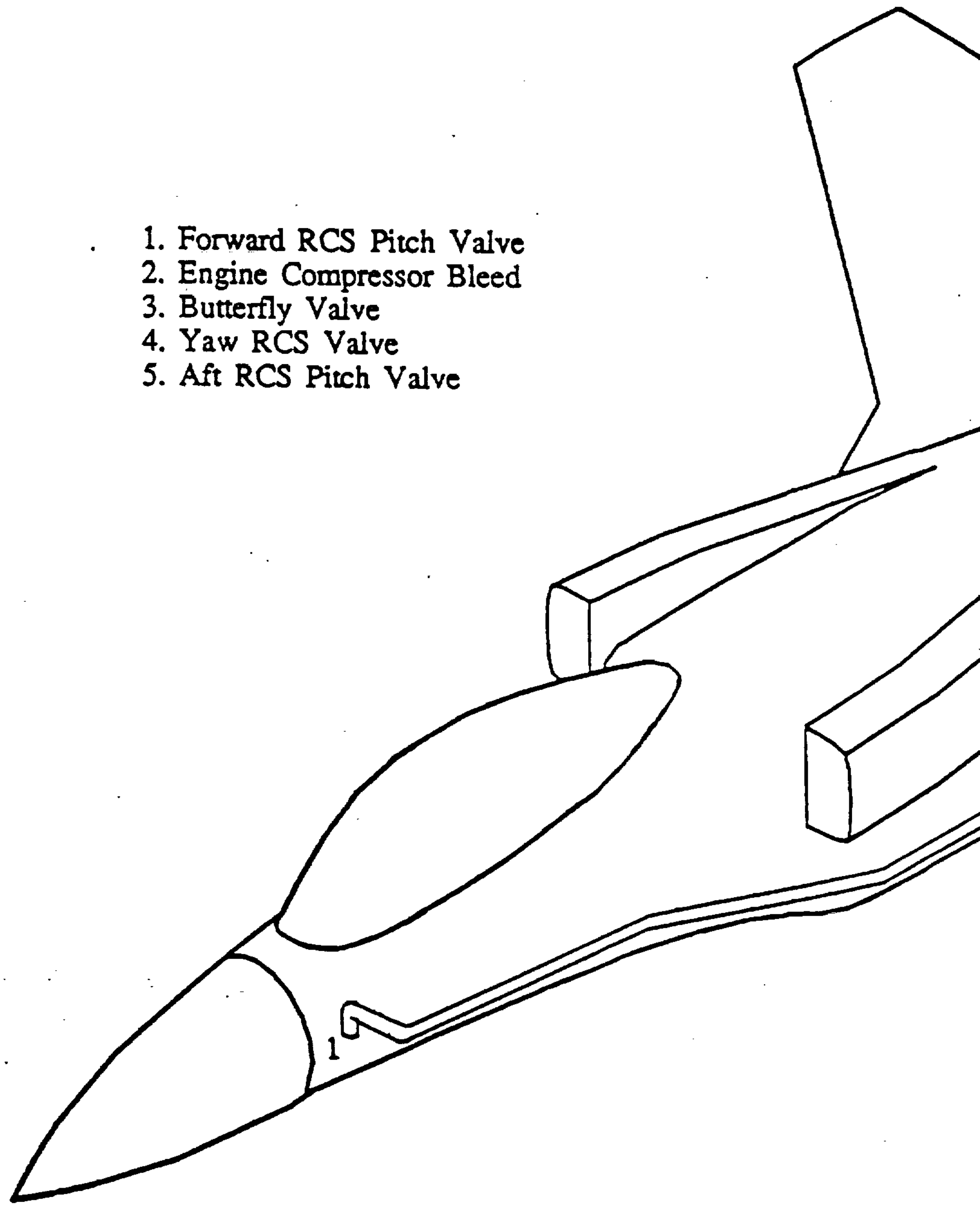

Figure 7.8 RCS syst

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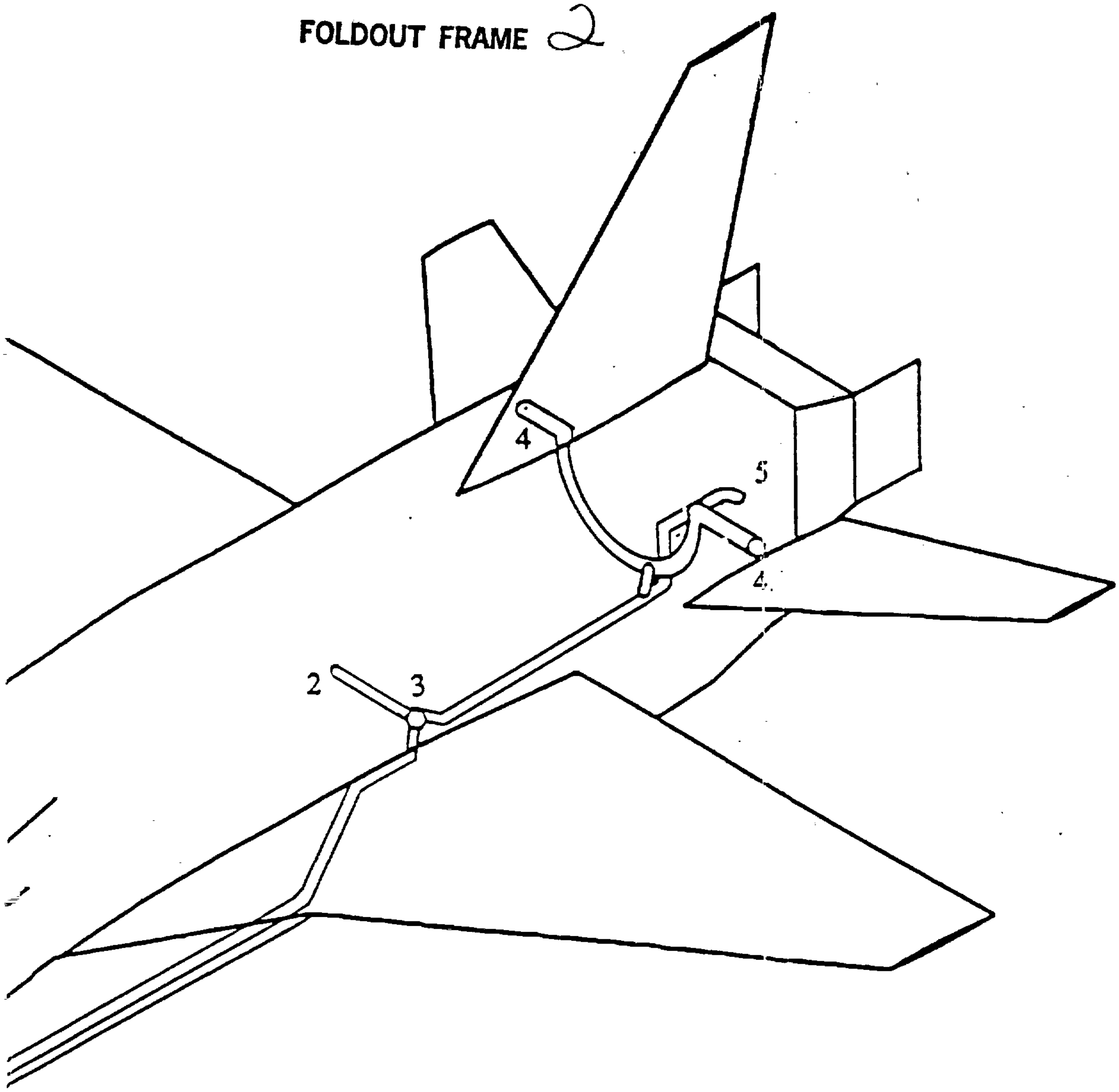

n layout for the Monarch

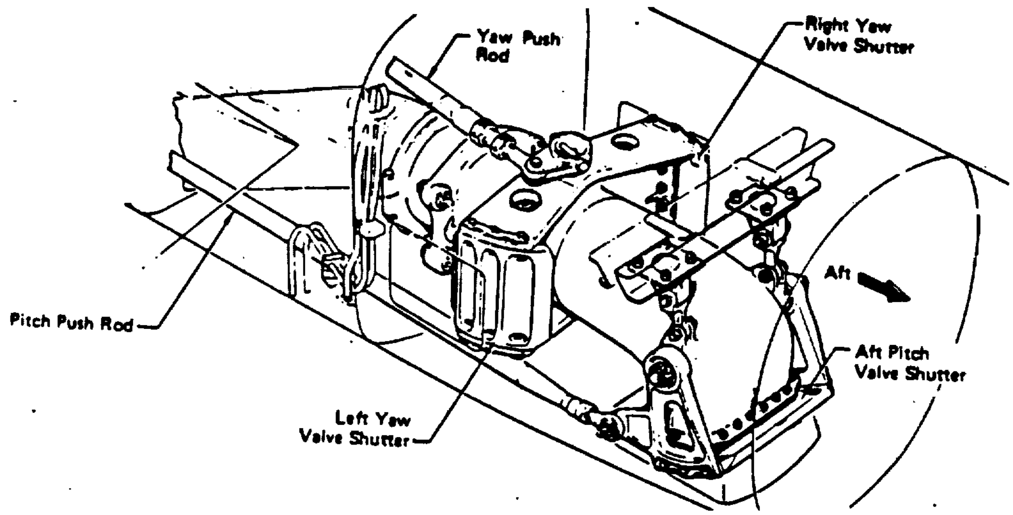

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Figure 7.9 Fuselage Aft Pitch and Yaw Reaction Valve and Controls

7.3.2 Suckdown Predictions

The method used to calculate the effects of suckdown are from Reference 7.3. This method takes into account:

* number of engine posts,
* geometry of the aircraft,
* pressure ratio at the nozzles,
* in and out of ground effects,
* and fountain/core effects.

The suckdown was calculated during Phase 1 of the design. The Monarch was a two post configuration in Phase 1. The suckdown predictions resulted in a 25% loss in lift versus thrust. The suckdown was assumed to be 10% for the lift engine during preliminary sizing. Therefore, the lift engine was undersized. For Phase 2 a second ventral nozzle was added so that the Monarch would become a three post configuration, which typically reduces suckdown. The suckdown was calculated for the Monarch in Phase 2 which resulted in a suckdown of 10%. Therefore, the Monarch was changed to a three post configuration so that the lift engine did not need resizing. The comparisons between the two post and three post configurations are shown in Figure 7.10.

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The suckdown predictions are important because the hover requirement sizes the lift engine for this aircraft. Therefore, any reduction in ground effects will result in a lower engine weight, and eventually lead to a lower life cycle cost of the Monarch.

The equations used to calculate the suckdown for the Monarch are from Reference 7.3. The results of the trade study between two and three post configurations is shown in Figure 7.8. Both configurations had identical geometry, the total nozzle area also remained the same for both configurations. The three post configuration needed 23% less engine thrust than the two post configuration at a height of four feet above the ground. Figure 7.10 shows that at heights above 15 feet the three post configuration has no advantages over the two post configuration.

The reason that the three post configuration has better in-ground effects is due to the thrust "fountain core" developed between the three nozzle posts. The "fountain core" produces lift because of the jet flow that is trapped under the fuselage due to the three separate jet flows impinging on each other. When the configuration has only two posts the upwash can not develop into a "core" and becomes a radial wall jet which does not produce as much lift as the three post configurations "fountain core".

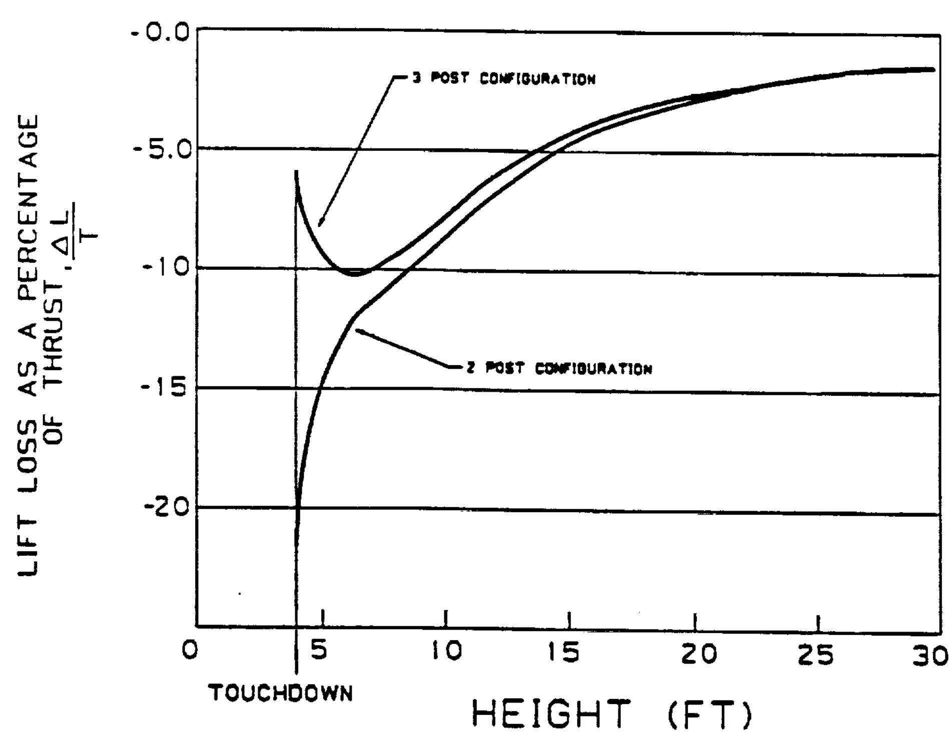

Figure 7.10 Suckdown Comparison for the Monarch

7.3.3 Hot Gas Reingestion

Hot Gas Reingestion is the term used to describe any flow mechanism by which hot exhaust gases from the propulsive system of an aircraft can return to the air intake of the same system. HGR is an especially important problem for STOVL aircraft operating near the ground and using propulsive lift.

Extensive theoretical research as well as full scale experiments have identified three ways in which the jet exhaust flows of a STOVL aircraft might recirculate back to the engine inlets. They are:

1) **Near Field Reingestion**--This is caused by the flows from separate lift jets meeting on the ground creating an upward or fountain flow which impinges on and is redirected by the aircraft undersurface. Some may travel directly on a short time scale to the engine inlets with little opportunity for mixing thereby retaining a high percentage of jet exit temperature and potentially causing severe HGR. It is shown in Figure 7.11 (Reference 7.4).

2) **Mid Field reingestion or Intermediate Thrust Reverser**--This is caused when some of the recirculating flow in the ground jet and the forward moving part of the fountain is blown back by headwind into the intake after some opportunity for mixing with ambient air. It is shown in Figure 7.9 (Reference 7.4).

3) **Far Field Reingestion**--This is caused when the ground flows travel radially outward mixing progressively with exhaust air to recirculate into the intake on a much longer time-scale driven by the effects of buoyancy and entrainment. The reingestion air temperature is then relatively low so Far Field Reingestion is not usually a serious problem. It is shown in Figure 7.13 (Reference 7.4).

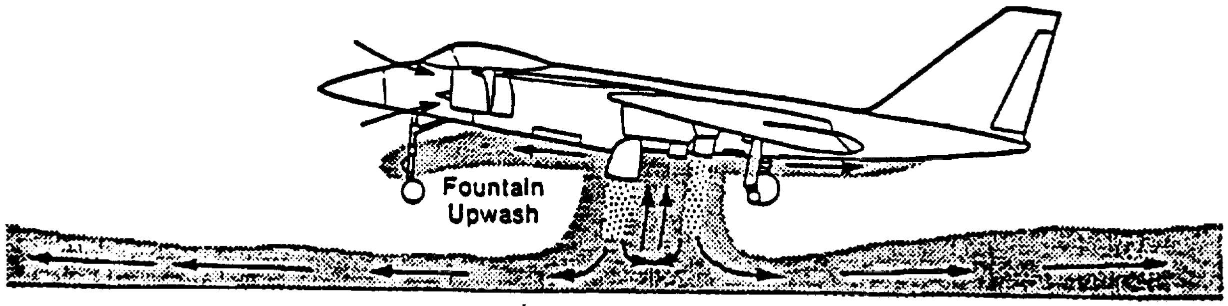

Figure 7.11 Example Near Field Reingestion (copied from Ref. 7.4)

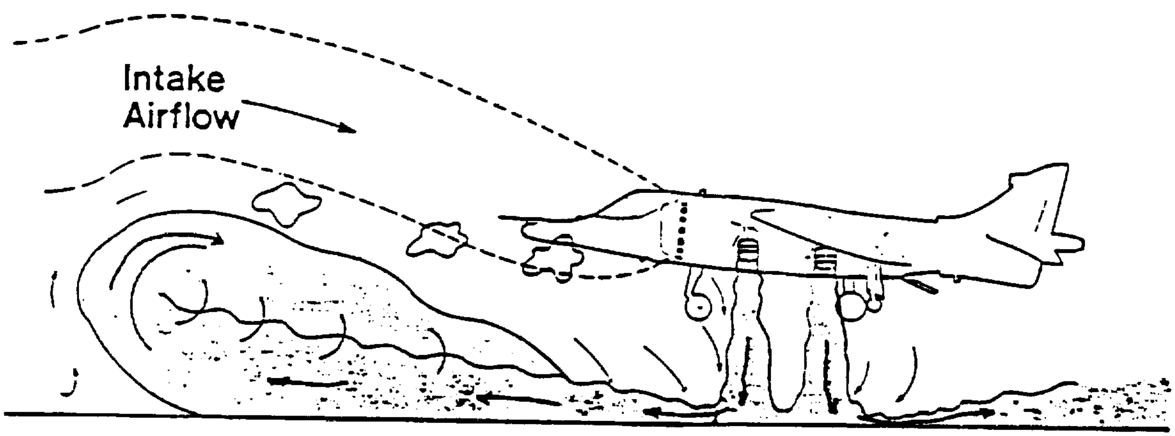

Figure 7.12 Example Mid Field Reingestion (copied from Ref. 7.4)

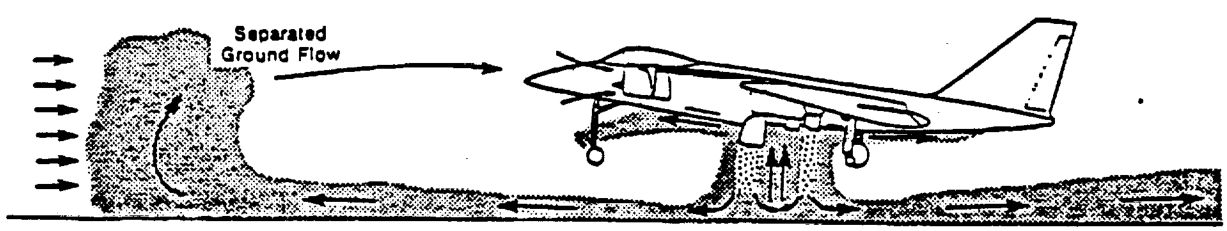

Figure 7.13 Example Far Field Reingestion (copied from Ref. 7.4)

All internal-combustion engines, and gas turbines in particular, are very sensitive to an increases in air intake temperatures. This arises from several causes: 1) Warmer air is less dense, and the mass flow of the working fluid is therefore reduced, resulting in a loss of thrust. 2) The speed of sound in air increases with temperature, and the compressor blade Mach number at a given rotational speed is therefore reduced; this reduces the compressor capability in both non-dimensional (corrected) airflow and pressure ratio. 3) A higher air inlet temperature results in higher gas temperatures throughout the engine, so that turbine temperatures become excessive; to prevent this, thrust demand must be reduced. 4) Air inlet temperatures which change rapidly in time or space (temperature distortion) may cause compressor stall (surge) (Reference 7.4).

A major determinant of the severity of HGR is the number and location of the vertical jet exhaust nozzles on the airplane. In a near ground environment, the flow of each jet will impact the ground, then spread radially. If the flow of one jet meets the flow of another jet, the flow will join and rise. A two jet configuration will result in a long wall of upward flow being generated between the two jets. A three-jet configuration will produce a concentrated fountain at the point where the flow of all three jets combine. There will also be three walls extending from this fountain where two of the jets combine. A four jet configuration (like the Harrier) will produce a more concentrated fountain at the center of the four jets and four walls will extend from it. It is the upward airflow that will reach the inlets and cause Near Field Reingestion so regardless of the number of nozzles, it is desired to keep the fountain and wall airflows away from the inlet area.

7.14 is a top-view of the Monarch showing the location of all nozzles and inlets as well as the fountain that is created by the engine flow. Notice that there is not a wall of airflow under the fuselage at the location of the inlets.

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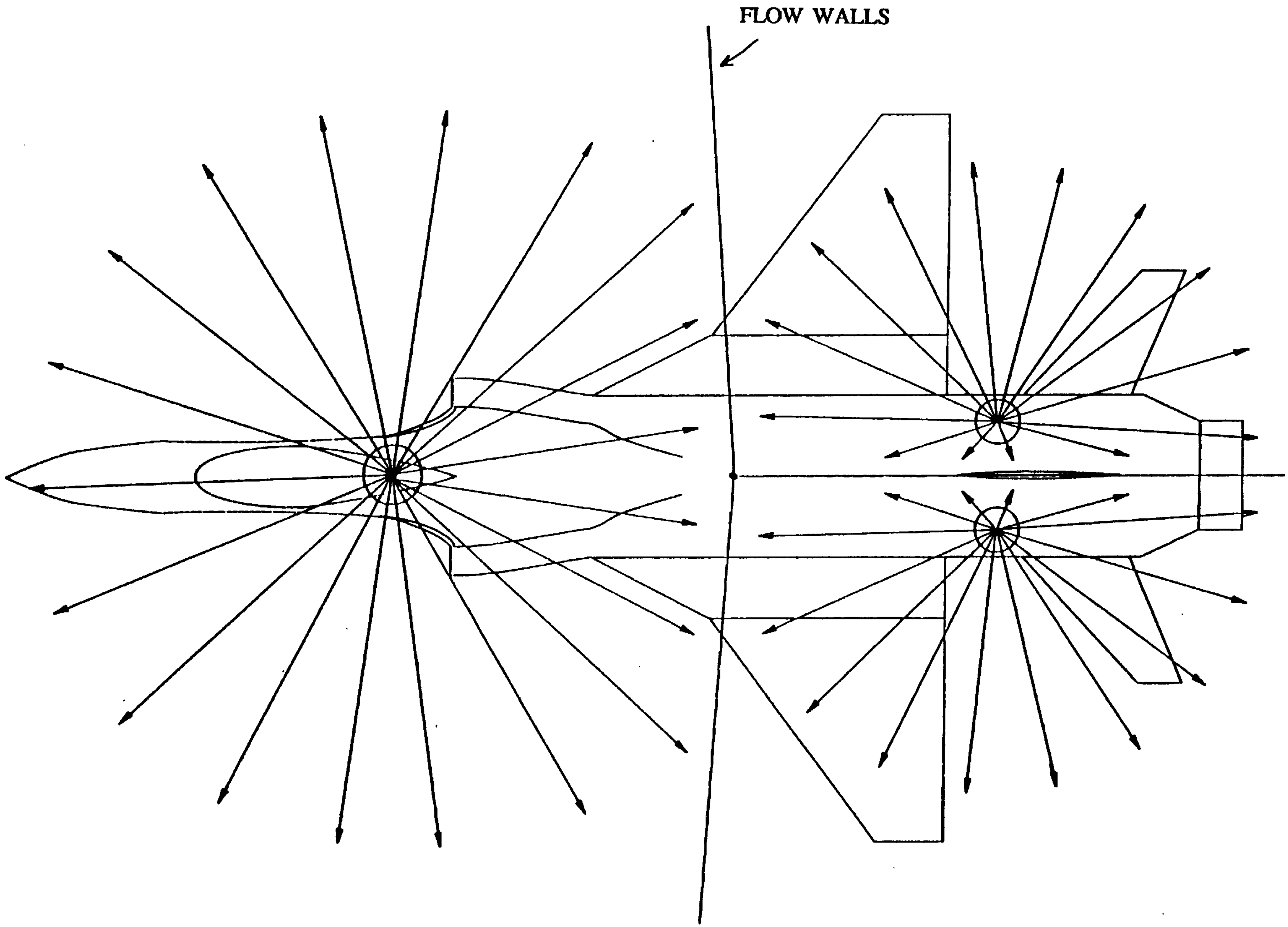

Figure 7.14 Monarch Hover Airflow Pattern

There has been extensive research into many methods of alleviating HGR by making slight modifications to an airplane configuration. If HGR does become a major problem for the Monarch, then one or more of these modifications should be made. The following list shows some of these ideas.

1) Attempt to deflect the Lift/Cruise nozzles outboard if there is not a significant loss in thrust. A slight deflection, as shown in Figure 7.15, will sufficiently direct the flow away from the airplane. Studies have shown that this may alleviate the ITR enough that the loss of thrust due to the nozzle angle is more than compensated by the improve engine performance. (Reference 7.5)

2) Place deflector shields near the nozzles to direct the flow away from the inlets. It may be possible to integrate current doors to the landing gear and the missile bay to also act as this type of shield, or it may be necessary to make separate shields that retract into the fuselage. Figure 7.16 demonstrates using a door that covers the Lift engine nozzle.

3) Create an "air curtain" around the inlets by ducting compressor air from the Lift/Cruise engine out of the fuselage near the inlets. This air flow will entrain and remove the hot gases that would otherwise enter the inlets. According to Reference 7.Y, approximately 2 % of the engine air flow is necessary to create this type of curtain. An approximate location as well as a schematic of the air flow is shown in Figure 7.17.

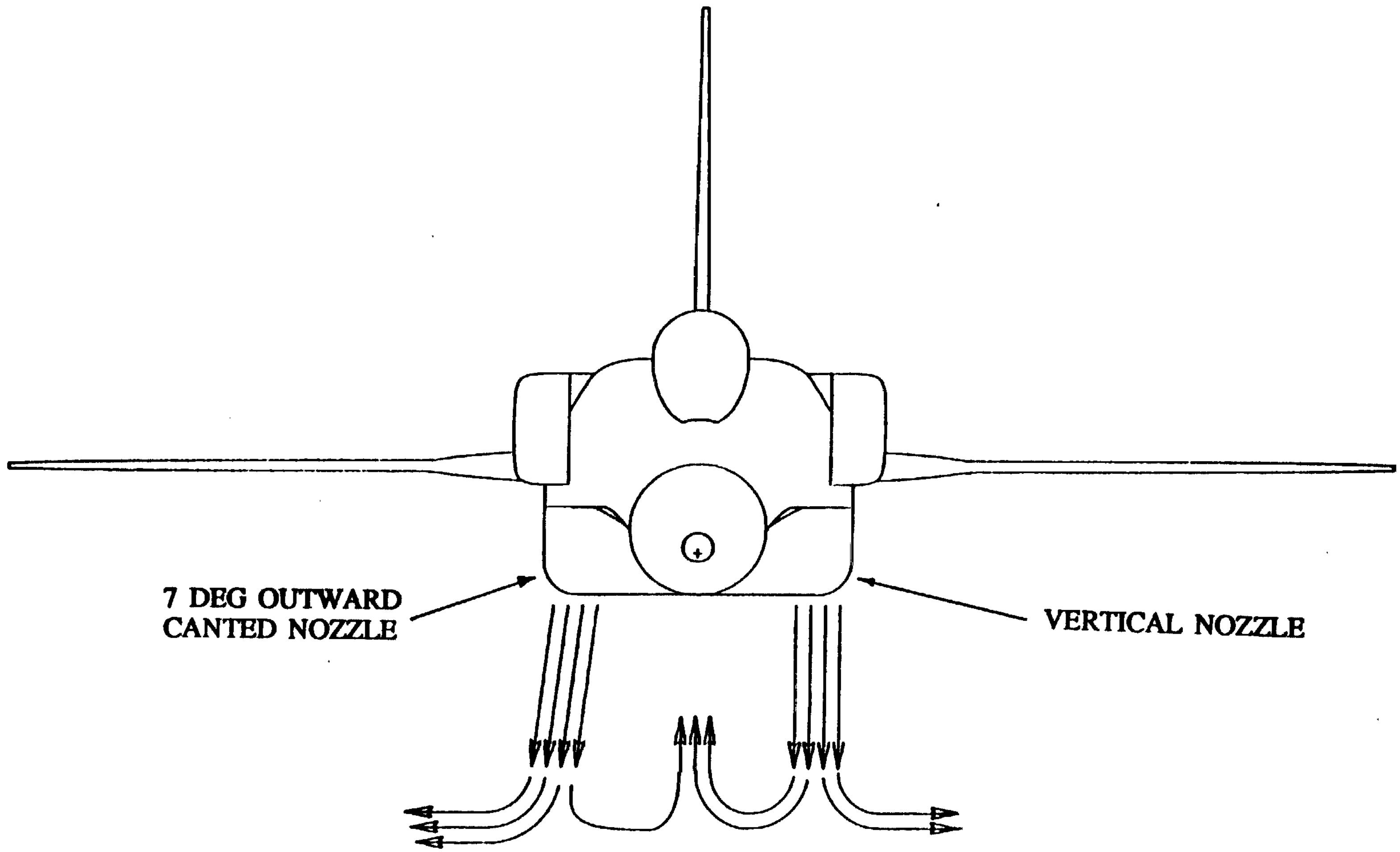

Figure 7.15 Lift/Cruise Nozzle Deflection Concept

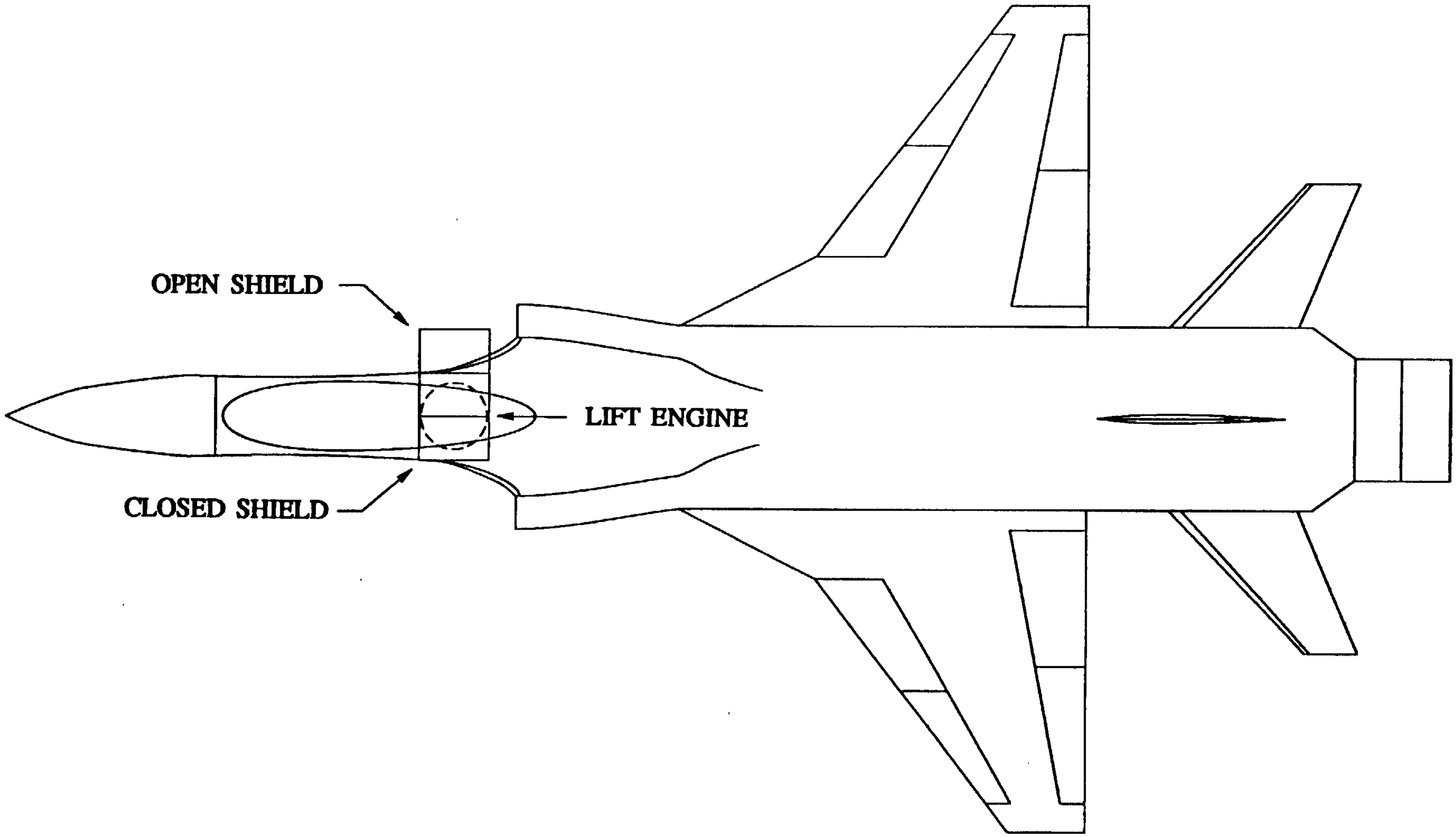

Figure 7.16 Nozzle Shield Concept

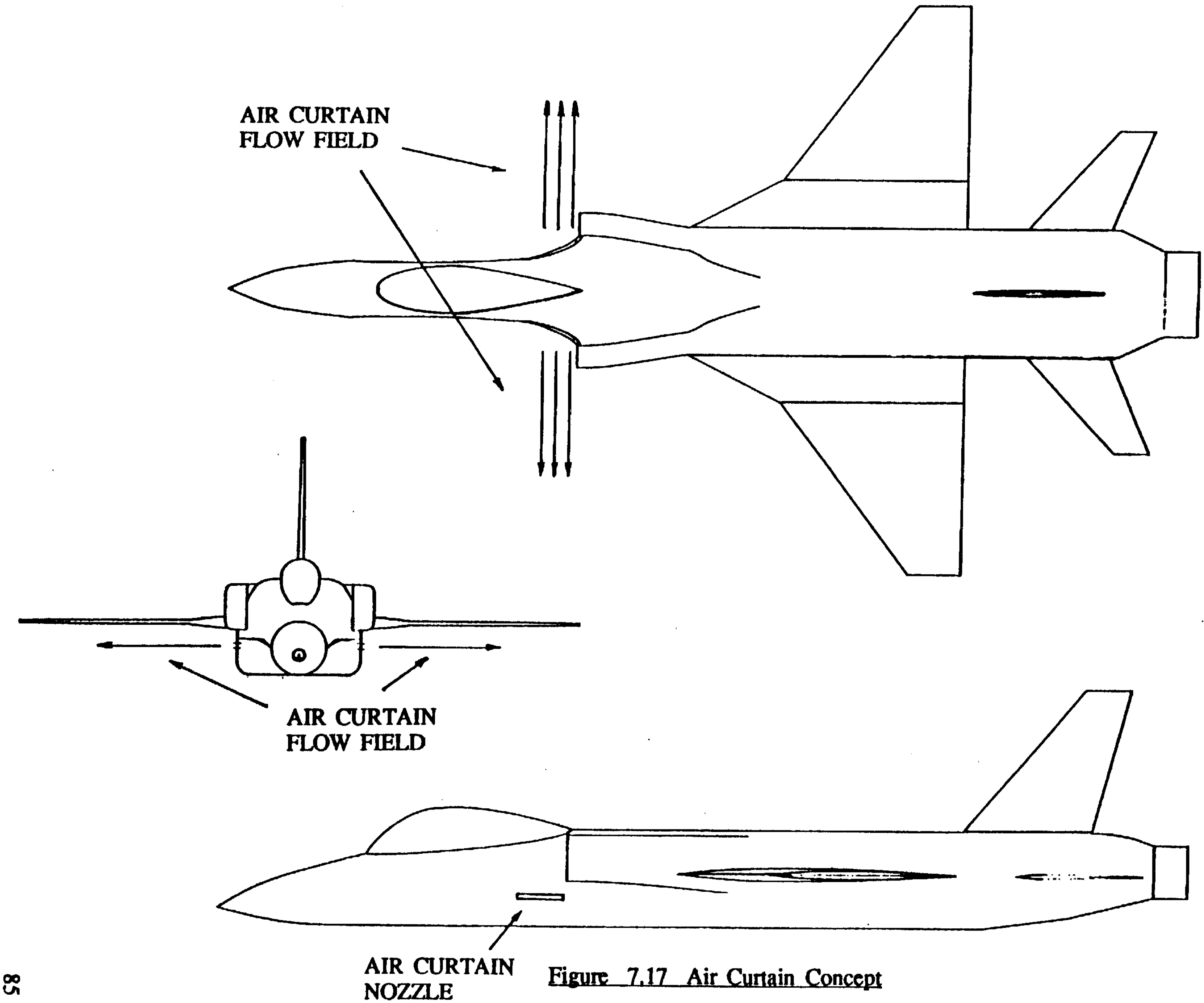

Figure 7.17 Air Curtain Concept

7.4 PILOT WORKLOAD ANALYSIS

Control requirements and pilot workload for STOVL aircraft are higher than that of conventional aircraft. The STOVL aircraft may be required to operate from conventional airfields, austere sites, and aircraft carriers. The capability for hover and low-speed flight and for rapidly transitioning between wing-borne and propulsion-borne flights permits the STOVL aircraft to operate into confined spaces associated with austere sites. These operations enforce precision of control of position, velocity, and attitude; such requirements exceed those imposed on conventional aircraft (Reference 7.6).

A major technological challenge to routine vertical flight operations of this class of aircraft in adverse weather and low-visibility conditions stems from the complex interaction of kinematics, aerodynamics, and propulsive forces and moments during transition as reflected in poor flying qualities as well as from limited control authorities. The availability of digital fly-by-wire controls makes it feasible to reduce the amount of pilot workload during takeoff, transition, and hover. To also help in reducing the pilot workload the number of control sticks will be reduced from three (Harrier AV-8B) to two. The digital fly-by-wire controls and the advancements made in flight control software will allow for reduced pilot workload.

The cockpit controls and displays for the Monarch are adapted from Reference 7.6. The cockpit controls and displays for transition is shown in Figures 7.18 and 7.19. The situation/director display (Figure 7.18) is a three-cue compensary flight director supplemented by situation information presented in both analog and digital format. The flightpath pursuit/situation display (Figure 7.19) projects a lead aircraft that is following the desired flight profile. The cockpit controls and displays for hover are shown in 7.20. The HUD format in transition and hover is shown in Figures 7.21 and 7.22, respectively.

The workload for the pilot at takeoff is reduced because the flight control software performs the nozzle and control surface deflections to minimize the takeoff distance. The methodology of the flight control system for takeoff is discussed in Section 7.1.

For landing the pilot will bring the aircraft to approach speed and at that time the pilot will have the option to select "landing". If the pilot selects landing the HUD will switch over to situation/director display and the cockpit controls will switch over for transition (Figure 7.18). The pilot will then be given the option to select the landing location with the Forward Looking Infra Red (FLIR). Once the landing location is selected the HUD will display the pursuit/situation display and also the cockpit controls will switch to the flightpath-centered pursuit (Figure 7.19). The pursuit/situation HUD will allow the pilot to follow the ghost plane and also the landing location will be displayed on the HUD. When the aircraft gets within hover range the hover HUD will be displayed and the flight controls will switch to the hover mode (Figure 7.20). The hover HUD will allow the pilot to see the desired hover point along with the other important information as shown in 7.22.

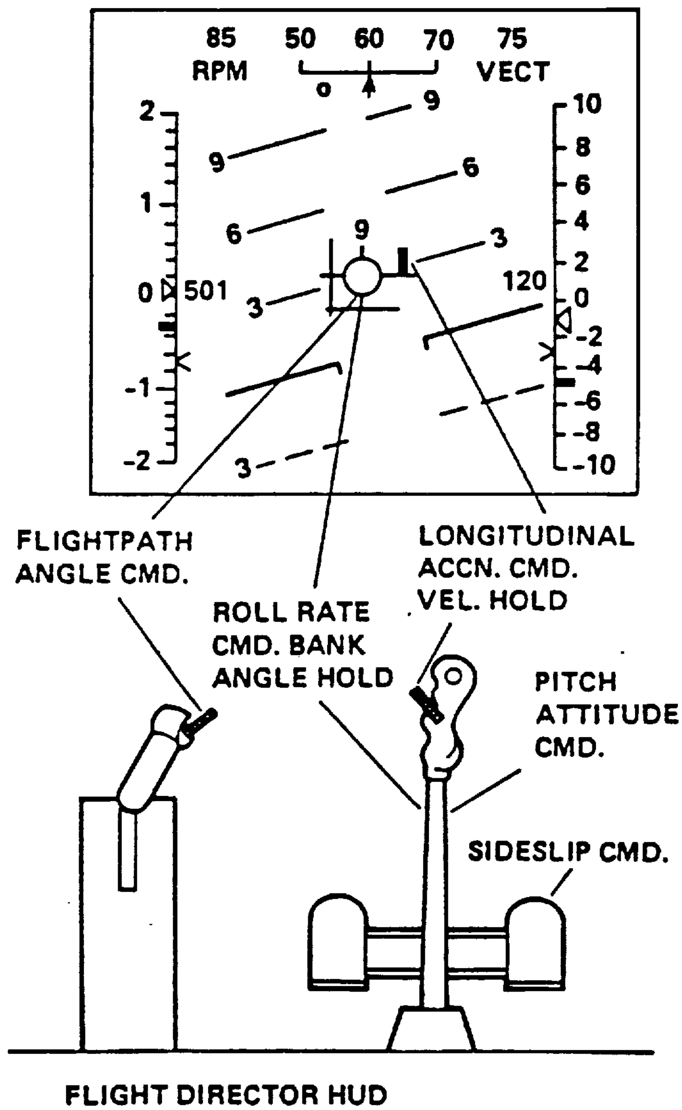

Figure 7.18 Cockpit Controls and Displays for Transition (Copied form Reference 7.6)

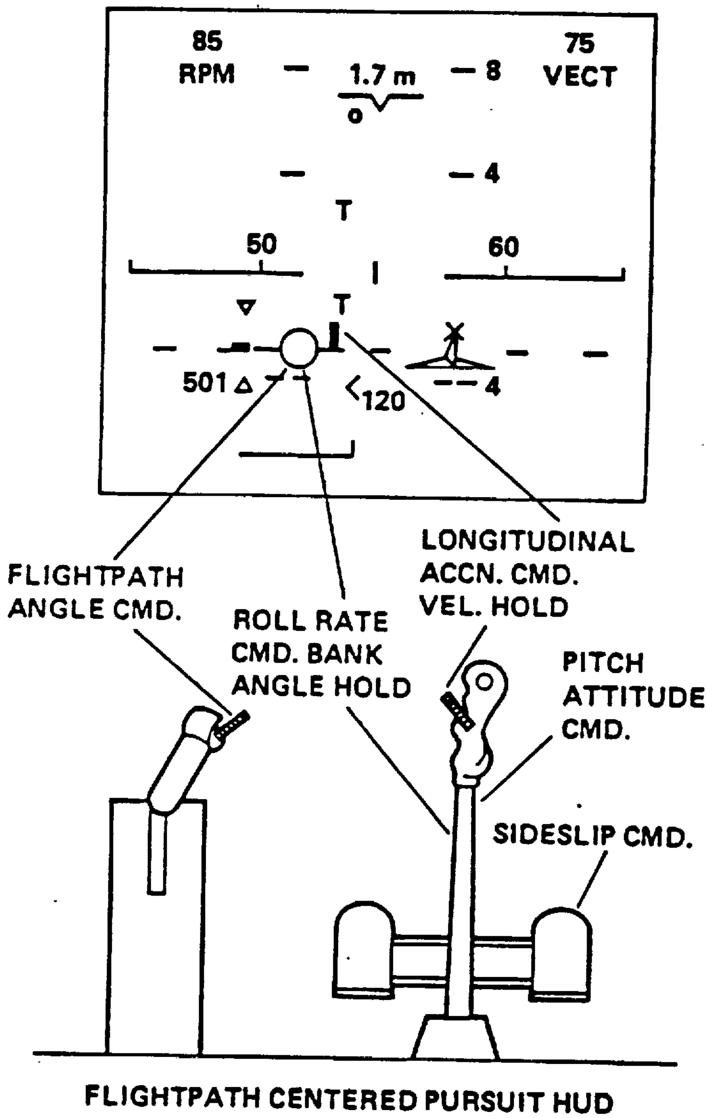

Figure 7.19 Cockpit Controls and Display for Approach (Copied from Reference 7.6)

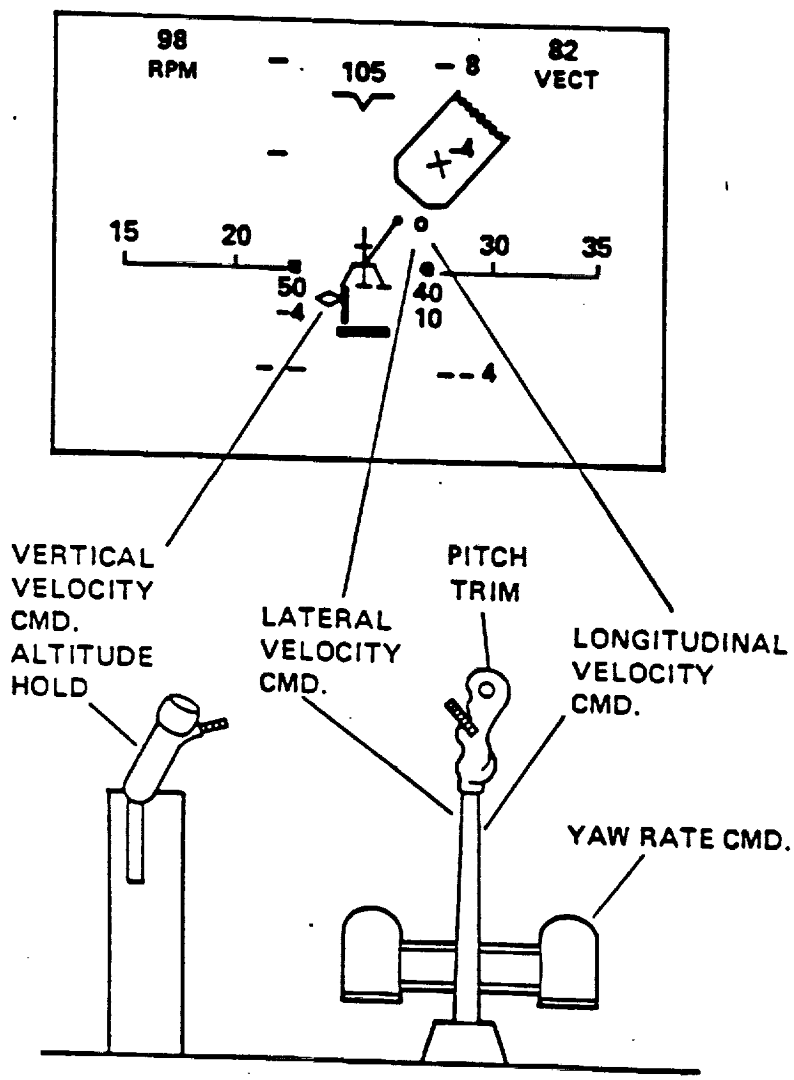

Figure 7.20 Cockpit Controls and Displays for Hover (Copied from Reference 7.6)

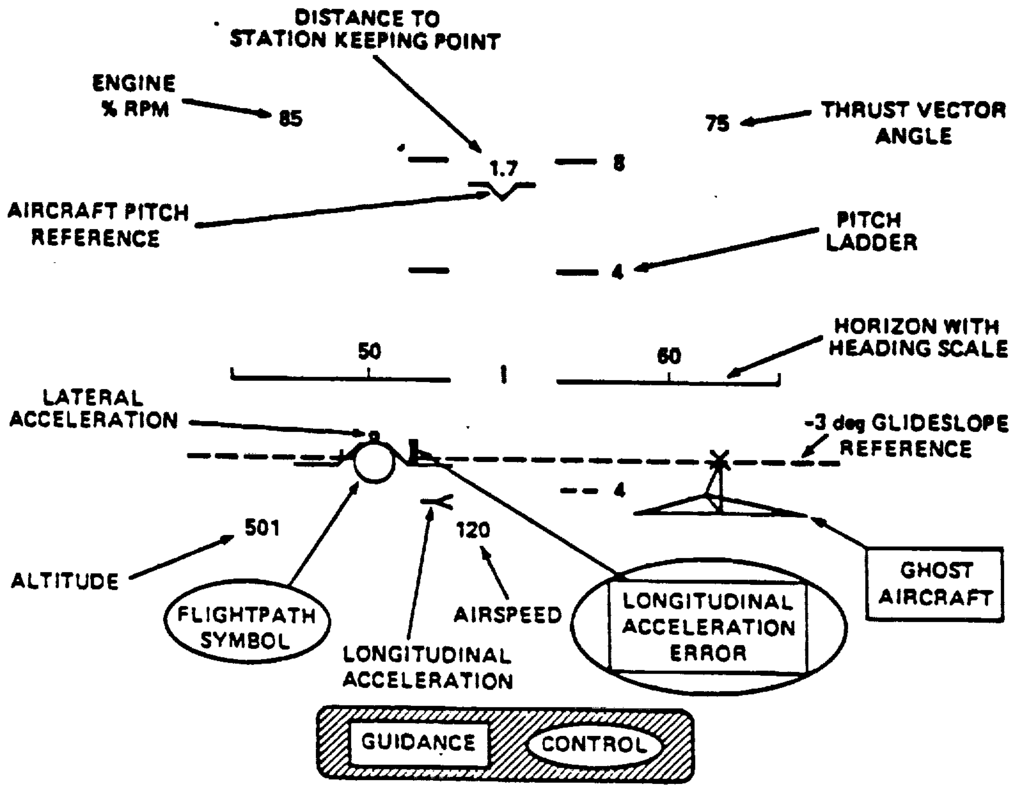

Figure 7.21 HUD Format in Transition (Copied from Reference 7.6)

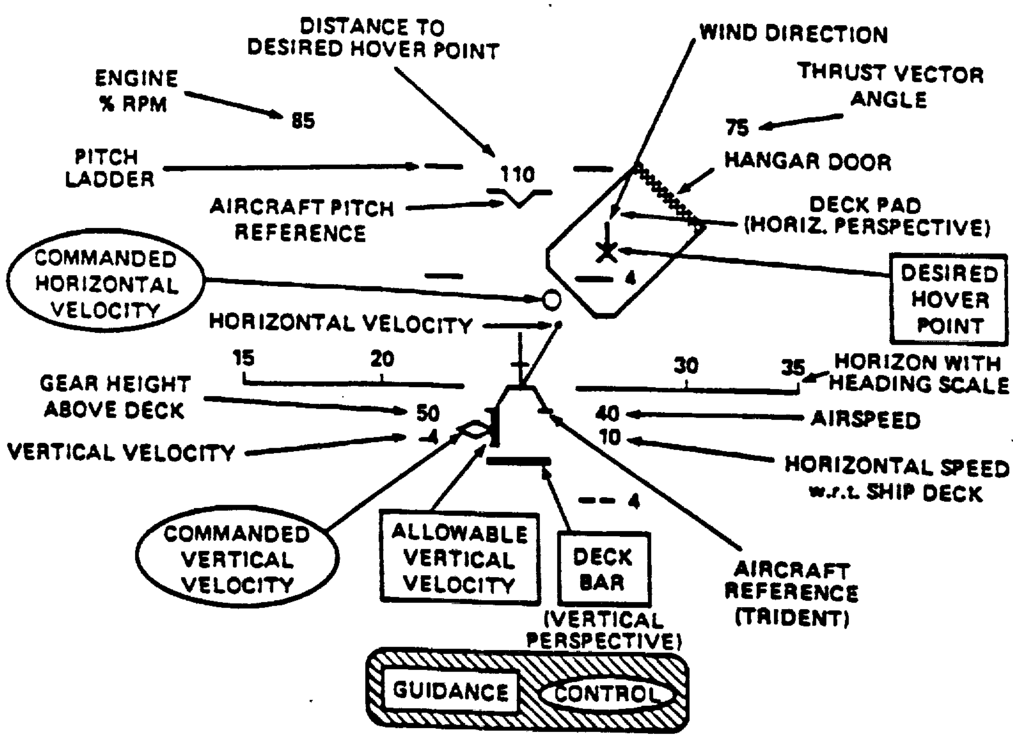

Figure 7.22 HUD Format in Hover (Copied from Reference 7.6)

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7.6 Foster, John D., and Schroeder, Jeffrey A., "Integrated Control and Display Research for Transition and Vertical Flight on the NASA V/STOL Research Aircraft", NASA TM 100029, October 1987.

8. PERFORMANCE DATA AND MISSION CAPABILITY

The purpose of this chapter is to present the performance data and mission capability of the Monarch aircraft. The drag characteristics of the aircraft is summarized and shown in Section 8.1. Section 8.2 presents the performance data and Section 8.3 presents the mission capability. The spreadsheets used to calculate the performance data and mission capability are shown in Appendix 2.

8.1 SUMMARY OF DRAG CHARACTERISTICS

The drag polars of the aircraft were calculated and are fully documented in Reference 8.1. The drag polars were adjusted to account for trim drag in Reference 8.2. The Monarch drag polars are shown in Table 8.1. The validity of the drag calculations is shown using Figure 8.1, where the skin friction coefficient of the Monarch is compared to similar aircraft. The wave drag for the configuration was calculated using the method of Reference 8.3 and actual data for the Grumman F-14 and the General Dynamics F-16 taken from Reference 8.4. The Monarch wave drag is shown in Figure 8.2.

Table 8.1 Monarch Drag Polars

H(ft)	M	Zero Lift Drag, C_{D_0}	Induced Drag Factor, 1/(PI*A*e)
0	0.20	0.02198	0.1091
100	0.85	0.02096	0.1022
10,000	0.90	0.02281	0.1002
15,000	0.90	0.02410	0.1003
30,000	0.90	0.02750	0.1103
30,000	1.20	0.04157	0.1006
30,000	1.60	0.04038	0.1008
40,000	0.80	0.02387	0.1103

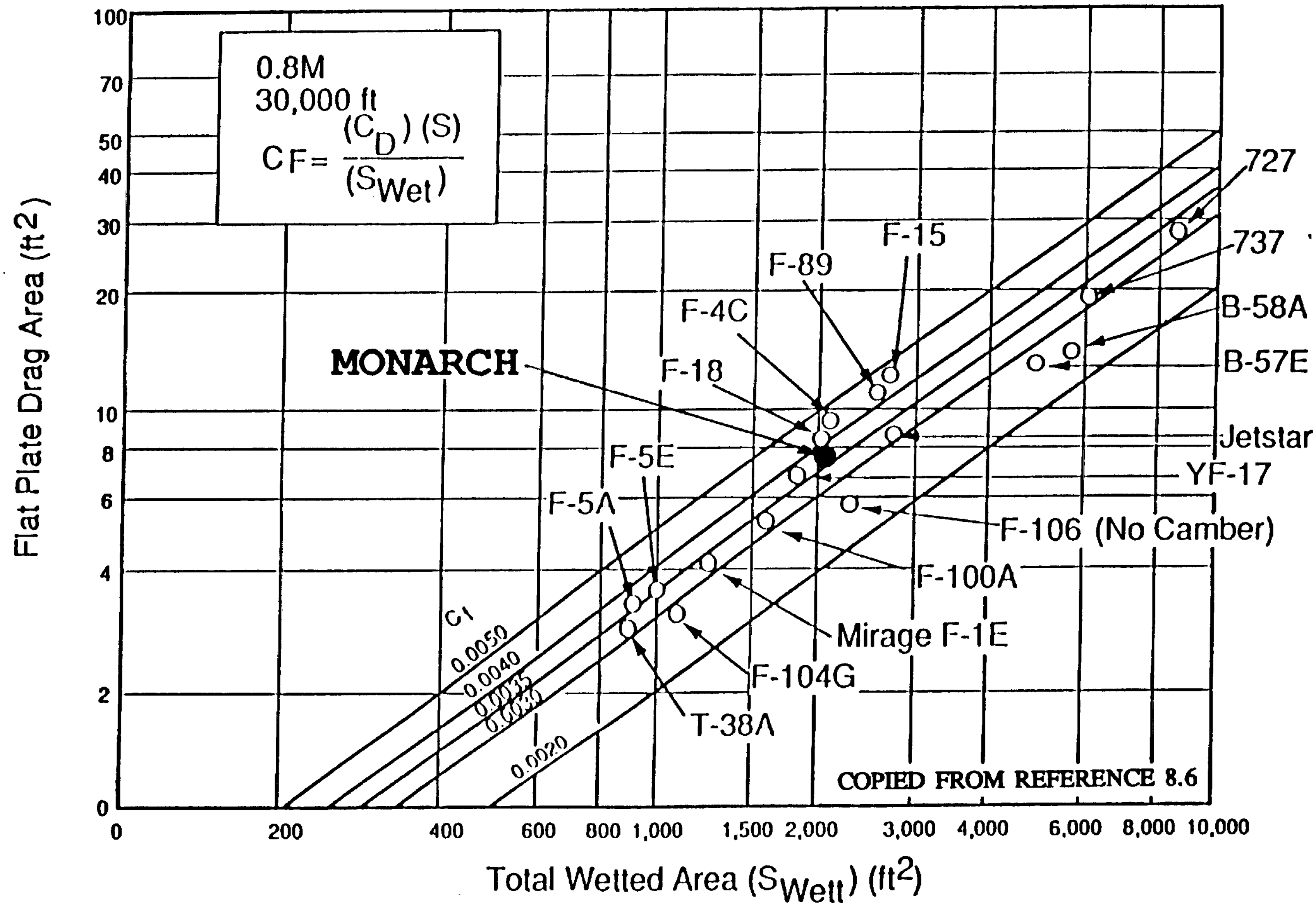

Figure 8.1 Skin Friction Coefficient Comparison for the Monarch

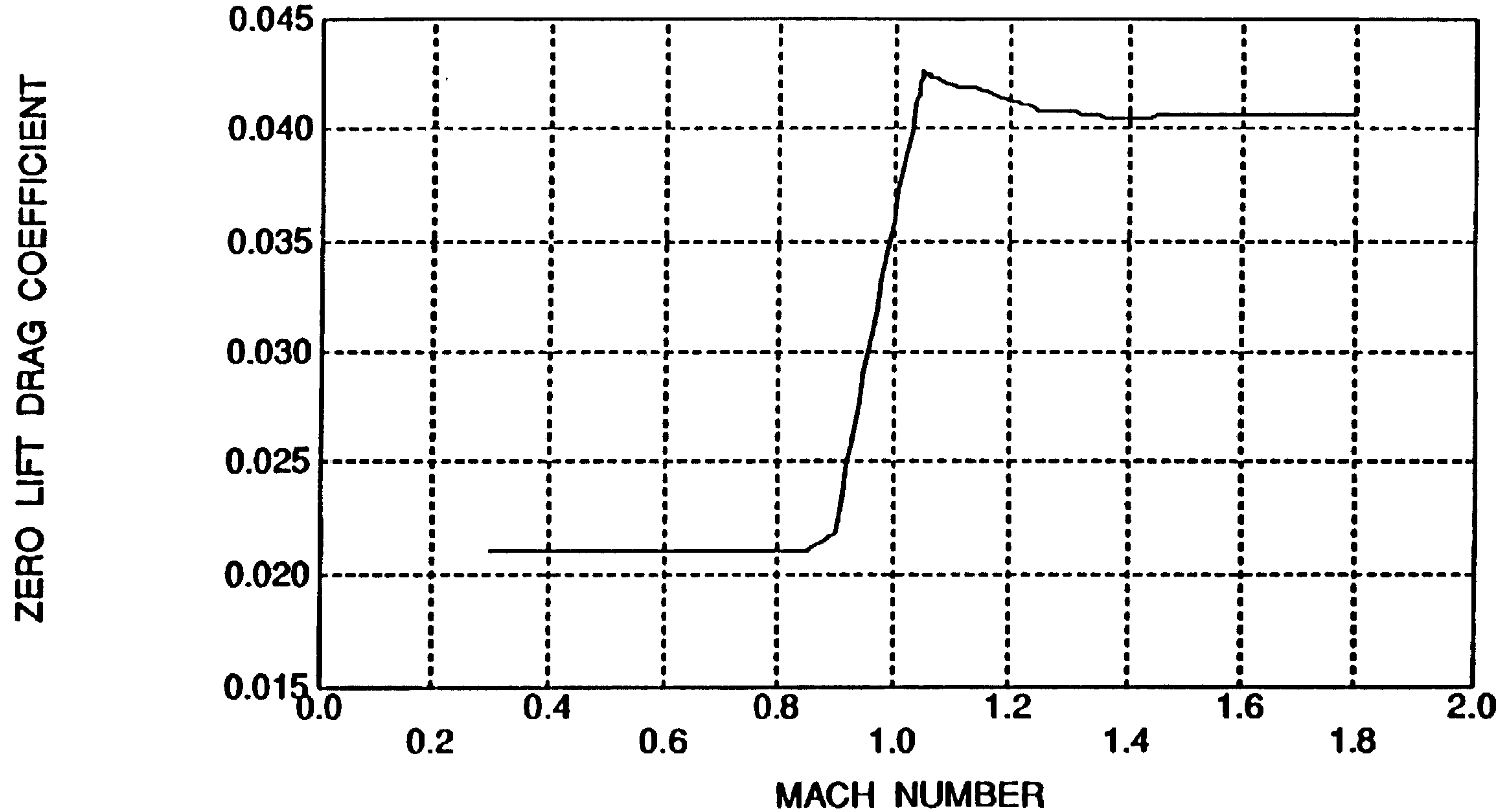

Figure 8.2 Wave Drag for the Monarch

8.2 PERFORMANCE DATA

The following performance data is presented:
* point performance verification
* sustained turn rate and load factor
* specific excess energy
* maximum ferry range

The Monarch also compared to operational fighters in the United States and Soviet Union to show its validity as a design and its combat effectiveness against these aircraft.

Point Performance Verification

Table 8.2 shows the point performance requirements from the mission specification and the values calculated for the Monarch. Note: All performance data presented are for a combat weight of 26,192 lbs which includes 50% fuel, two short range missiles, and half the ammunition for the cannon.

Table 8.2 Point Performance Verification for the Monarch

Performance Requirement	Required Value	Monarch Value
Time to Climb	40k in 2 minutes	1.75 min
1g Specific Excess Energy		
(2A) 30k 0.9M	500 ft/sec	505 ft/sec
(2B) 10K 0.9M	1,000 ft/sec	920 ft/sec
Sustained Turn Rate		
(3A) 0.8M/15k ft	15 deg/sec	15 deg/sec
(3B) 0.9M/30k ft	9 deg/sec	10 deg/sec
(3C) 1.2M/30k ft	8 deg/sec	9.9 deg/sec
(3D) 0.9M/15k ft	6.5 g	7.75 g
(3E) 1.6M/30k ft	4.5 g	8.70 g
Acceleration		
(4A) 30k ft 0.9M to 1.6M	70 sec	47.3 sec
(4B) 0.5M to 1.4M	80 sec	62.1 sec
(4C) 10k ft 0.3M to 0.9M	22 sec	18.4 sec
Landing Distance (ground roll)		
Without Chute	2,200 ft	2,100 ft

The Monarch meets all it required performance except for the 1000 ft/sec specific excess energy requirement. The improved performance of this aircraft as compared to the Phase I study is due to the upsizing of the engine of the Monarch which was done in the Phase II iteration.

Sustained Turn Rate and Load Factor

The sustained turn rate and load factor were calculated for the Monarch and are shown in Figures 8.3 and 8.4, respectively. The Monarch is capable of a sustained turn rate of 21 deg/sec at low level and can sustain 6 deg/sec at altitudes as high as 45,000 ft. A sustained load factor of 9 g's is maintained for much of the low altitude and high Mach number flight envelope. A 3 g sustained load factor is achievable at altitudes up to 50,000 ft.

The Monarch's turn performance at 15,000 ft is shown in Figure 8.5. This "dog house" plot shows the relationship between turn rate, load factor, turn radius, and Mach number. This plot shows that the Monarch can sustain high rates of turn over the operating Mach number range due to its high thrust engine. The maximum sustained turn rate at 15,000 ft for the aircraft is 16.9 deg/sec (thrust limited) and the maximum instantaneous turn rate is 17.3 deg/sec (lift limited).

Specific Excess Energy

The 1g specific excess energy for the flight envelope was calculated for the Monarch and is shown in Figure 8.6. The Monarch has a 1,000 ft/sec specific excess energy at high subsonic Mach numbers at altitudes below 10,000 ft. A specific excess energy of 600 ft/sec is achievable over a wide part of the high Mach number flight envelope.

Maximum Ferry Range

The maximum ferry range calculations are plotted in Figure 8.7. The maximum range of the aircraft is 1662 nm at 45,000 ft and M = 0.9. Range credit for climb was included in the calculations, as well as fuel use for climb, descent, and takeoff. This amount of range is feasible with fuel tanks fitted into the internal weapons bay volume. The aircraft uses two cylindrical tanks for this application.

The takeoff maximum thrust and combat weight versus Mach number are shown for the Monarch and several other fighter, respectively, in Figures 8.8 and 8.9. The Monarch fits into the trend of these other aircraft within reason.

As a measure of the Monarch's combat effectiveness, its turn rate and agility potential are compared to several other fighters. These plots are shown in Figures 8.10 and 8.11, respectively. The sustained turn rate (at 15,000 ft and Mach 0.9) of the Monarch exceeds the instantaneous turn rate of the Mig-21, Mig-23, and the F-15. The Monarch and the F-16 have comparable turn capabilities at this Mach and altitude. The agility potential shown is one of the only static agility metric available and is defined as:

$$\text{Agility Potential} = (T_{MAX}/W_{TO})/(W_{COM}/S)$$

The Monarch compares favorable to the F-14 and F-16, but falls short of the agility potential of the F-15. A lower wing loading for the Monarch would improve this ability, but would then make the aircraft less comfortable on a bombing mission.

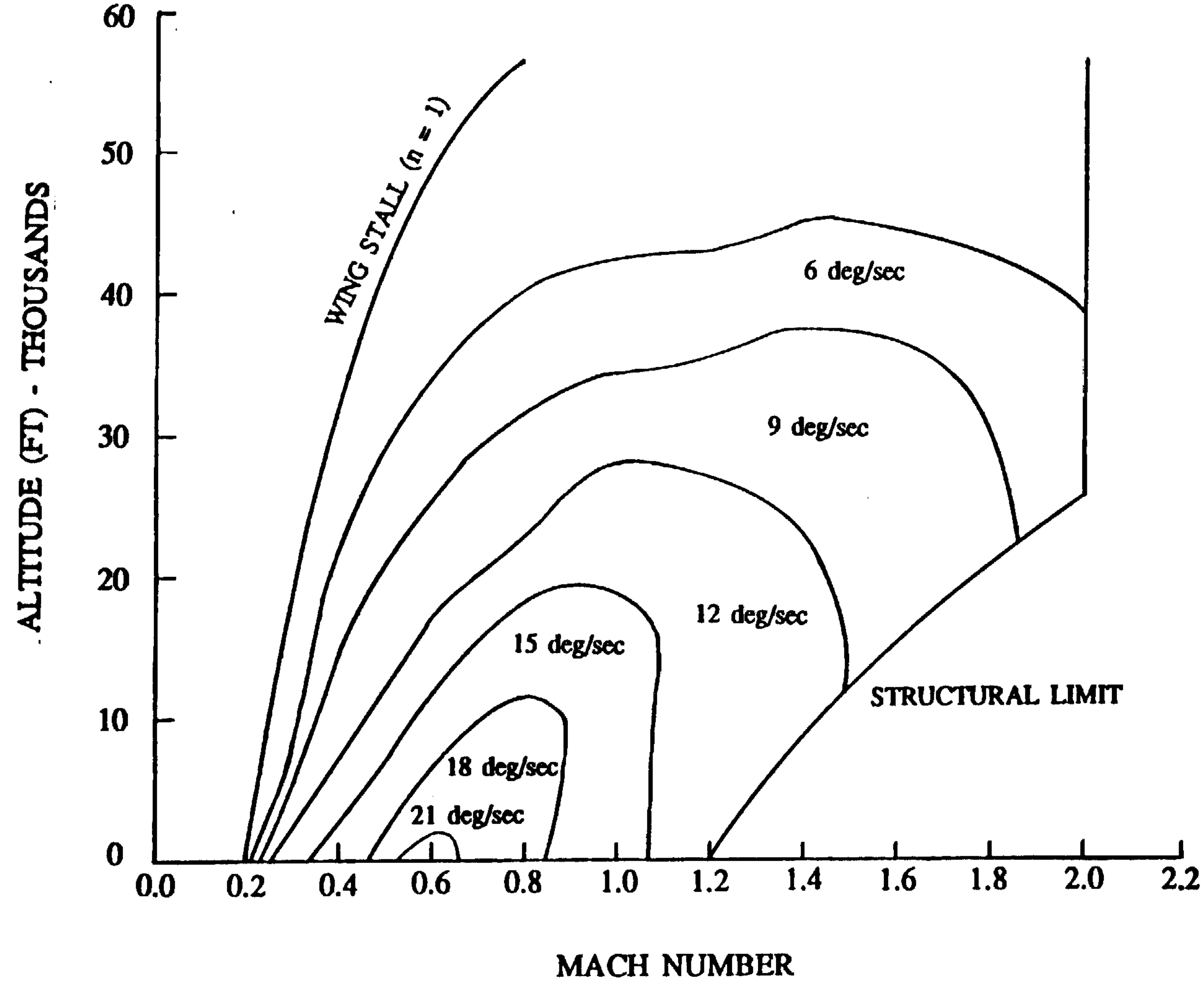

Figure 8.3 Sustained Turn Rate Capability of the Monarch Aircraft

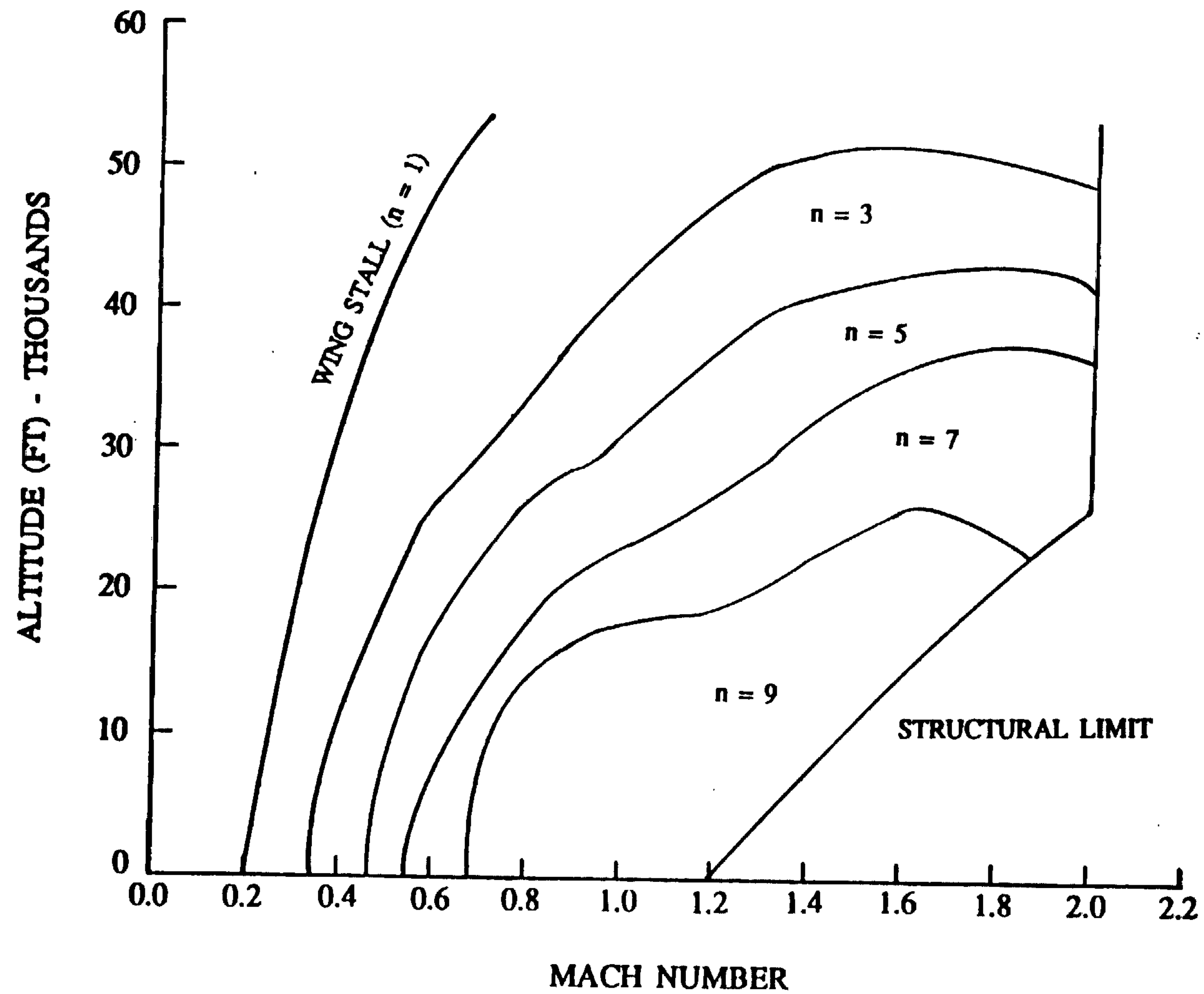

Figure 8.4 Sustained Load Factor Capability of the Monarch Aircraft

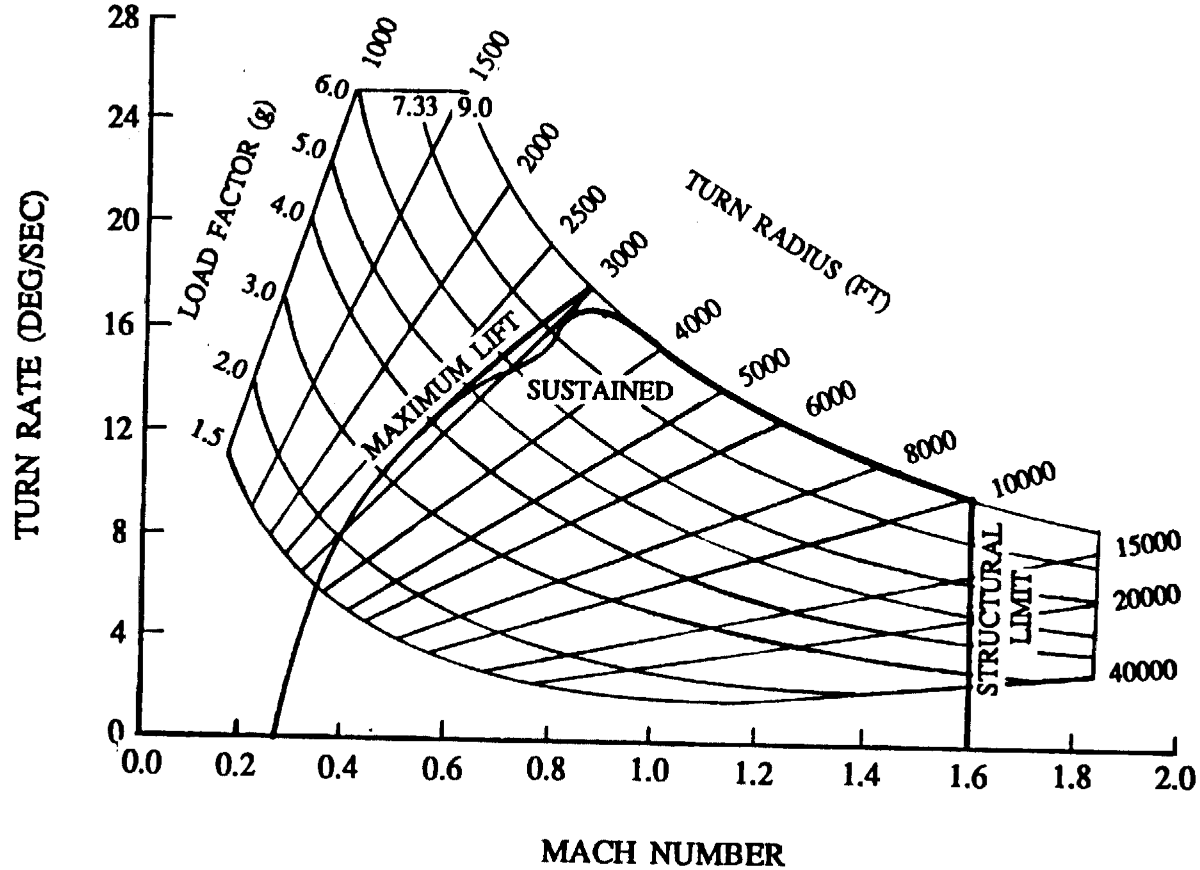

Figure 8.5 Turn Performance of the Monarch Aircraft at 15,000 ft

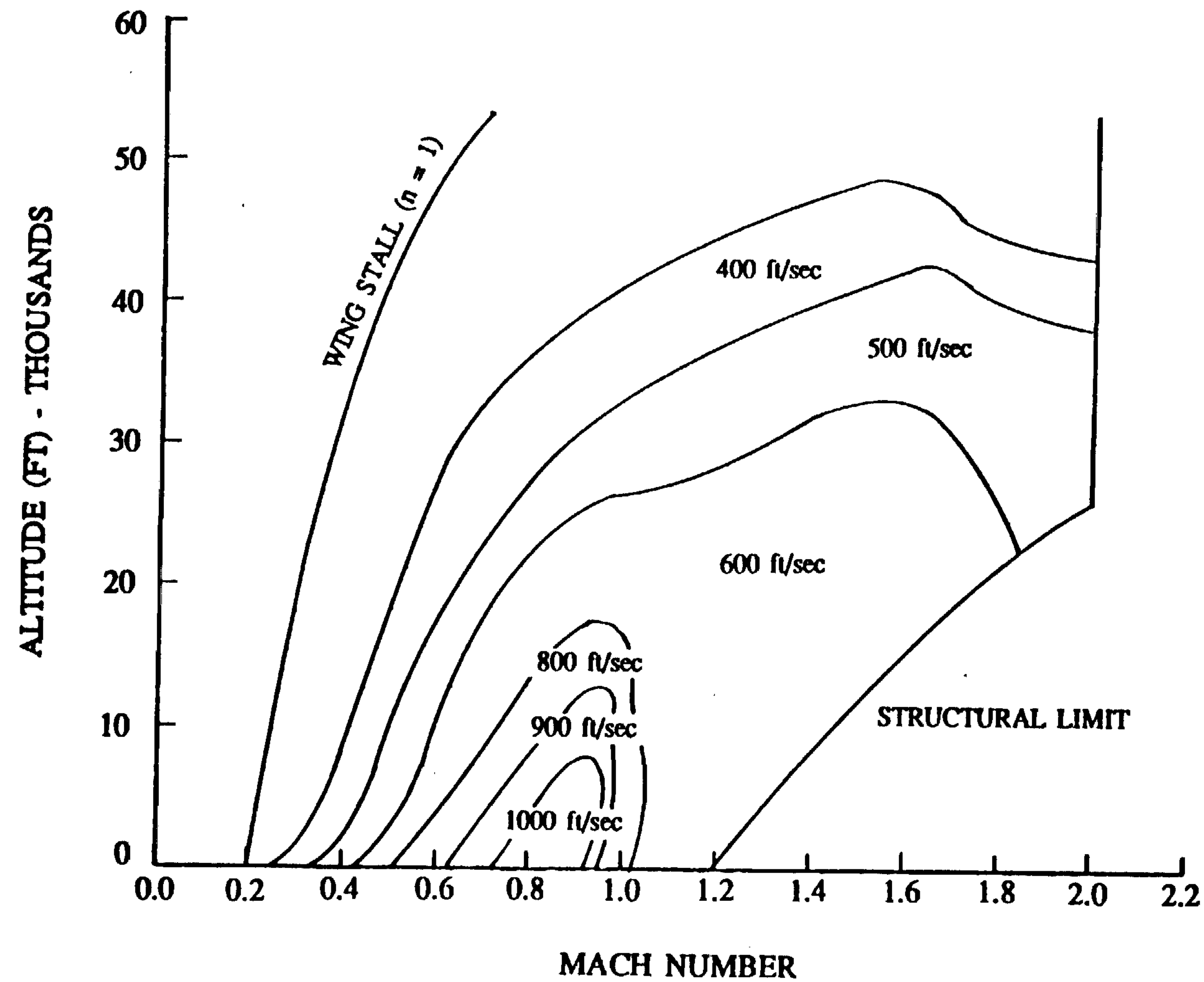

Figure 8.6 Specific Excess Energy Capability of the Monarch Aircraft

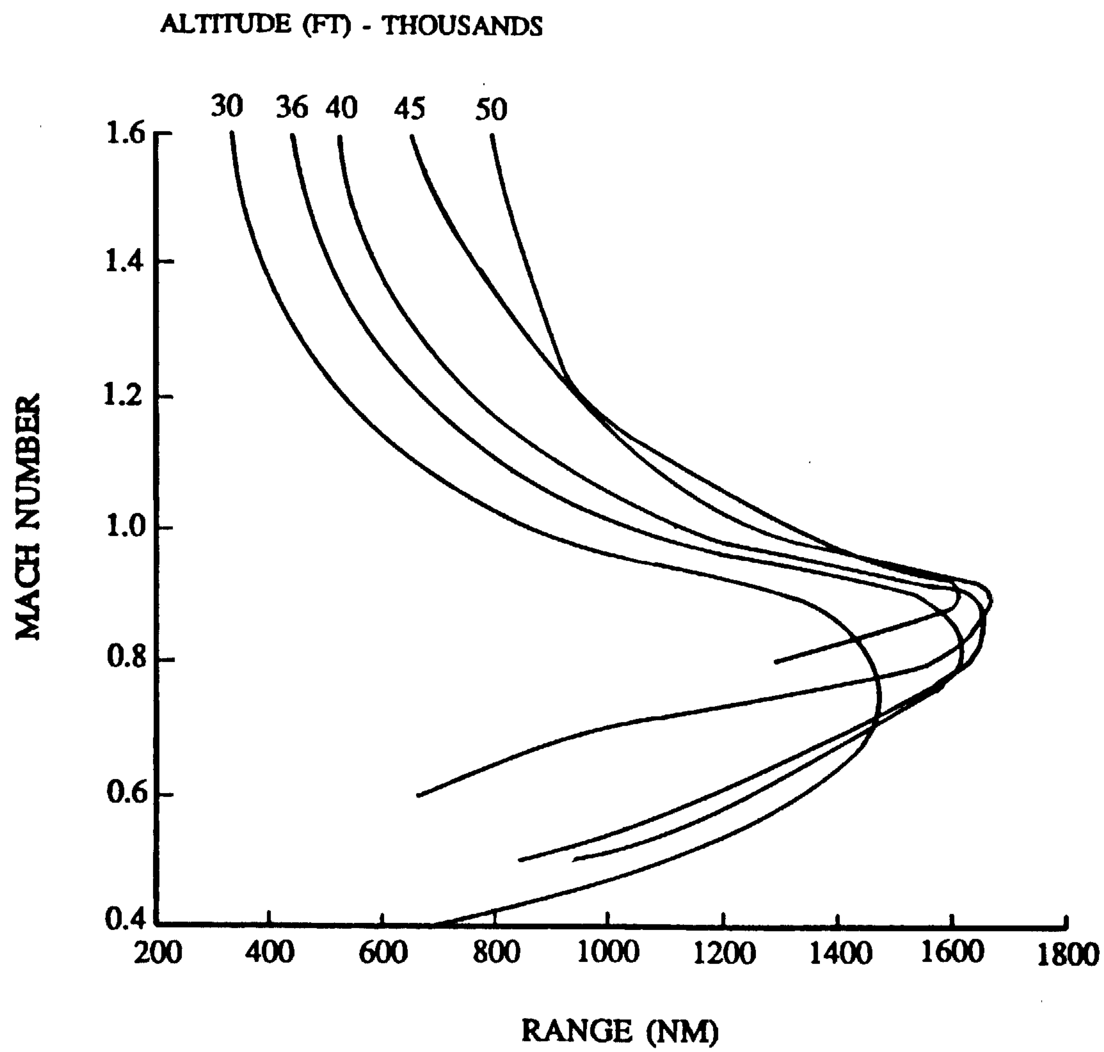

Figure 8.7 Maximum Ferry Range of the Monarch Aircraft

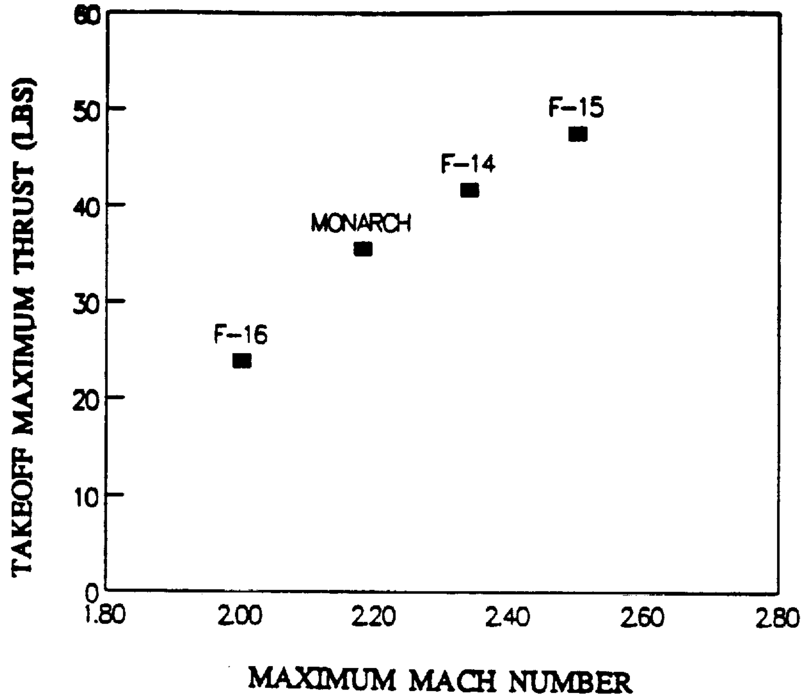

Figure 8.8 Takeoff Maximum Thrust and Maximum Mach Number Comparison

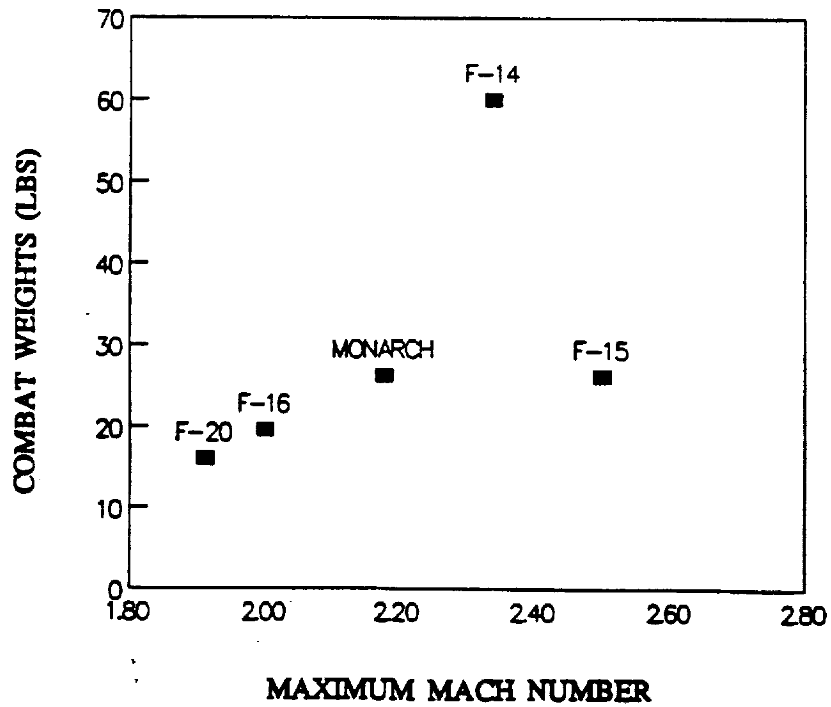

Figure 8.9 Combat Weight and Maximum Mach Number Comparison

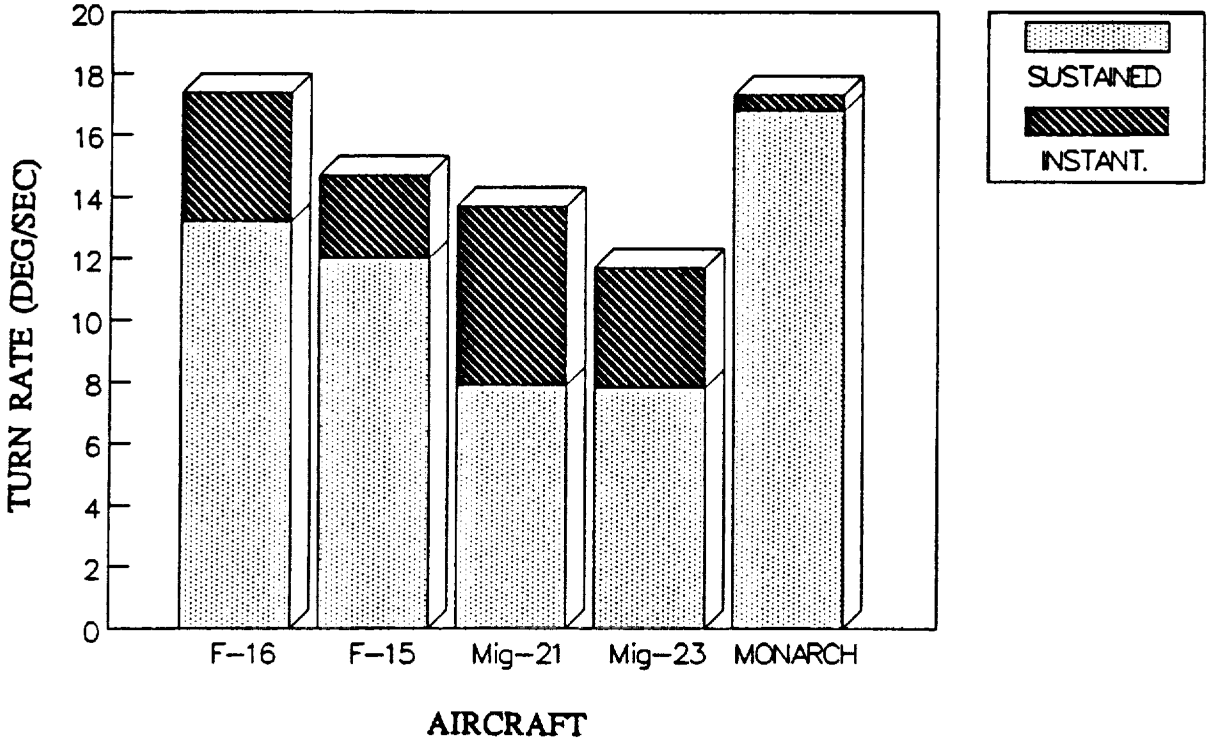

Figure 8.10 Sustained and Instantaneous Turn Rate Comparison

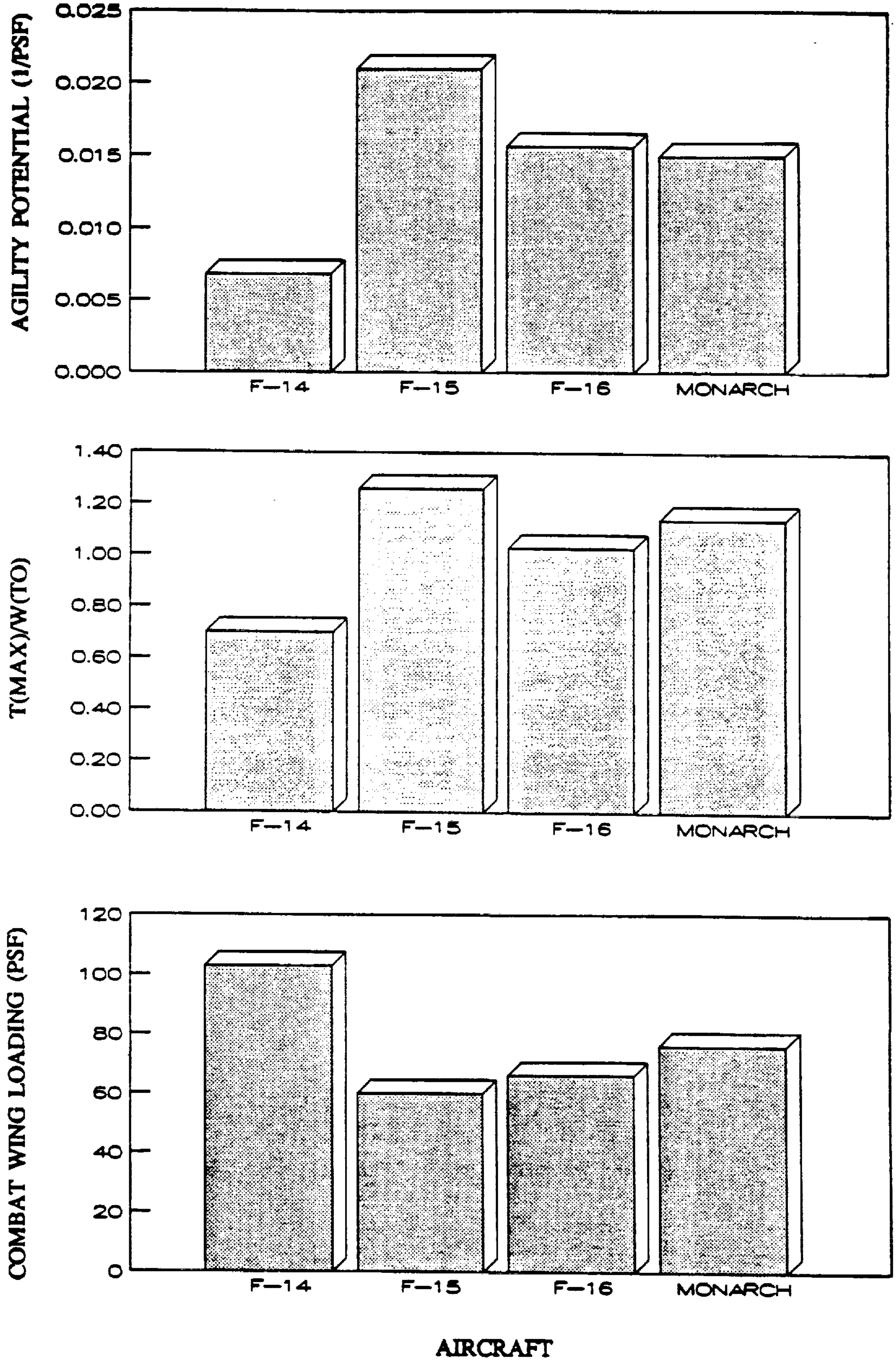

Figure 8.11 Agility Potential Comparison

8.3 MISSION CAPABILITY

The mission capability of the Monarch is measured by first verifying the design missions and second, taking the aircraft through typical fighter/attack missions to determine the aircraft's capability as a multi-role fighter. Tables 8.3 and 8.4, respectively, show the counter air mission and battlefield air interdiction missions fuel usage for the design mission. The supersonics (acceleration to and sustaining supersonic flight) of the counter air mission and the low level dash of the battlefield air interdiction mission dominate the aircraft fuel usage.

Table 8.3 Counter Air Mission Fuel Burn Summary

Phase	Fuel Burn
1. Engine Start/Warm Up	314 lbs
2. Taxi	279 lbs
3. Short Takeoff	360 lbs
4. Acceleration to Climb Speed	313 lbs
5. Climb	485 lbs
6. Subsonic Cruise - 100 nm	531 lbs
7. Acceleration to Supersonic Cruise	620 lbs
8. Supersonic Cruise - 50 nm	1334 lbs
9. Combat	1728 lbs
10. Supersonic Cruise - 50 nm	1325 lbs
11. Subsonic Cruise - 100 nm	571 lbs
12. Hover	227 lbs
13. Landing	114 lbs
14. Reserves	432 lbs
CA Mission Fuel Burn =	8634 lbs

Table 8.4 Battlefield Air Interdiction Mission Fuel Burn Summary

Phase	Fuel Burn
1. Engine Start/Warm Up	327 lbs
2. Taxi	307 lbs
3. Short Takeoff	376 lbs
4. Acceleration to Climb Speed	308 lbs
5. Climb	538 lbs
6. Subsonic Cruise - 200 nm	1331 lbs
7. Sea Level Dash In - 80 nm	1204 lbs
8. Strafe Run	864 lbs
9. Sea Level Dash Out - 80 nm	1110 lbs
10. Climb	326 lbs
11. Subsonic Cruise - 200 nm	1124 lbs
12. Hover	246 lbs
13. Landing	121 lbs
14. Reserves	432 lbs
Battlefield Air Interdiction Mission =	8614 lbs

Typical NATO fighter/attack mission profiles were obtained from Reference 8.5. The missions are:

Figure 8.12 Mass Intercept
Figure 8.13 Transport/Helicopter Intercept
Figure 8.14 AWACS/High Value Asset Protection
Figure 8.15 Two Stage Mission

The figures show the Monarch's range and speed capability in these missions. The high value asset protection mission and the two stage mission offer unique advantages for a STOVL type aircraft. As shown in the profiles, a STOVL aircraft can operate from dispersed bases and thus save fuel and cut down on response time.

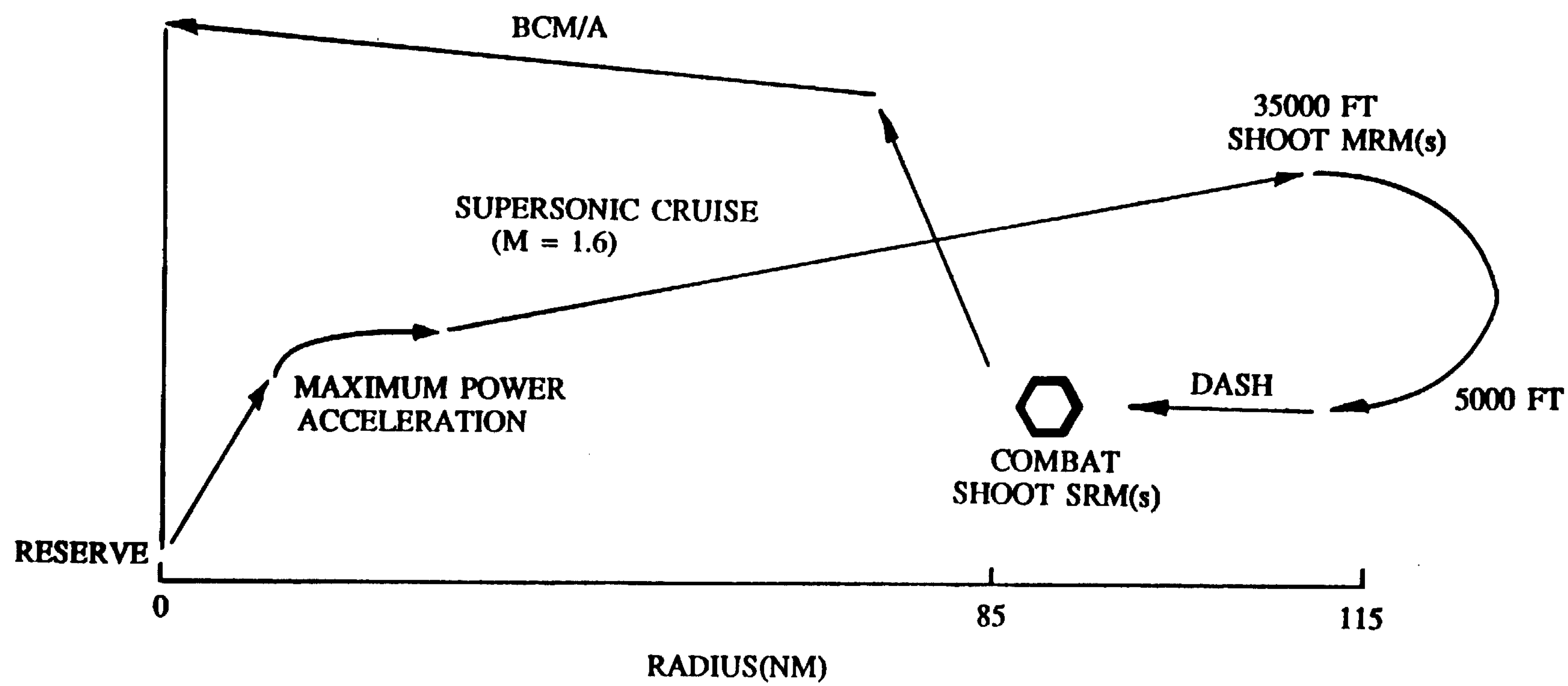

Figure 8.12 Mass Intercept Mission of the Monarch Aircraft

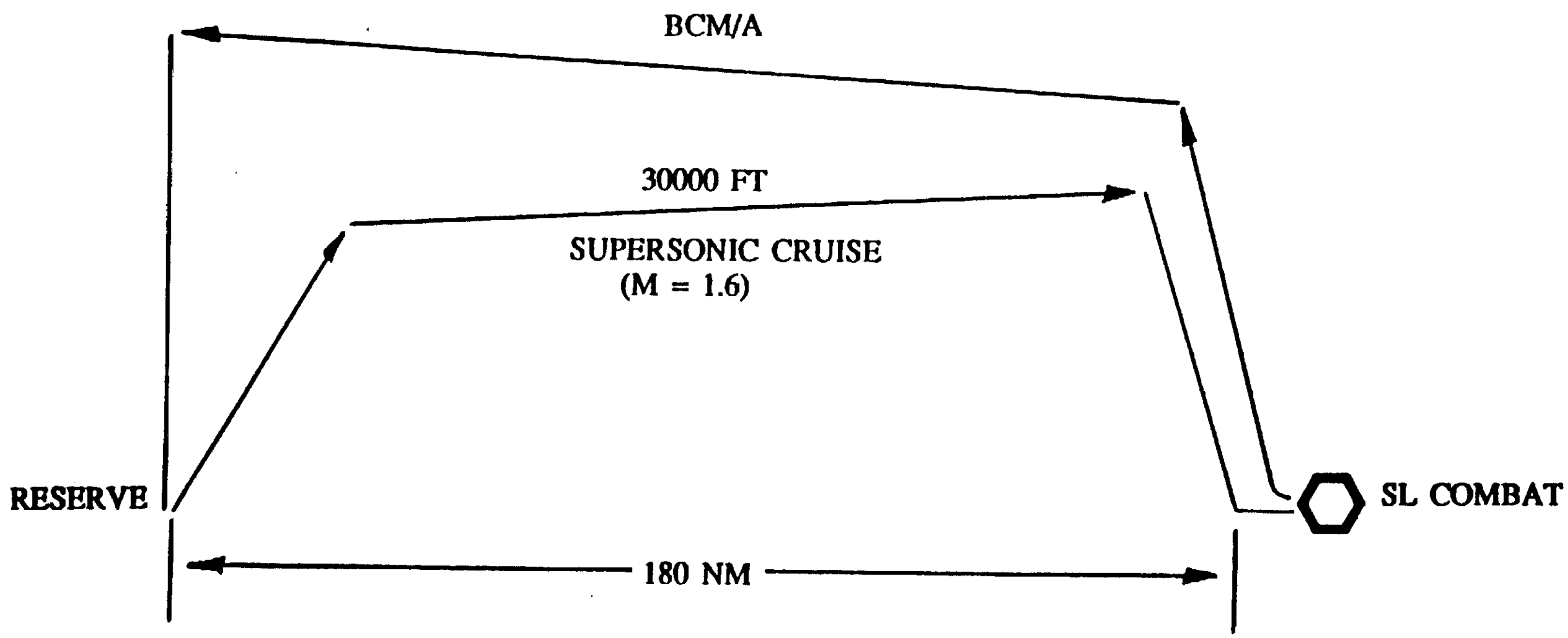

Figure 8.13 Transport/Helicopter Intercept Mission of the Monarch Aircraft

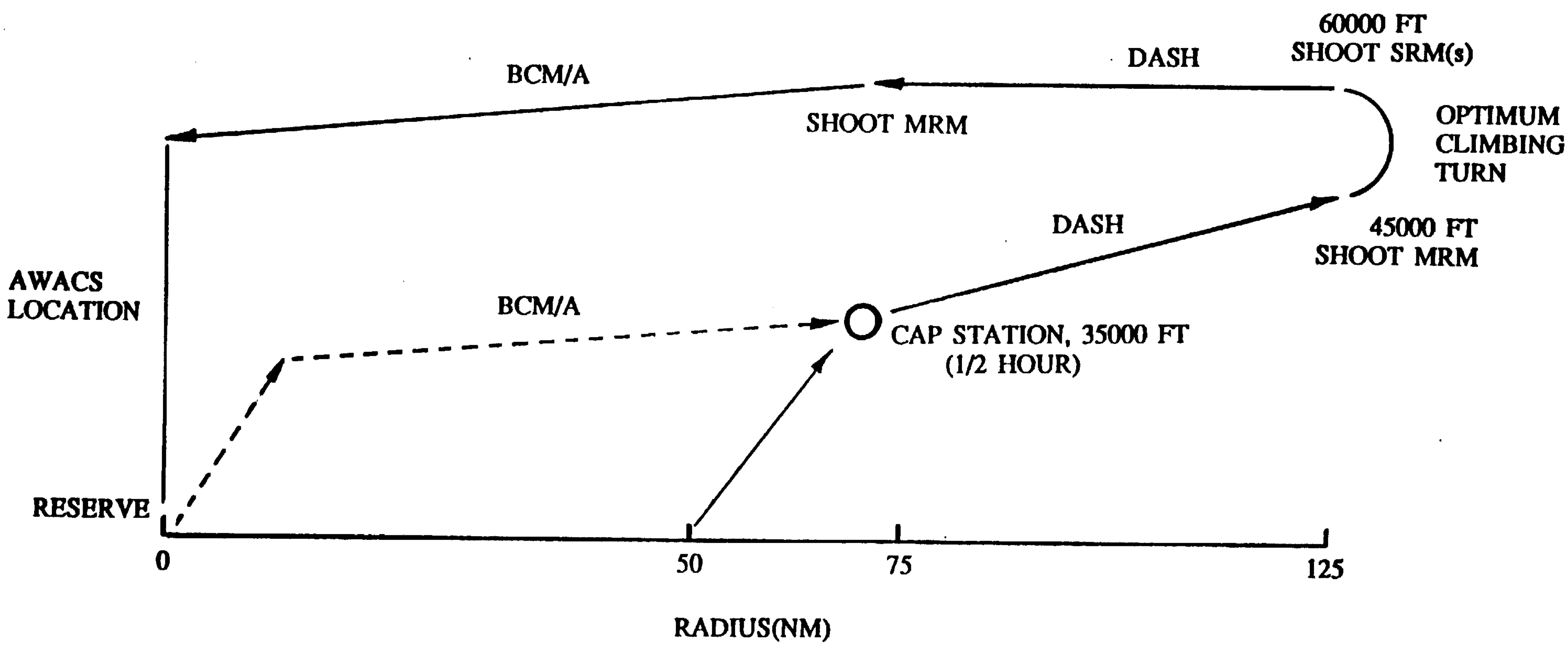

Figure 8.14 High Value Assest Protection (AWACS) Mission of the Monarch Aircraft

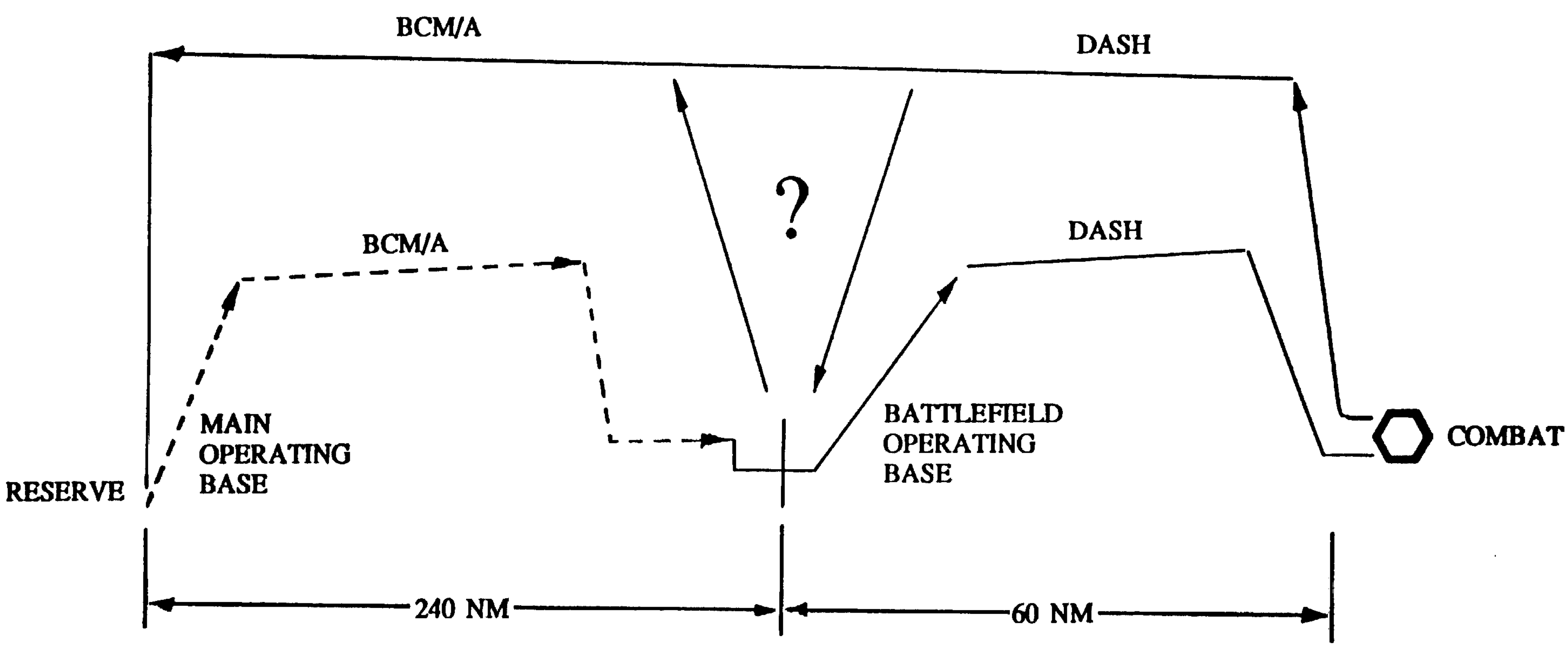

Figure 8.15 Two Stage Mission of the Monarch Aircraft

REFERENCES FOR CHAPTER 8

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8.6 Northrop Lecture Series to NASA Ames, NASA Ames Research Center, Moffet Field, CA, Summer 1989.

9. STABILITY AND CONTROL

The purpose of this chapter is to document the results of the stability and control analysis for the Monarch fighter. The following topics are covered in this chapter:

9.1 Flight Conditions

9.2 Trim Diagrams

9.3 Stability and Control Derivatives

9.4 Dynamic Stability and Control Analysis

9.4.1 Longitudinal

9.4.2 Lateral

9.4.3 Directional

9.5 Roll Performance

9.6 Inertia Coupling

9.7 Spin Departure

9.8 Low Level Ride Qualities

9.9 Vertical Tail/Rudder Removal Study

9.1 FLIGHT CONDITIONS

This section presents the selection of eight flight conditions which are representative of the flight envelope of the Monarch. A description and list of the parameters of each flight condition is also given.

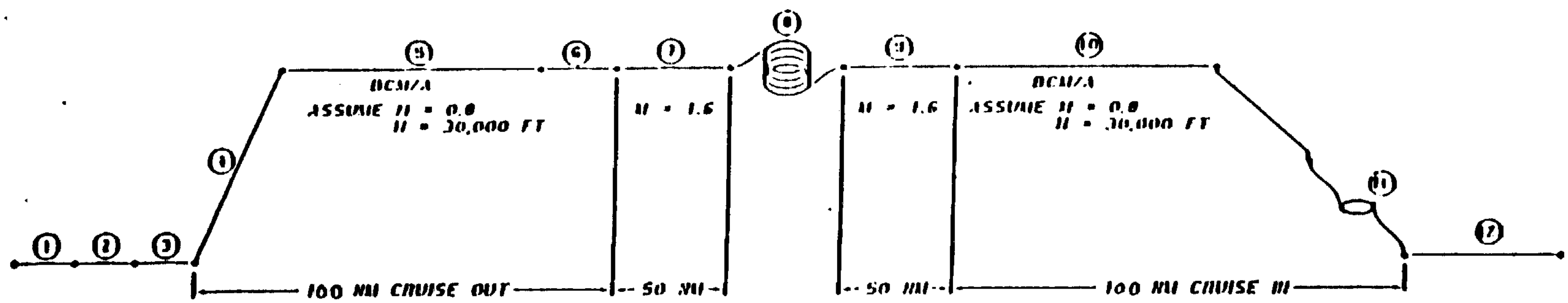
Eight flight conditions were chosen to represent the flight envelope of the Monarch fighter. They were chosen from the Counter-Air (CA) and Battlefield Air Interdiction (BAI) Mission profiles as depicted in Figures 9.1 and 9.2, respectively.

A description of the flight conditions follows:

FC 1: CA, Phase 3, Takeoff/Hover/Transition phase.

FC 2: BAI #1, Phase 6, Low altitude, high subsonic dash out to ordnance drop.

FC 3: CA, Phase 8, Subsonic performance point.



1. ENGINE START/WARM-UP
2. TAXI
3. STO
4. CLIMB
5. DCM/A CRUISE OUT
6. ACCELERATE TO SUPERSONIC CRUISE MACH NUMBER, M = 1.6
7. SUPERSONIC CRUISE OUT - M = 1.6, H = 30,000 FT
8. COMBAT (EXPEND HALF STORES) H = 30,000 FT
9. SUPERSONIC CRUISE IN - M = 1.6, H = 30,000 FT
10. DCM/A CRUISE IN
11. ONE MINUTE HOVER
12. LAND WITH 5% INTERNAL FUEL

Figure 9.1 Counter-Air Mission Profile for the Monarch Fighter

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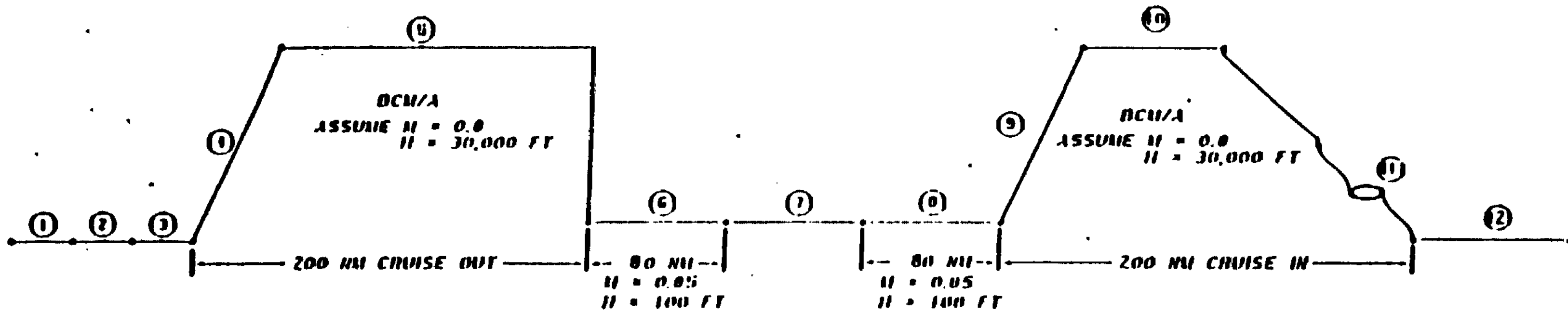

1. ENGINE START/WARM-UP
2. TAXI
3. STO
4. CLIMB TO CRUISE OUT ALTITUDE
5. BCM/A CRUISE OUT
6. DASH OUT
7. WEAPONS DROP HALF LOAD
8. DASH IN
9. CLIMB TO CRUISE IN ALTITUDE
10. BCM/A CRUISE IN
11. ONE MINUTE HOVER
12. LAND WITH 5% INTERNAL FUEL

Figure 9.2 Battlefield Interdiction Mission Profile for the Monarcg Fighter

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FC 4: CA, Phase 8, Subsonic maneuver, load factor = 4.25.

FC 5: CA, Phase 5, Subsonic maneuver, load factor = 6.0.

FC 6: CA, Phase 9, Supersonic performance point.

FC 7: CA, Phase 7, High altitude, supersonic cruise.

FC 8: BAI #2, Phase 5, High altitude subsonic cruise.

Table 9.1 summarizes the parameters of each flight condition.

Table 9.1 Flight Conditions for the Monarch Fighter

Flight Condition	Altitude	Mach Number	Load Factor
1	0 ft	0.20	1.0
2	100 ft	0.85	1.0
3	10,000 ft	0.90	1.0
4	15,000 ft	0.90	4.25
5	30,000 ft	0.90	6.0
6	30,000 ft	1.20	1.0
7	30,000 ft	1.60	1.0
8	40,000 ft	0.80	1.0

9.2 TRIM DIAGRAMS

This section presents the trim diagrams for the Monarch fighter. The method of Reference 9.1 was used in constructing the trim diagrams. Detailed calculations of the trim data are documented in Reference 9.2.

The airplane lift versus angle of attack curve and airplane lift versus pitching moment curve were calculated according to the methods of Reference 9.1. The curves were constructed for horizontal tail deflections ranging from -30 degrees to +30 degrees in ten degree increments.

The forward and aft c.g. travel lines and the horizontal tail stall loci form the boundaries of the trim triangle. Within the bounds of the trim triangle, the horizontal stabilator deflections necessary to trim the airplane for a range of lift coefficients are determined. For each flight condition the lift coefficient was determined , knowing the airplane weight, load factor, altitude and velocity. From the respective trim diagram it was determined if the airplane could be trimmed and, if so, what stabilator deflection was required.

At the end of Phase I design, the Monarch was designed with a canard for longitudinal control. The detailed stability analysis required for the development of the trim diagrams revealed that the canard design had an unacceptable margin of longitudinal instability. This led to the removal of the canard and the incorporation of a conventional tail aft stabilator into the design of the Monarch.

Originally a symmetric airfoil was selected for the horizontal stabilator design. The symmetric airfoil displayed a low stall angle of attack and made the airplane untrimmable in all flight conditions. A cambered 6% thick airfoil was incorporated to improve the tail stall characteristics. In addition, a full span fixed slat, similar to the stabilator design on the McDonnell F-4E Phantom II, is used. These changes provided adequate longitudinal control power throughout the c.g. ranges of all flight conditions. According to criteria found in Reference 9.3, the drag divergence Mach Number of the horizontal stabilator and the wing were determined. From this it was determined that the drag divergence Mach Number of the stabilator was higher than that of the wing. Therefore the horizontal stabilator will retain control power at high subsonic Mach Number when the flow over the wing becomes supersonic.

The trim diagrams for the Monarch fighter are shown in Figures 9.3 through 9.10 for the eight flight conditions. Where the center of gravity limits cut into the trimmable range of the aircraft, the fuel management system will keep the center of gravity from moving into these areas. This will keep the aircraft prevent from moving into untrimmable flight conditions.

The trim diagram for flight condition 1 (Figure 9.3) reflects the lift increments and corresponding pitching moments for a 40 degree trailing edge plain flap and 20 degree leading edge slat. The thrust from the lift engine and the main engine are balanced to augment the aerodynamic lift during takeoff as described in Chapter 11.

From the trim diagrams it was determined that the Monarch can be trimmed with reasonable stabilator deflections for all flight conditions. Table 9.2 lists the lift coefficient and the required stabilator deflection to trim for each flight condition.

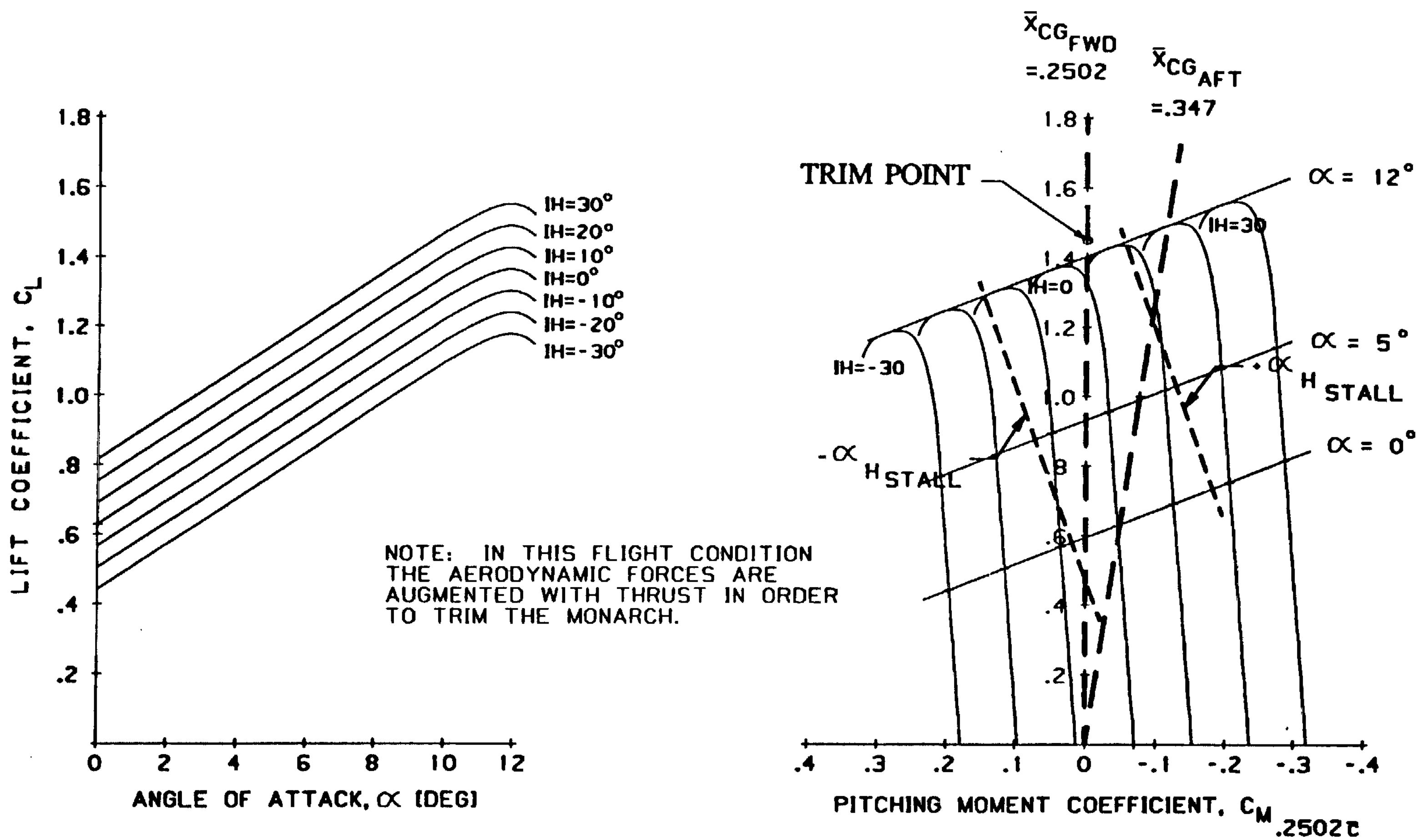

Figure 9.3 Trim Diagram for Flight Condition 1

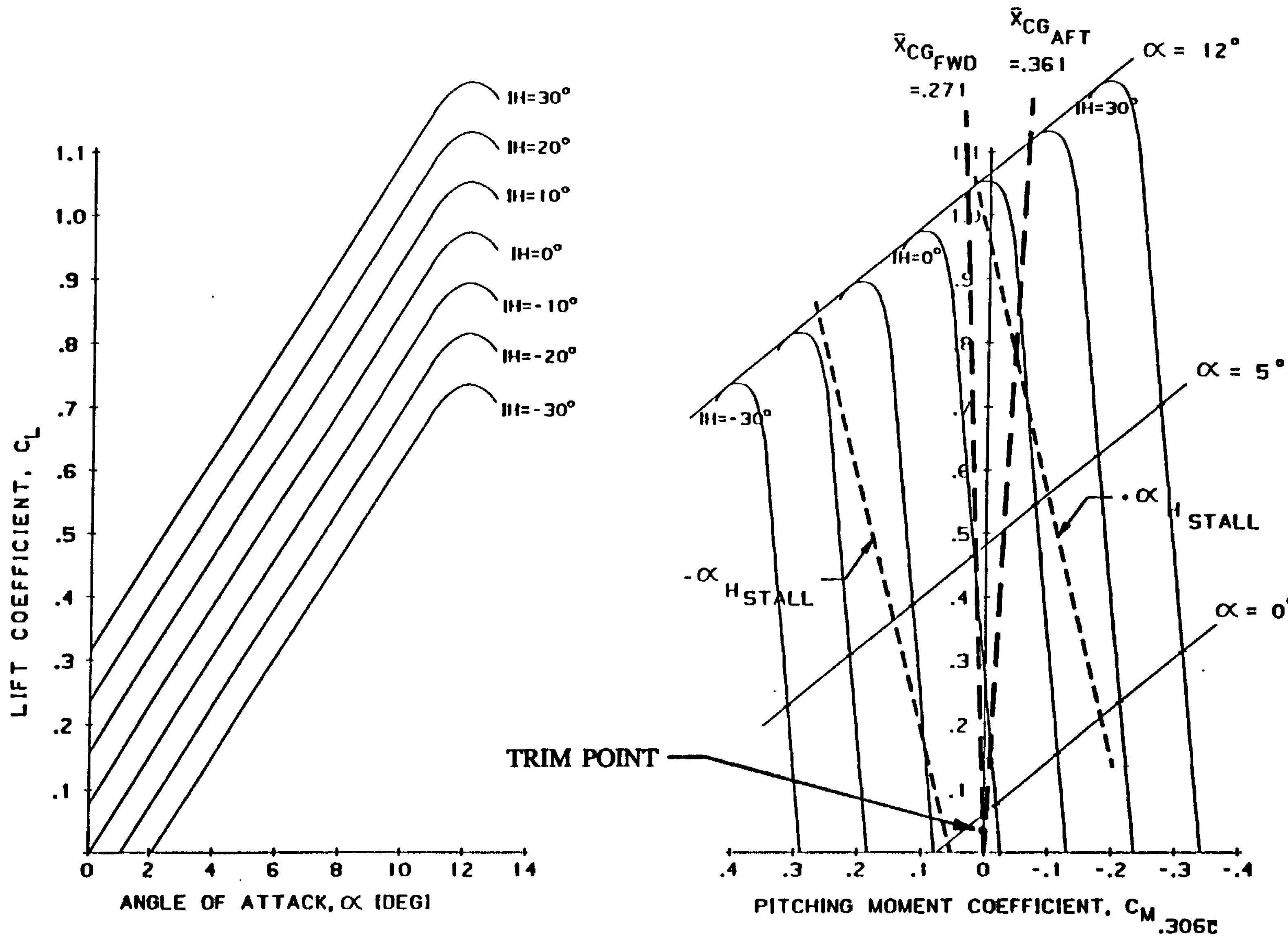

Figure 9.4 Trim Diagram for Flight Condition 2

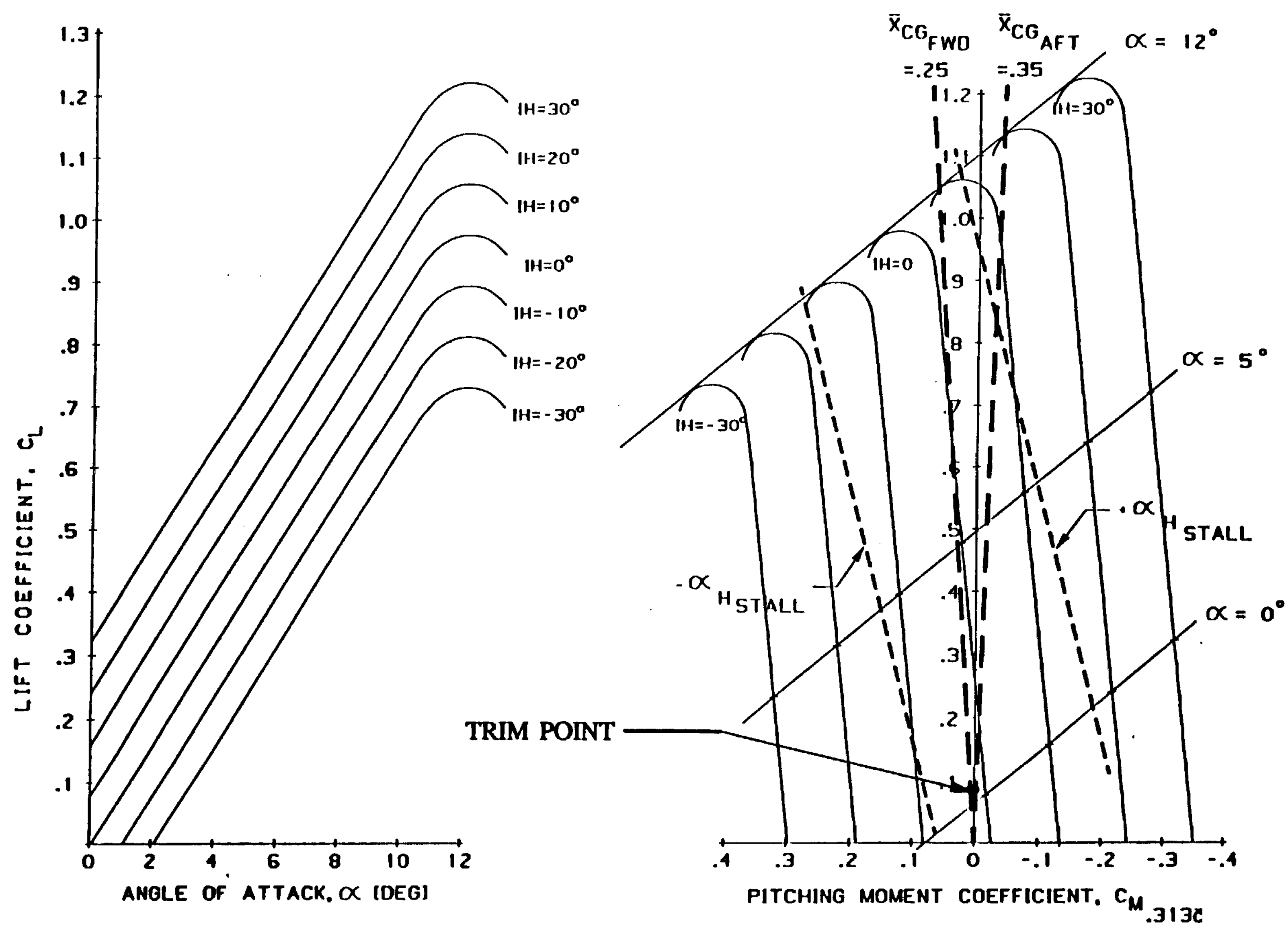

Figure 9.5 Trim Diagram for Flight Condition 3

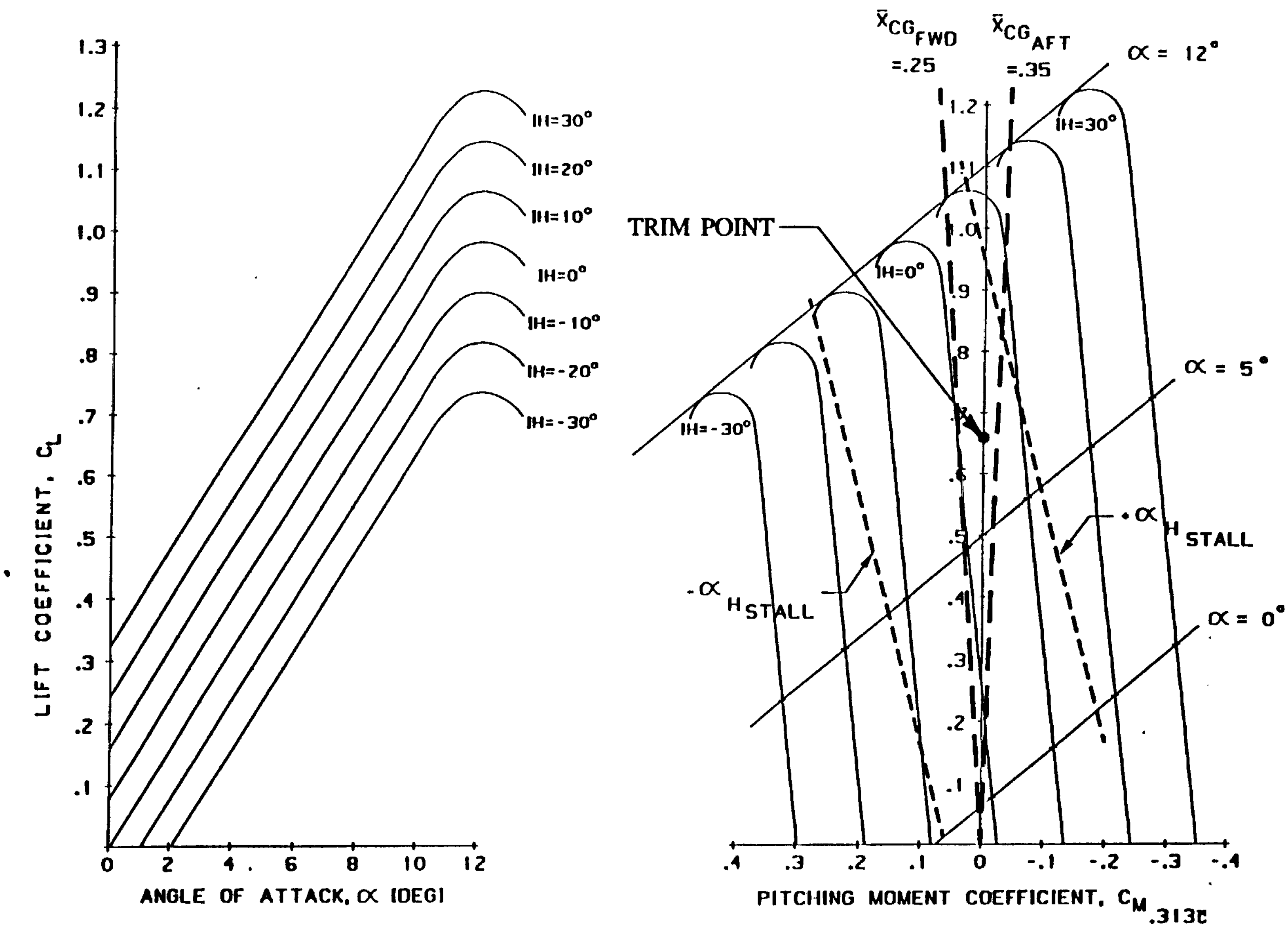

Figure 9.6 Trim Diagram for Flight Condition 4

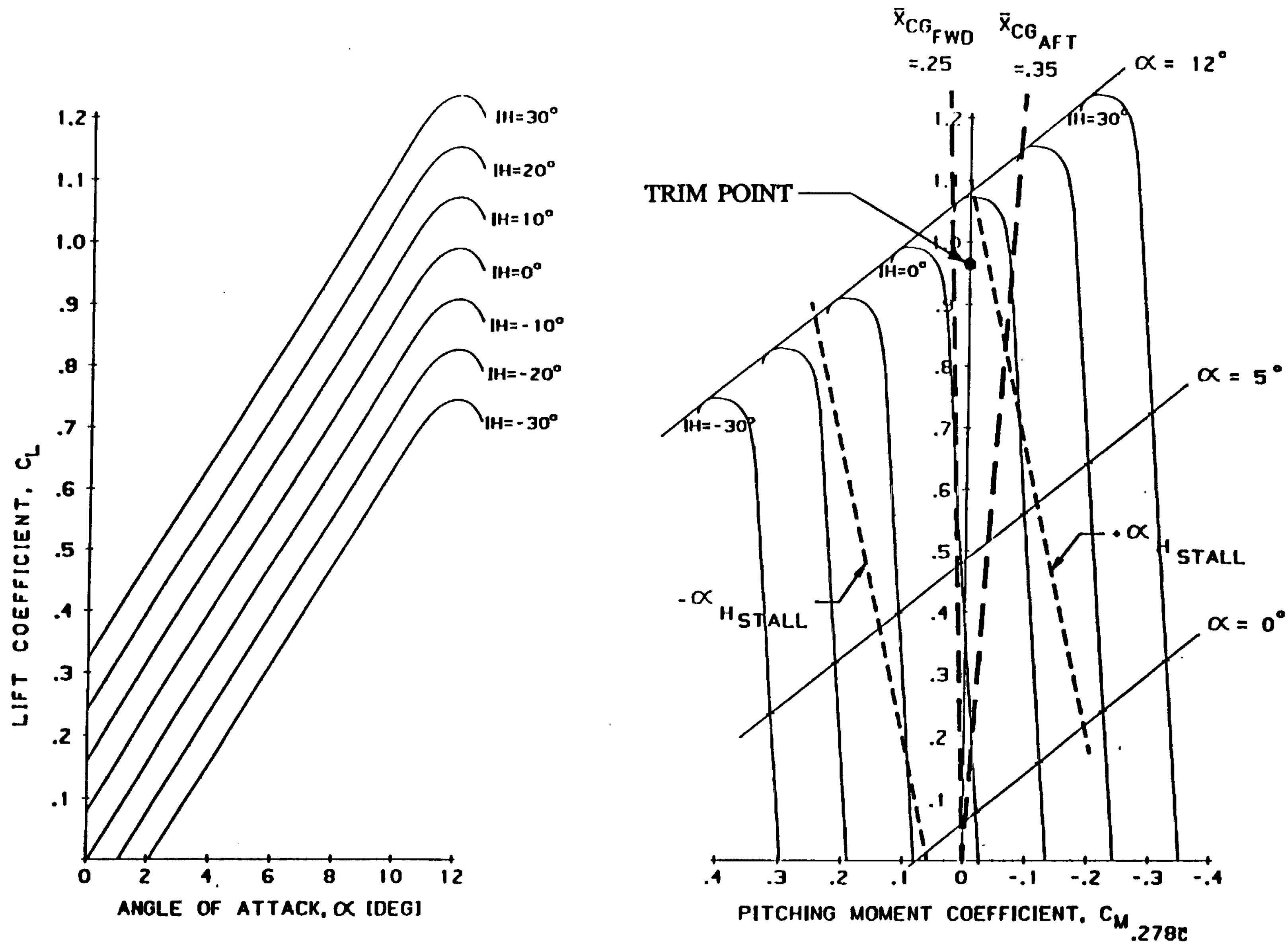

Figure 9.7 Trim Diagram for Flight Condition 5

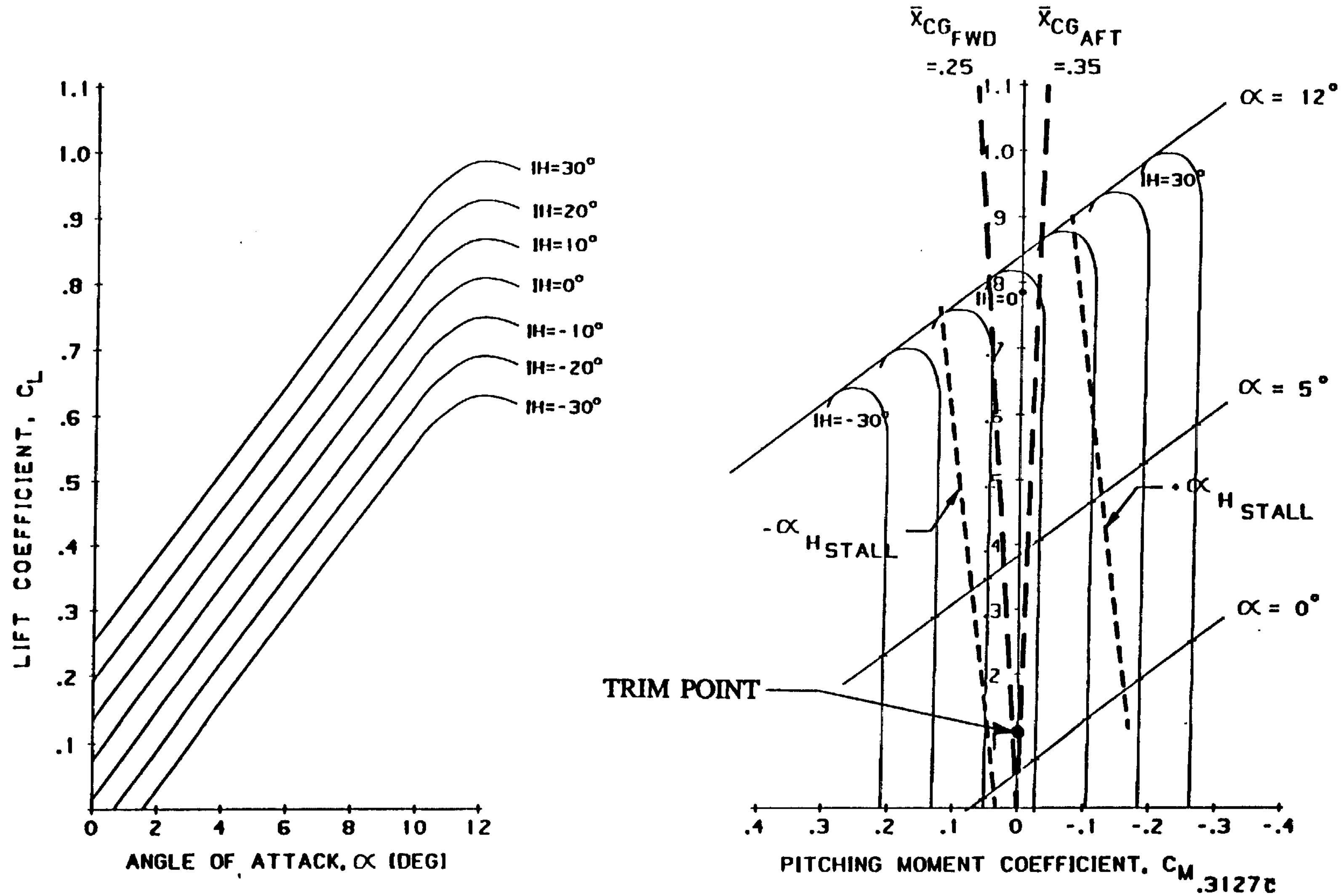

Figure 9.8 Trim Diagram for Flight Condition 6

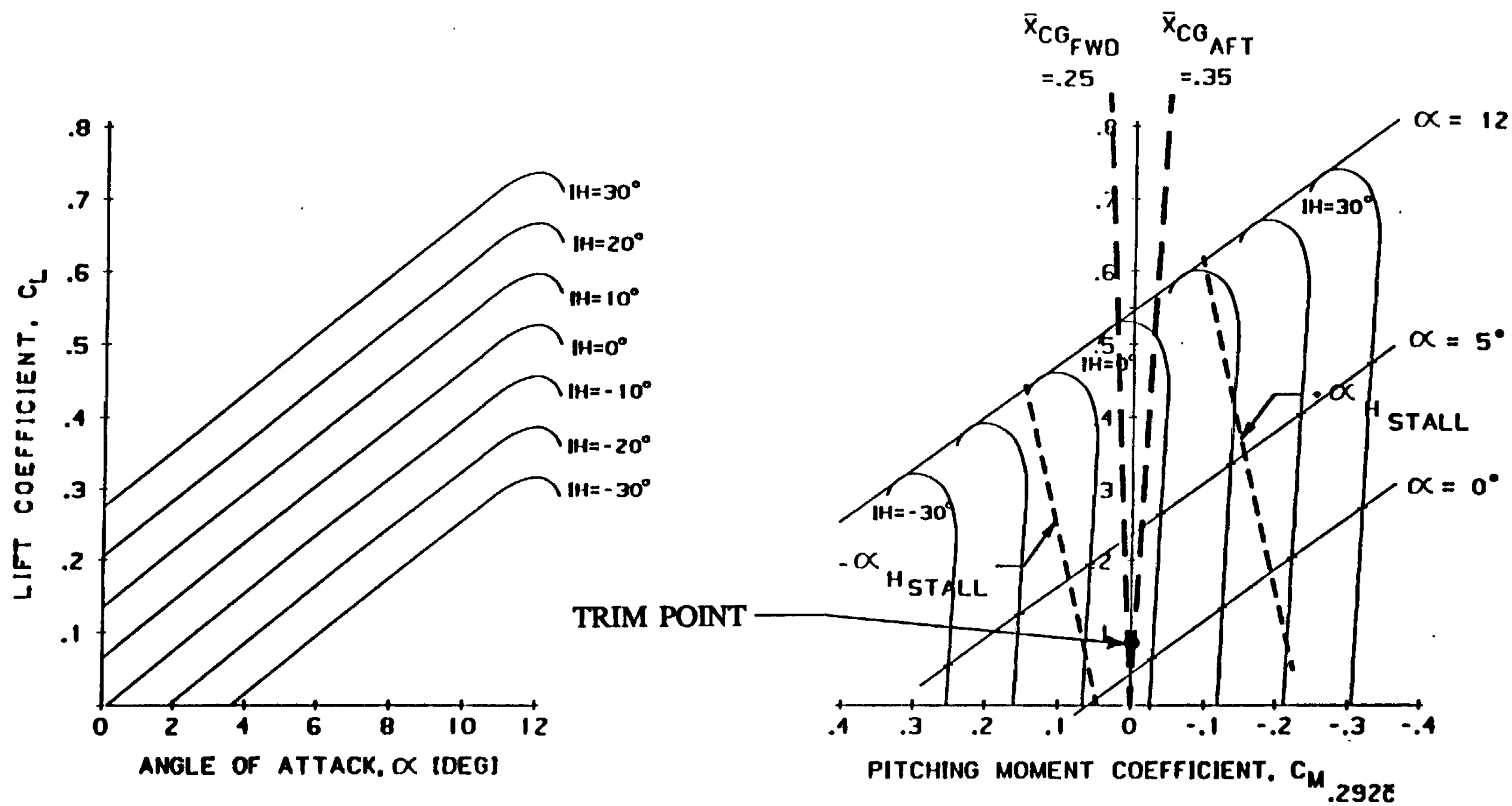

Figure 9.9 Trim Diagram for Flight Condition 7

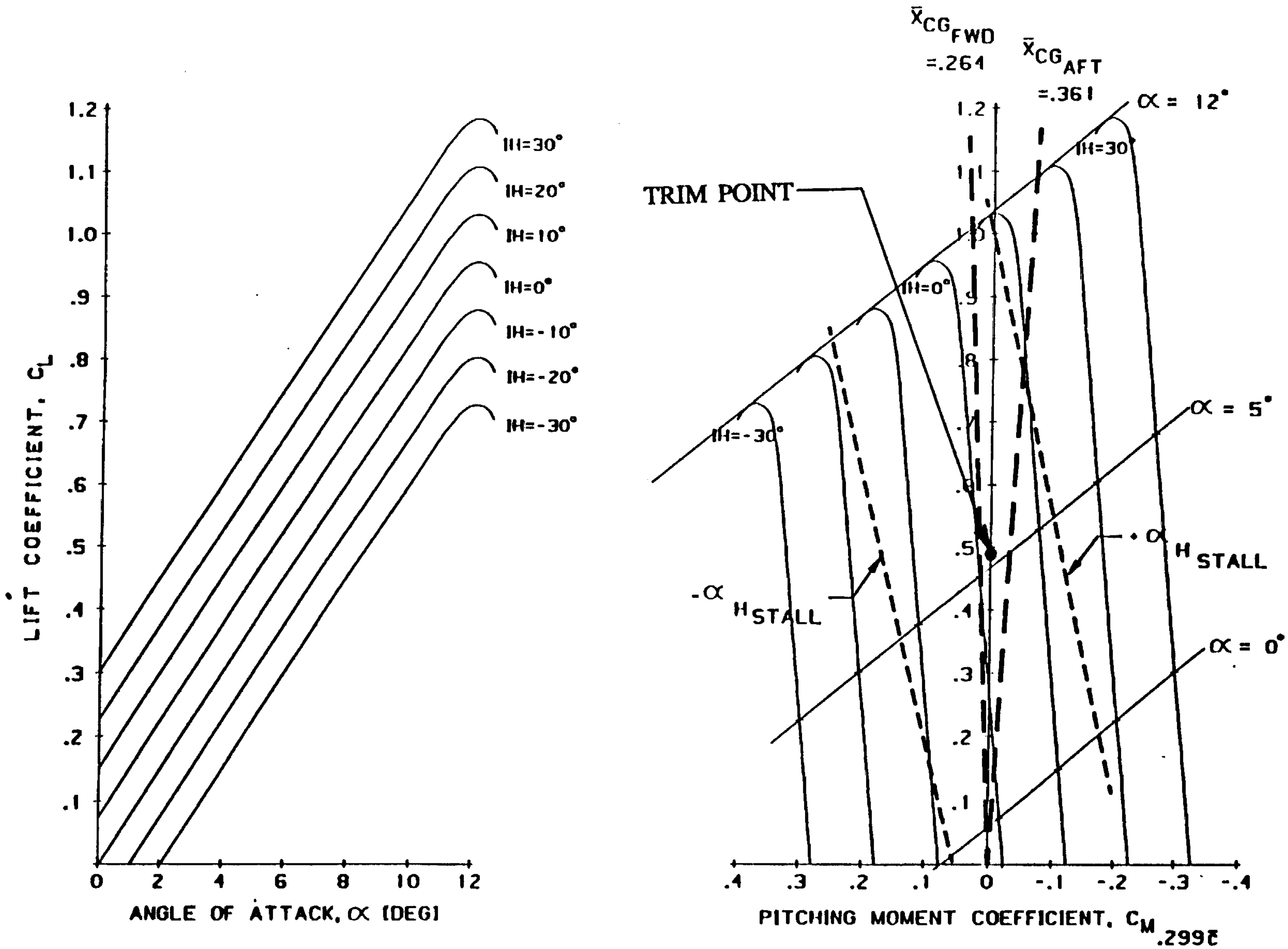

Figure 9.10 Trim Diagram for Flight Condition 8

Table 9.2 Required Stabilator Deflection to Trim

FC	Lift Coefficient	Stabilator Deflection, deg
1	1.495	0.0
2	0.076	-1.5
3	.083	-2.0
4	.661	3.7
5	.968	6.0
6	.111	-3.9
7	.069	-3.0
8	.489	3.0

9.3 STABILITY AND CONTROL DERIVATIVES

The stability and control derivatives for the Monarch fighter are presented in this section. The methods of References 9.1 and 9.4 were used to compute the derivatives for the eight flight conditions. Reference 9.2 documents the detailed calculations of the stability and control derivatives for the Monarch fighter.

The longitudinal, lateral-directional and thrust derivatives of the Monarch fighter for the eight flight conditions are presented in Tables 9.3 through 9.10. Also presented in these tables are the geometric and flight condition parameters required for the calculation of the dimensional derivatives.

The thrust derivatives were calculated with data obtained from the Pratt & Whitney engine deck (Reference 9.5).

Due to the fact that the rudder had been eliminated (Section 9.8), directional control was achieved using the 2-D main vectoring nozzle. Because the vanes for vectoring the thrust directionally are comparable to a control surface (6 sq. ft.), aerodynamic control derivatives for the vanes were calculated along with the control derivatives due to the thrust vectoring.

The longitudinal and lateral-directional derivatives of the Monarch fighter were compared to data of Reference 9.6. Reference 9.6 is a graphical presentation of the stability and control derivatives for supersonic fighters as a function of Mach Number. Figures 9.11 through 9.16 are copies of selected data from Reference 9.6 with the values for the Monarch fighter included. The values for the Monarch are illustrated with a circled dots in these figures.

9.4 DYNAMIC STABILITY CONTROL AND ANALYSIS

Due to time constraints, three flight conditions were chosen for dynamic stability and control analysis. These conditions were chosen to cover the least similar flight regimes. Flight condition 2 was chosen to represent a high speed, low altitude ground attack phase. Flight condition 4 is representative of air-to-air combat at a typical

Table 9.3 Stability and Control Derivatives for Flight Condition 1

Geometric and Flight Parameters:

S =	347.9	(sq ft)	I_xx_S =	13785	(slug*ft^2)
b =	33.67	(ft)	I_yy_S =	90780	(slug*ft^2)
c_bar =	12	(ft)	I_zz_S =	82563	(slug*ft^2)
mass =	954.8	(slugs)	I_xz_S =	-13839	(slug*ft^2)

Longitudinal:

C_D_1 =	0.2660
C_D_u =	0.0000
C_D_a =	1.1060
C_D_ih =	0.0000
C_T_x_1 =	0.2660
C_T_x_u =	-0.5091
C_M_T_1 =	0.0089
C_M_T_u =	-0.0177
C_M_T_a =	0.1024
C_L_1 =	1.4950
C_L_u =	0.0440
C_L_a =	3.7470
C_L_adot =	0.5182
C_L_q =	5.9700
C_L_ih =	0.3970
C_M_1 =	0.0647
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C_M_adot =	-0.5959
C_M_q =	-4.2060
C_M_ih =	-0.5280

Lateral-Directional:

C_y_B =	-0.4630	
C_y_p =	0.0130	
C_y_r =	0.2580	
C_y_dA =	0.0000	
C_y_dR =	0.0183	(aero)
C_y_dR =	0.3043	(thrust)
C_l_B =	-0.2394	
C_l_p =	-0.2155	
C_l_r =	0.3810	
C_l_dA =	0.4150	
C_l_dR =	-0.0029	(aero)
C_l_dR =	-0.0497	(thrust)
C_n_B =	-0.0034	
C_n_T_B =	0.0259	
C_n_p =	-0.0900	
C_n_r =	-0.2980	
C_n_dA =	-0.1630	
C_n_dR =	-0.0111	(aero)
C_n_dR =	-0.1904	(thrust)

Table 9.4 Stability and Control Derivatives for Flight Condition 2

Geometric and Flight Parameters:

S =	347.9	(sq ft)
b =	33.67	(ft)
c_bar =	12	(ft)
mass =	881.8	(slugs)

I_xx_S =	15370	(slug*ft^2)
I_yy_S =	88824	(slug*ft^2)
I_zz_S =	86851	(slug*ft^2)
I_xz_S =	1035	(slug*ft^2)

Longitudinal:

C_D_1 =	0.0241
C_D_u =	0.0850
C_D_a =	0.0621
C_D_ih =	0.0102
C_T_x_1 =	0.0241
C_T_x_u =	-0.0467
C_M_T_1 =	0.0008
C_M_T_u =	-0.0016
C_M_T_a =	0.0217
C_L_1 =	0.0760
C_L_u =	0.0820
C_L_a =	4.4380
C_L_adot =	0.6249
C_L_q =	9.2080
C_L_ih =	0.5040
C_M_1 =	0.0076
C_M_u =	-0.0057
C_M_a =	0.4420
C_M_adot =	-0.6838
C_M_q =	-6.9500
C_M_ih =	-0.6700

Lateral-Directional:

C_y_B =	-0.5030	
C_y_p =	-0.1150	
C_y_r =	0.3110	
C_y_dA =	0.0000	
C_y_dR =	0.0183	(aero)
C_y_dR =	0.0239	(thrust)
C_l_B =	-0.0708	
C_l_p =	-0.2840	
C_l_r =	0.1650	
C_l_dA =	0.2520	
C_l_dR =	-0.0006	(aero)
C_l_dR =	-0.0008	(thrust)
C_n_B =	0.0112	
C_n_T_B =	0.0056	
C_n_p =	0.0680	
C_n_r =	-0.3200	
C_n_dA =	-0.0050	
C_n_dR =	-0.0115	(aero)
C_n_dR =	-0.0150	(thrust)

Table 9.5 Stability and Control Derivatives for Flight Condition 3

Geometric and Flight Parameters:

S =	347.9	(sq ft)	I_xx_S =	9904	(slug*ft^2)
b =	33.67	(ft)	I_yy_S =	87959	(slug*ft^2)
c_bar =	12	(ft)	I_zz_S =	81040	(slug*ft^2)
mass =	744.7	(slugs)	I_xz_S =	820	(slug*ft^2)

Longitudinal:

C_D_1 =	0.0246
C_D_u =	0.1260
C_D_a =	0.0751
C_D_ih =	0.0147
C_T_x_1 =	0.0246
C_T_x_u =	-0.0429
C_M_T_1 =	0.0009
C_M_T_u =	-0.0015
C_M_T_a =	0.0307
C_L_1 =	0.0830
C_L_u =	0.1700
C_L_a =	4.5770
C_L_adot =	0.6393
C_L_q =	10.5510
C_L_ih =	0.5190
C_M_1 =	0.0090
C_M_u =	-0.0111
C_M_a =	0.4920
C_M_adot =	-0.6952
C_M_q =	-8.3400
C_M_ih =	-0.6900

Lateral-Directional:

C_y_B =	-0.6010	
C_y_p =	-0.1370	
C_y_r =	0.3720	
C_y_dA =	0.0000	
C_y_dR =	0.0183	(aero)
C_y_dR =	0.0533	(thrust)
C_l_B =	-0.0832	
C_l_p =	-0.2950	
C_l_r =	0.1780	
C_l_dA =	0.5110	
C_l_dR =	-0.0006	(aero)
C_l_dR =	-0.0018	(thrust)
C_n_B =	0.0665	
C_n_T_B =	0.0078	
C_n_p =	0.0790	
C_n_r =	-0.3140	
C_n_dA =	-0.0110	
C_n_dR =	-0.0114	(aero)
C_n_dR =	-0.0322	(thrust)

Table 9.6 Stability and Control Derivatives for Flight Condition 4

Geometric and Flight Parameters:

S =	347.9	(sq ft)
b =	33.67	(ft)
c_bar =	12	(ft)
mass =	744.7	(slugs)

I_xx_S =	10631	(slug*ft^2)
I_yy_S =	87959	(slug*ft^2)
I_zz_S =	80313	(slug*ft^2)
I_xz_S =	-7202	(slug*ft^2)

Longitudinal:

C_D_1 =	0.0688
C_D_u =	0.1260
C_D_a =	0.6064
C_D_ih =	0.0272
C_T_x_1 =	0.0688
C_T_x_u =	-0.1220
C_M_T_1 =	0.0024
C_M_T_u =	-0.0042
C_M_T_a =	0.0383
C_L_1 =	0.6610
C_L_u =	0.6500
C_L_a =	4.5770
C_L_adot =	0.6393
C_L_q =	10.5510
C_L_ih =	0.5190
C_M_1 =	0.0711
C_M_u =	-0.0879
C_M_a =	0.4920
C_M_adot =	-0.6952
C_M_q =	-8.3140
C_M_ih =	-0.6900

Lateral-Directional:

C_y_B =	-0.6010	
C_y_p =	-0.0900	
C_y_r =	0.3860	
C_y_dA =	0.0000	
C_y_dR =	0.0183	(aero)
C_y_dR =	0.0763	(thrust)
C_l_B =	-0.1600	
C_l_p =	-0.2950	
C_l_r =	0.1650	
C_l_dA =	5.1100	
C_l_dR =	-0.0019	(aero)
C_l_dR =	-0.008	(thrust)
C_n_B =	0.0802	
C_n_T_B =	0.0098	
C_n_p =	0.0200	
C_n_r =	-0.3670	
C_n_dA =	-0.0080	
C_n_dR =	-0.0010	(aero)
C_n_dR =	-0.0469	(thrust)

Table 9.7 Stability and Control Derivatives for Flight Condition 5

Geometric and Flight Parameters:

S =	347.9	(sq ft)
b =	33.67	(ft)
c_bar =	12	(ft)
mass =	876.8	(slugs)
I_xx_S =	9545	(slug*ft^2)
I_yy_S =	90165	(slug*ft^2)
I_zz_S =	987	(slug*ft^2)
I_xz_S =	3283	(slug*ft^2)

Longitudinal:

C_D_1 =	0.1292
C_D_u =	0.1260
C_D_a =	0.9840
C_D_ih =	0.0441
C_T_x_1 =	0.1292
C_T_x_u =	-0.1867
C_M_T_1 =	0.0045
C_M_T_u =	-0.0065
C_M_T_a =	0.0728
C_L_1 =	0.9680
C_L_u =	0.0450
C_L_a =	4.5770
C_L_adot =	0.6598
C_L_q =	10.5510
C_L_ih =	0.5190
C_M_1 =	0.0698
C_M_u =	-0.1287
C_M_a =	0.3300
C_M_adot =	-0.7430
C_M_q =	-8.3140
C_M_ih =	-0.6900

Lateral-Directional:

C_y_B =	-0.6010	
C_y_p =	-0.0650	
C_y_r =	0.3910	
C_y_dA =	0.0000	
C_y_dR =	0.0183	(aero)
C_y_dR =	0.1385	(thrust)
C_l_B =	-0.2050	
C_l_p =	-0.2950	
C_l_r =	0.1200	
C_l_dA =	0.5110	
C_l_dR =	-0.0027	(aero)
C_l_dR =	-0.0204	(thrust)
C_n_B =	0.0941	
C_n_T_B =	0.0861	
C_n_p =	-0.0140	
C_n_r =	-0.4000	
C_n_dA =	-0.1290	
C_n_dR =	-0.0111	(aero)
C_n_dR =	-0.0858	(thrust)

Table 9.8 Stability and Control Derivatives for Flight Condition 6

Geometric and Flight Parameters:

S =	347.9	(sq ft)
b =	33.67	(ft)
c_bar =	12	(ft)
mass =	761.9	(slugs)
I_xx_S =	6694	(slug*ft^2)
I_yy_S =	88352	(slug*ft^2)
I_zz_S =	78707	(slug*ft^2)
I_xz_S =	486	(slug*ft^2)

Longitudinal:

C_D_1 =	0.0416
C_D_u =	-0.1200
C_D_a =	0.0817
C_D_ih =	0.0181
C_T_x_1 =	0.0416
C_T_x_u =	-0.0303
C_M_T_1 =	0.0014
C_M_T_u =	-0.0011
C_M_T_a =	0.0524
C_L_1 =	0.1110
C_L_u =	-0.0500
C_L_a =	3.7980
C_L_adot =	0.3244
C_L_q =	10.1420
C_L_ih =	0.3990
C_M_1 =	-0.0011
C_M_u =	-0.0154
C_M_a =	-0.0390
C_M_adot =	-0.3764
C_M_q =	-6.8550
C_M_ih =	-0.4510

Lateral-Directional:

C_y_B =	-0.6260	
C_y_p =	-0.1390	
C_y_r =	0.4130	
C_y_dA =	0.0000	
C_y_dR =	0.0183	(aero)
C_y_dR =	0.0453	(thrust)
C_l_B =	-0.0885	
C_l_p =	-0.3350	
C_l_r =	0.1630	
C_l_dA =	0.1630	
C_l_dR =	-0.0007	(aero)
C_l_dR =	-0.0018	(thrust)
C_n_B =	0.1048	
C_n_T_B =	0.0133	
C_n_p =	0.0770	
C_n_r =	-0.4160	
C_n_dA =	-0.0047	
C_n_dR =	-0.0114	(aero)
C_n_dR =	-0.0282	(thrust)

Table 9.9 Stability and Control Derivatives for Flight Condition 7

Geometric and Flight Parameters:

S =	347.9	(sq ft)
b =	33.67	(ft)
c_bar =	12	(ft)
mass =	845.9	(slugs)

I_xx_S =	10152	(slug*ft^2)
I_yy_S =	8967	(slug*ft^2)
I_zz_S =	83546	(slug*ft^2)
I_xz_S =	410	(slug*ft^2)

Longitudinal:

C_D_1 =	0.0408
C_D_u =	0.0000
C_D_a =	0.0300
C_D_ih =	0.0084
C_T_x_1 =	0.0408
C_T_x_u =	-0.0420
C_M_T_1 =	0.0014
C_M_T_u =	-0.0015
C_M_T_a =	0.0329
C_L_1 =	0.0690
C_L_u =	-0.1550
C_L_a =	2.3780
C_L_adot =	0.2569
C_L_q =	5.0180
C_L_ih =	0.4440
C_M_1 =	-0.0033
C_M_u =	0.0000
C_M_a =	-0.1130
C_M_adot =	-0.3034
C_M_q =	-3.6780
C_M_ih =	-0.5110

Lateral-Directional:

C_y_B =	-0.4250	
C_y_p =	-0.0940	
C_y_r =	0.2800	
C_y_dA =	0.0000	
C_y_dR =	0.0183	(aero)
C_y_dR =	0.0428	(thrust)
C_l_B =	-0.0590	
C_l_p =	-0.2750	
C_l_r =	0.1530	
C_l_dA =	0.0970	
C_l_dR =	-0.0007	(aero)
C_l_dR =	-0.0017	(thrust)
C_n_B =	-0.0083	
C_n_T_B =	0.0080	
C_n_p =	0.0560	
C_n_r =	-0.4320	
C_n_dA =	-0.0018	
C_n_dR =	-0.0118	(aero)
C_n_dR =	-0.0270	(thrust)

Table 9.10 Stability and Control Derivatives for Flight Condition 8

Geometric and Flight Parameters:

S =	347.9	(sq ft)
b =	33.67	(ft)
c_bar =	12	(ft)
mass =	929.1	(slugs)
I_xx_S =	13169	(slug*ft^2)
I_yy_S =	88119	(slug*ft^2)
I_zz_S =	83378	(slug*ft^2)
I_xz_S =	-5241	(slug*ft^2)

Longitudinal:

C_D_1 =	0.5350
C_D_u =	0.0510
C_D_a =	0.4770
C_D_ih =	0.0189
C_T_x_1 =	0.5350
C_T_x_u =	-0.0188
C_M_T_1 =	0.0018
C_M_T_u =	-0.0007
C_M_T_a =	0.1282
C_L_1 =	0.4890
C_L_u =	0.0200
C_L_a =	4.3190
C_L_adot =	0.6059
C_L_q =	8.4730
C_L_ih =	0.4850
C_M_1 =	0.0452
C_M_u =	-0.0180
C_M_a =	0.3990
C_M_adot =	-0.6672
C_M_q =	-6.2550
C_M_ih =	-0.6460

Lateral-Directional:

C_y_B =	-0.4910	
C_y_p =	-0.0840	
C_y_r =	0.3120	
C_y_dA =	0.0000	
C_y_dR =	0.0183	(aero)
C_y_dR =	0.0536	(thrust)
C_l_B =	-0.1263	
C_l_p =	-0.2690	
C_l_r =	0.2100	
C_l_dA =	0.3990	
C_l_dR =	-0.0016	(aero)
C_l_dR =	-0.0046	(thrust)
C_n_B =	0.0436	
C_n_T_B =	0.0328	
C_n_p =	0.0300	
C_n_r =	-0.2950	
C_n_dA =	-0.0431	
C_n_dR =	-0.0116	(aero)
C_n_dR =	-0.0334	(thrust)

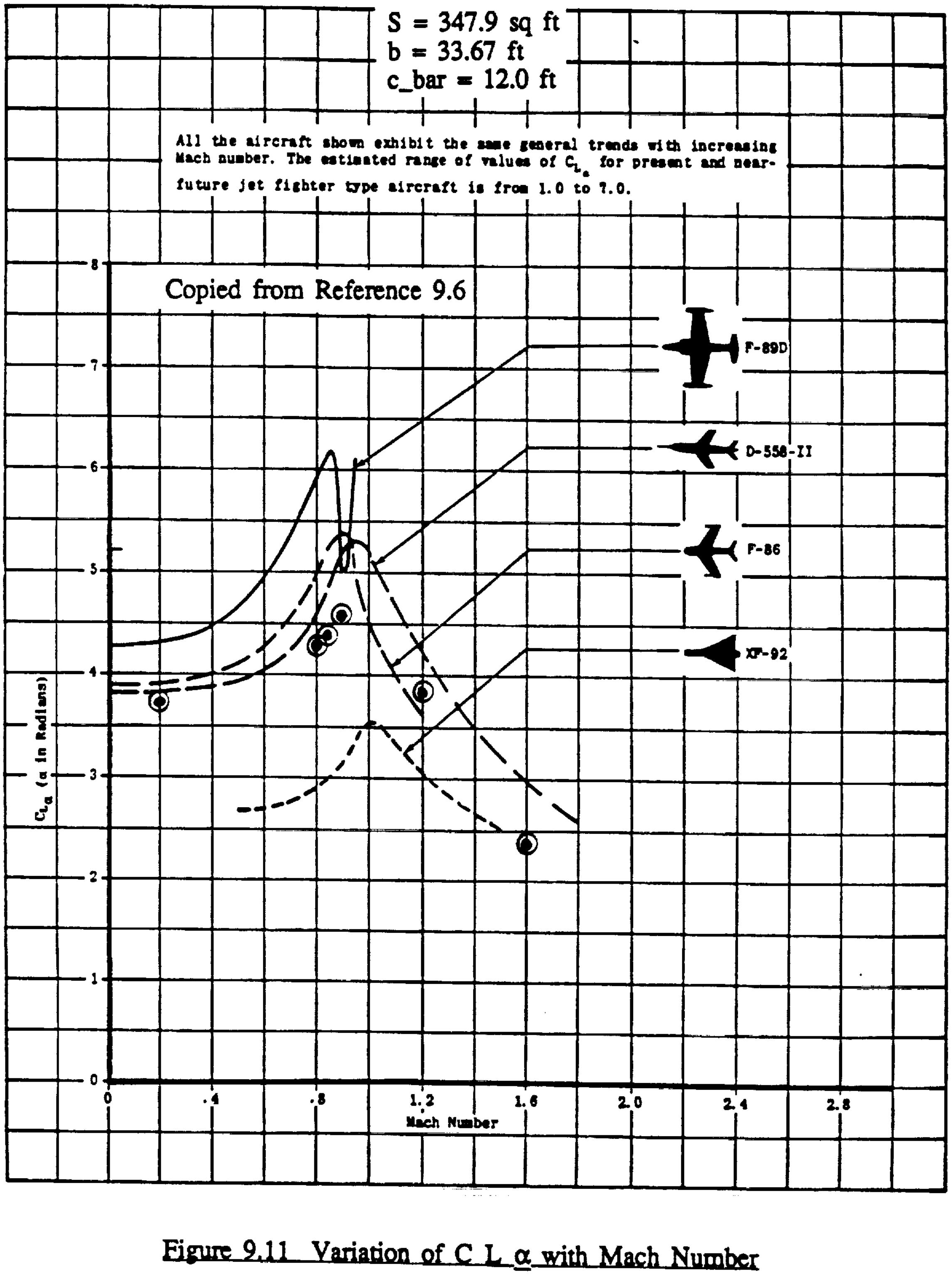

Figure 9.11 Variation of C L α with Mach Number

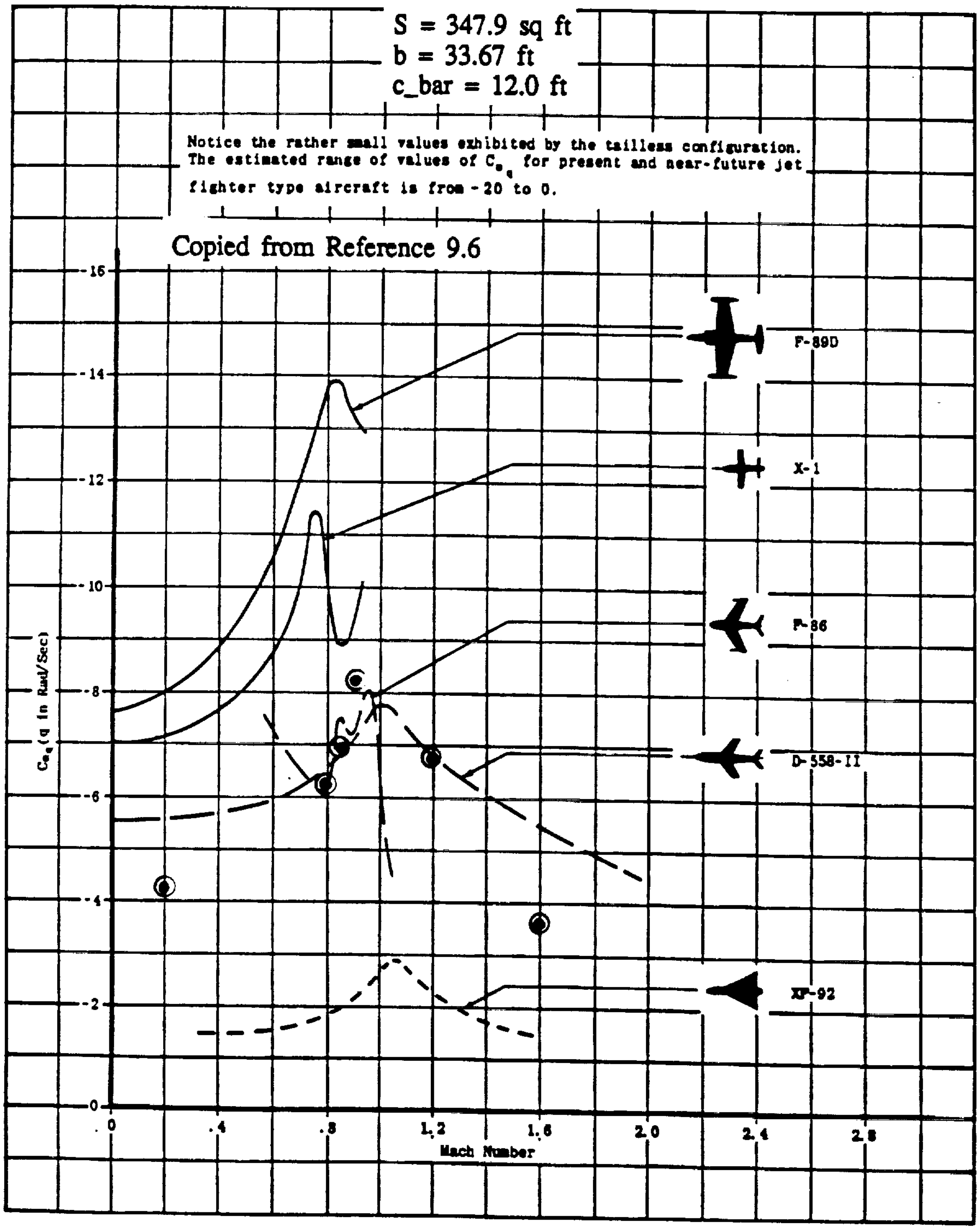

Figure 9.12 Variation of C M q with Mach Number

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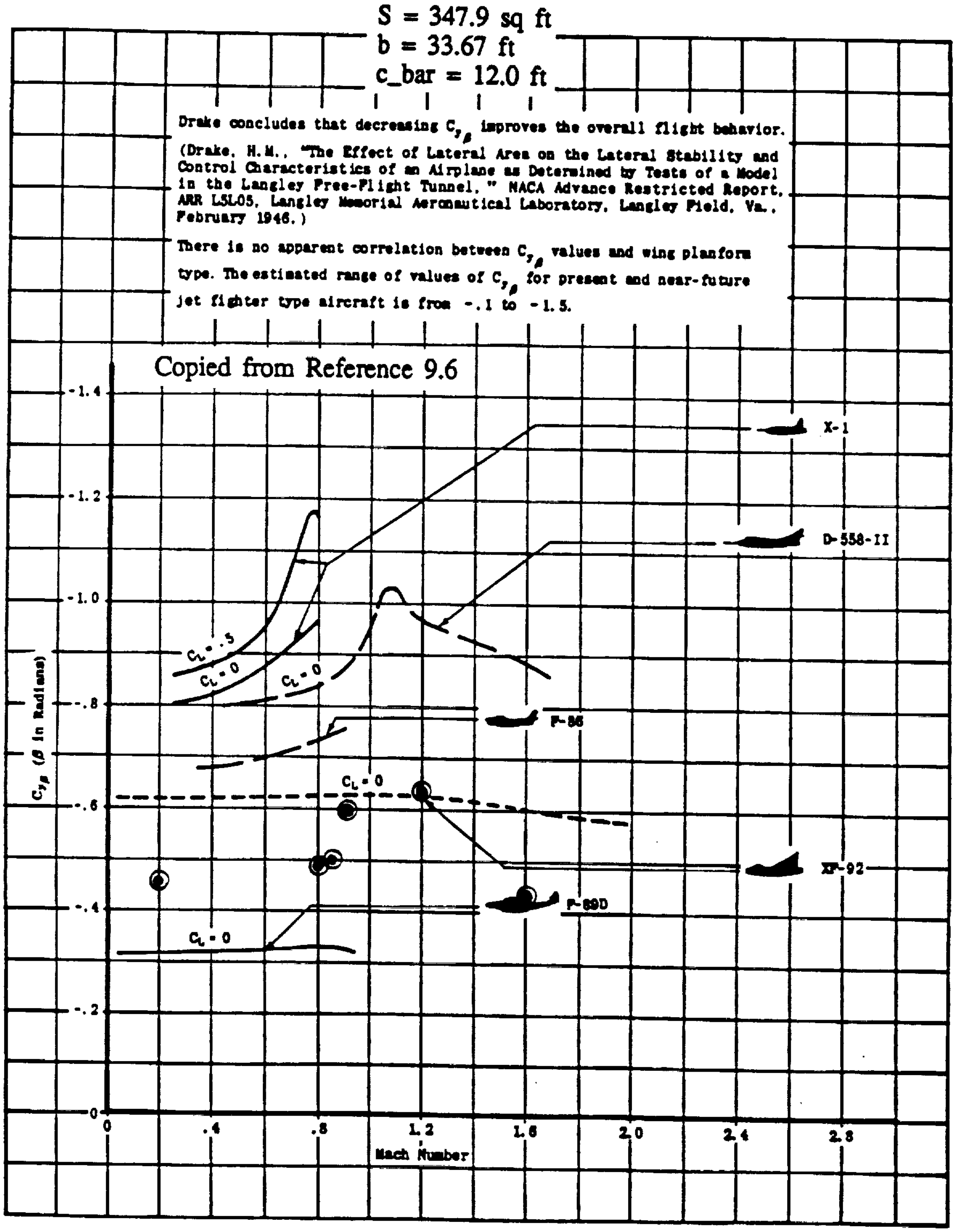

Figure 9.13 Variation of C y ß with Mach Number

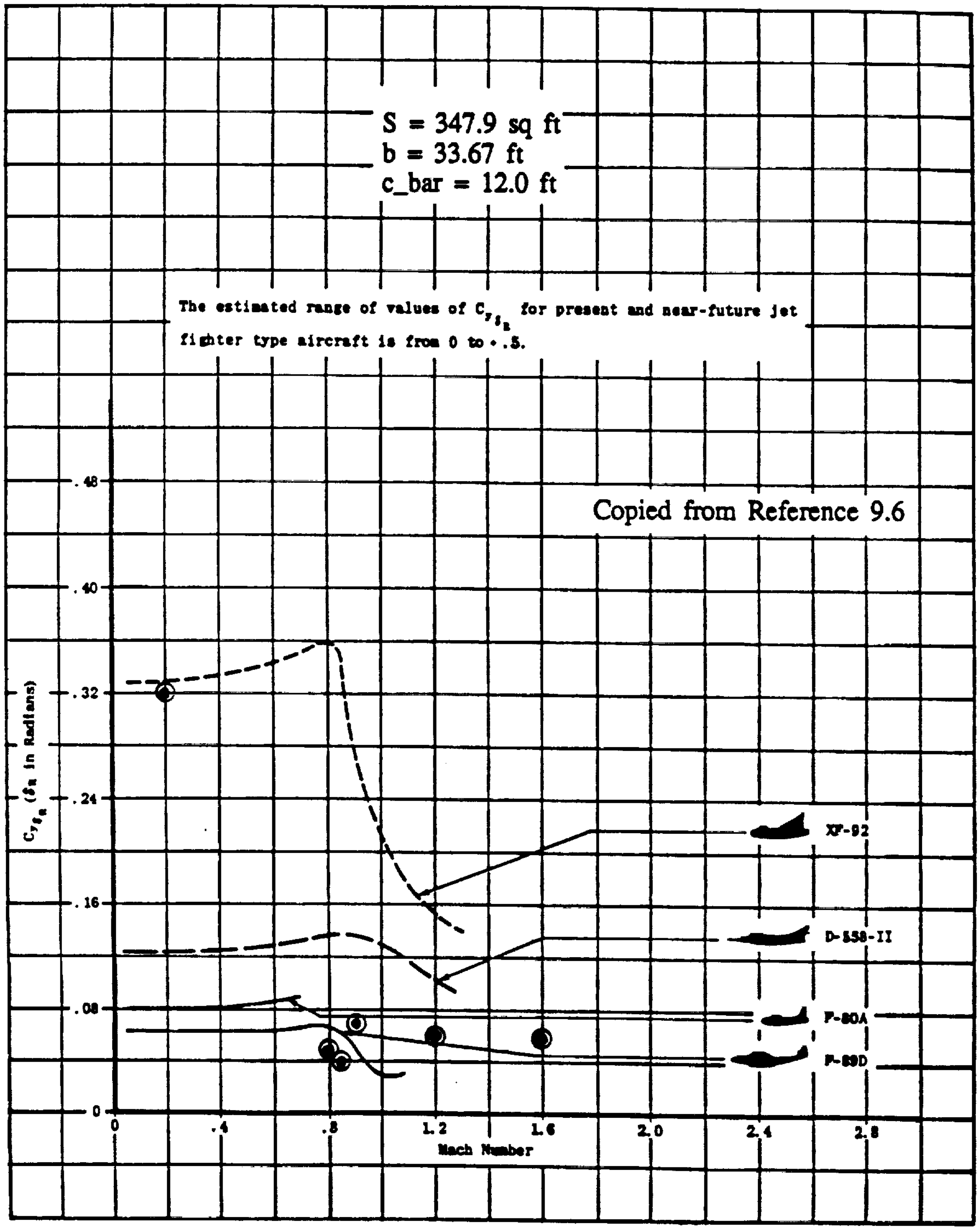

Figure 9.14 Variation of C y δR with Mach Number

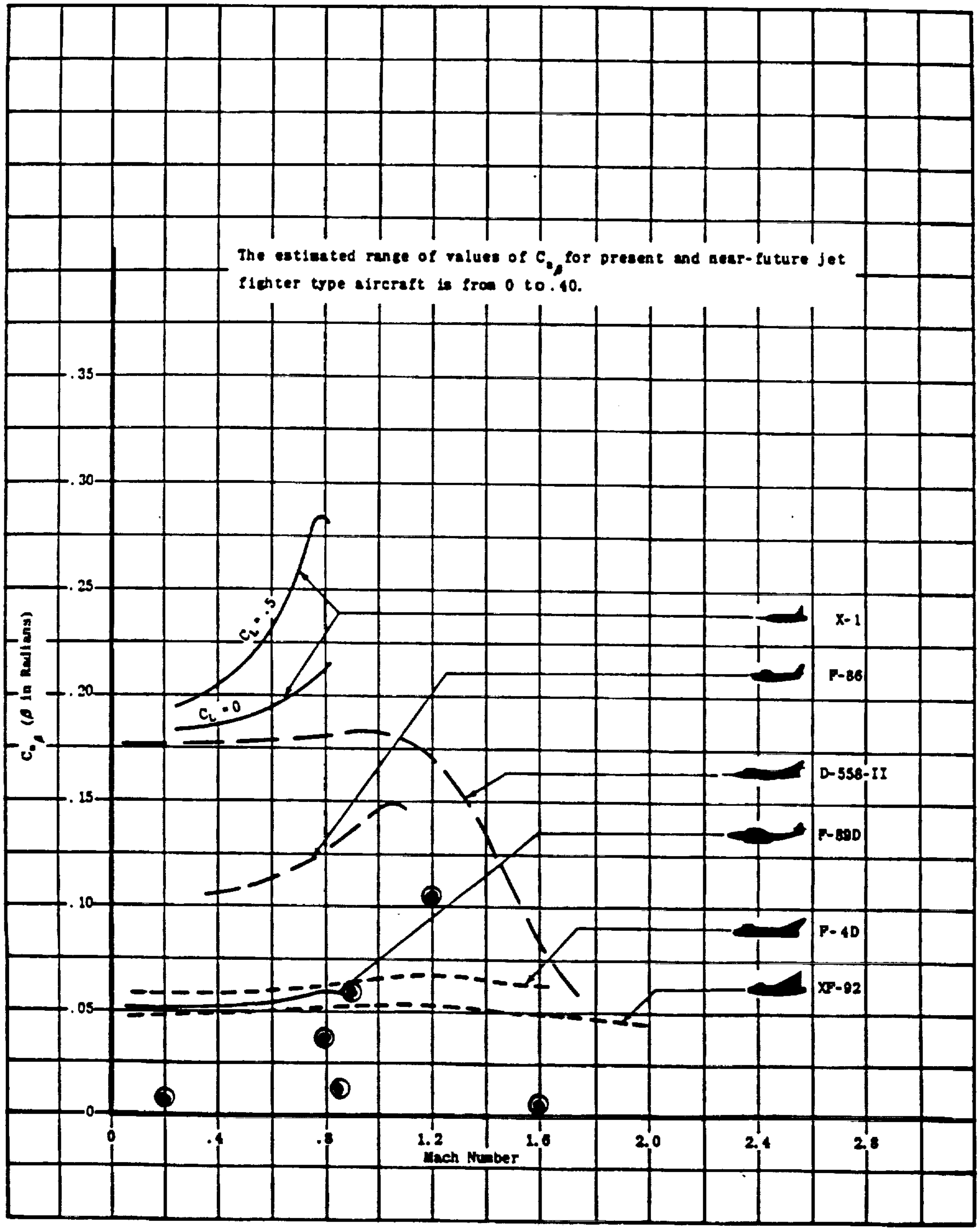

Figure 9.15 Variation of C n β with Mach Number

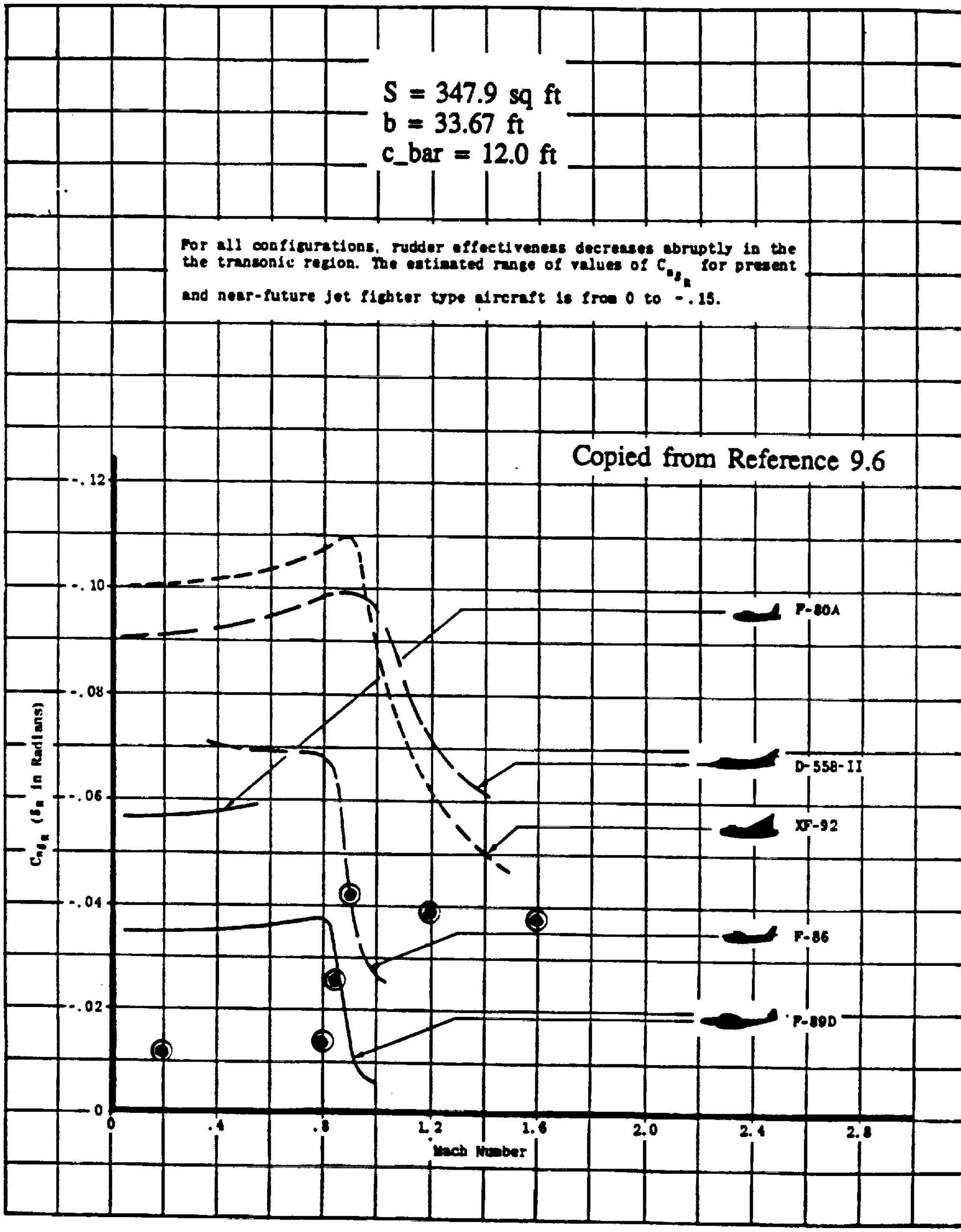

Figure 9.16 Variation of C n δR with Mach Number

engagement altitude. Flight condition 7 represents the aircraft during a typical supersonic cruise.

A digital control stability analysis was done for each aircraft axis. A generic z-plane root locus showing lines of constant damping and lines of constant zeta-omega n appears in Figure 9.17. These boundaries will be used to designate target areas in the z-plane in the sections that follow.

9.4.1 Longitudinal

The unaugmented longitudinal dynamic stability characteristics of the Monarch appear in Table 9.11. As indicated by these data, the aircraft has at least one unstable characteristic for every flight condition. The longitudinal dynamic characteristics required for Level 1 handling qualities by MIL-F-8785C (as per Reference 9.7) appear in Table 9.12. The calculations required for the short period frequency requirements appear in Appendix 3. Figures 9.18-9.20 show the open loop root loci in the z-plane, including a target area where the short period poles of the system must be placed to achieve MIL-F-8785C level 1 handling qualities.

Pitch rate feedback was used to stabilize the aircraft. Compensating equations were chosen so that the original poles of the open loop system would be cancelled by directly placing a zero on the calculated pole location. New poles were placed in locations in the z-plane that would give the Monarch level 1 flying qualities. A summary of the compensation equations appears in Table 9.13. The calculations that determined these locations appears in Appendix 3. A sampling rate of 100 cycles per second was assumed from Reference 9.8. The block diagram of the pitch SAS appears in Figure 9.21. PC MATLAB was used to determine the root locus of the discrete system with complete compensation.

Figures 9.22-9.24 show the root loci for the longitudinal closed loop system for the three flight conditions. An enlargement of the short period pole location has ben included to show its placement. As the only phugoid requirement specifies a damping ratio greater than 0.04, the phugoid roots were relocated on the stable portion of the real axis for an equivalent damping ratio of one.

Sampling Frequency = T_s = 0.01/sec

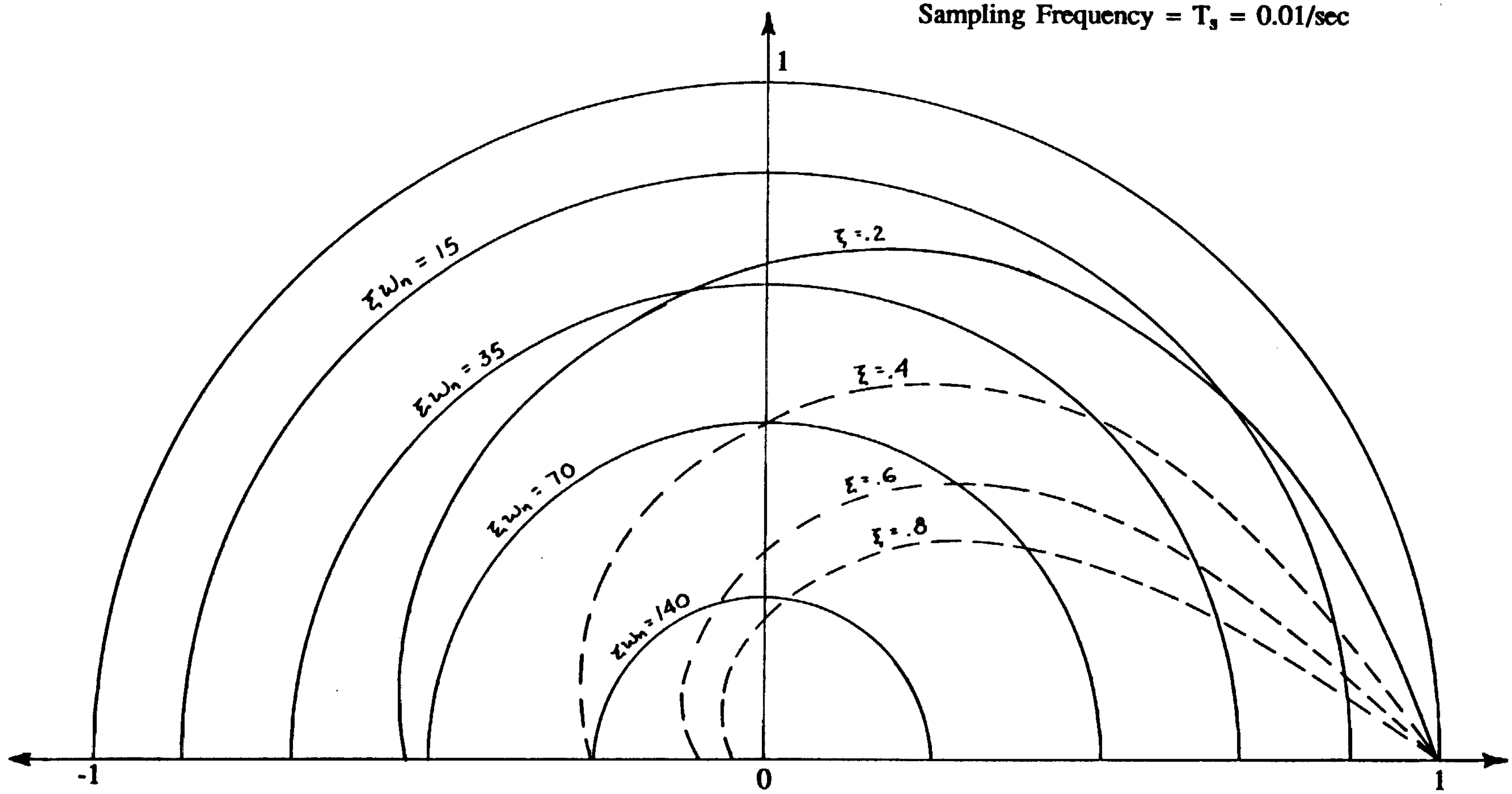

Figure 9.17 - Z-Plane Diagram

Table 9.11 - Unaugmented Longitudinal Dynamic Stability Characteristics

Flight Condition	Omega sp (rad/sec)	Zeta sp	T. Const.1	T. Const.2
1	.12	-.02	-2.42	.81
2	.05	.16	-.36	.14
3	.067	.42	-.40	.15
4	.14	.28	-.413	.172
5	.104	.208	-.588	.289
6	.23	.976	-7.33	.594
7	2.16	.345	-39.63	27.08
8	2.16	.345	- .602	.384

Table 9.12 - Longitudinal Dynamic Stability Requirements

		S.P. Freq. (rad/sec)		Damping Ratio	
Flight Condition	Phase Type	Min	Max	Min	Max
2	Ground Attack	4.3	16	.35	1.30
4	Combat	3.8	14	.35	1.30
7	Cruise	1.8	12	.30	2.00

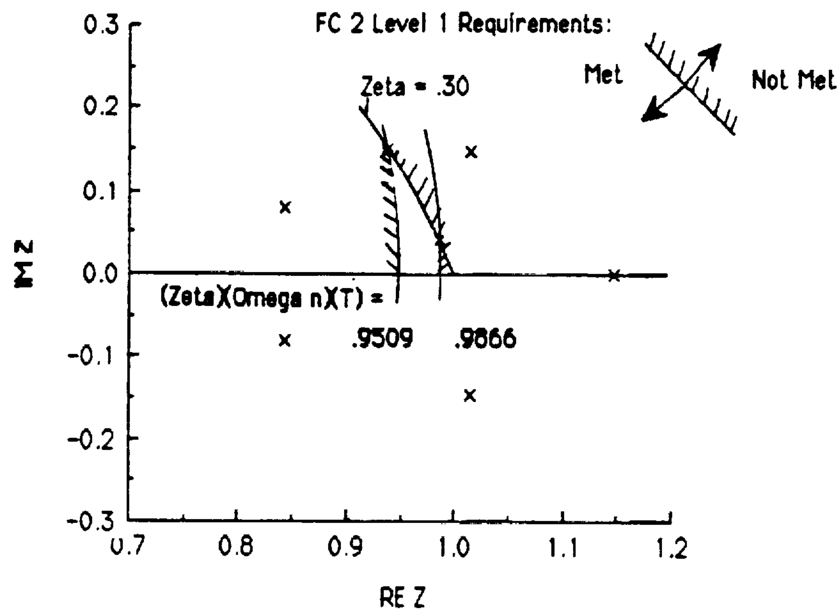

Figure 9.18 - Longitudinal Z-Plane Open Loop Root Locus, Flight Condition 2

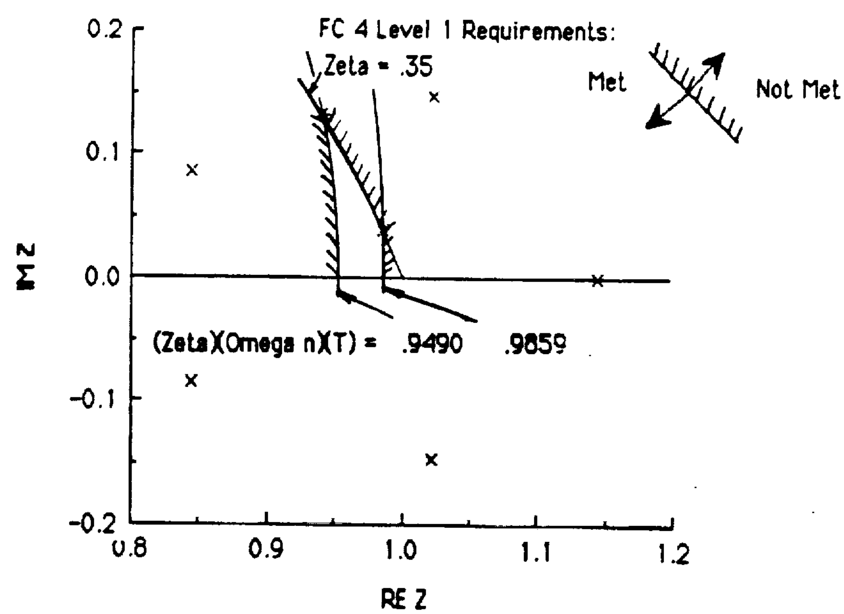

Figure 9.19 - Longitudinal Z-Plane Open Loop Root Locus, Flight Condition 4

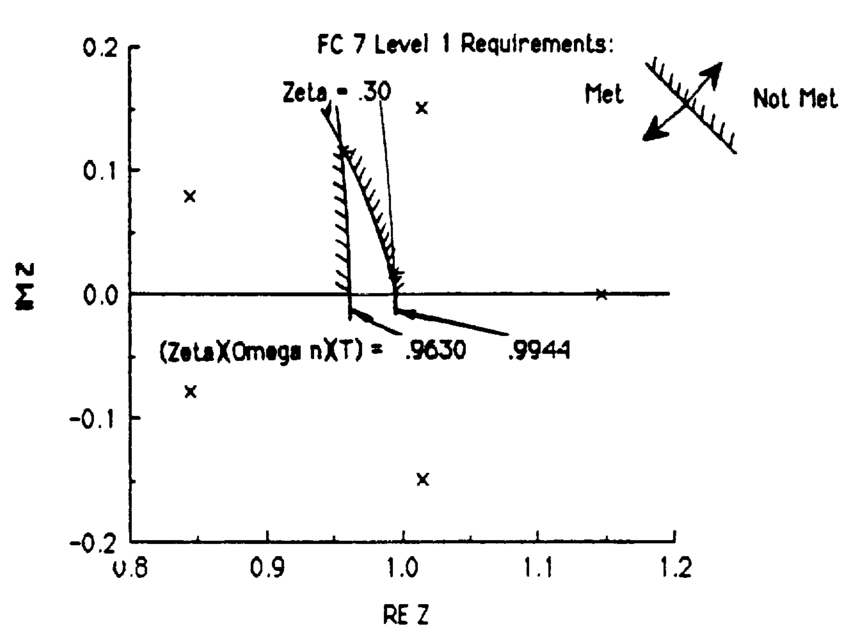

Figure 9.20 - Longitudinal Z-Plane Open Loop Root Locus, Flight Condition 7

Table 9.13 - Longitudinal Compensation Equation Summary

Flight Condition	Compensator Numerator
2	$(z - 1.1467)(z^2 - 2.0296z + 1.05169)(z^2 - 1.6868z + .717660)$
4	$(z - 1.1431)(z^2 - 2.0424z + 1.06452)(z^2 - 1.6868z + .718413)$
7	$(z - 1.1467)(z^2 - 2.0296z + 1.05169)(z^2 - 1.6868z + .717660)$

Flight Condition	Compensator Denominator
2	$(z + .24)(z + .5)(z + .7)(z^2 - 1.97131z + .973316)$
4	$(z + .24)(z + .5)(z + .7)(z^2 - 1.97038z + .972003)$
7	$(z + .24)(z + .5)(z + .7)(z^2 - 1.98839z + .988742)$

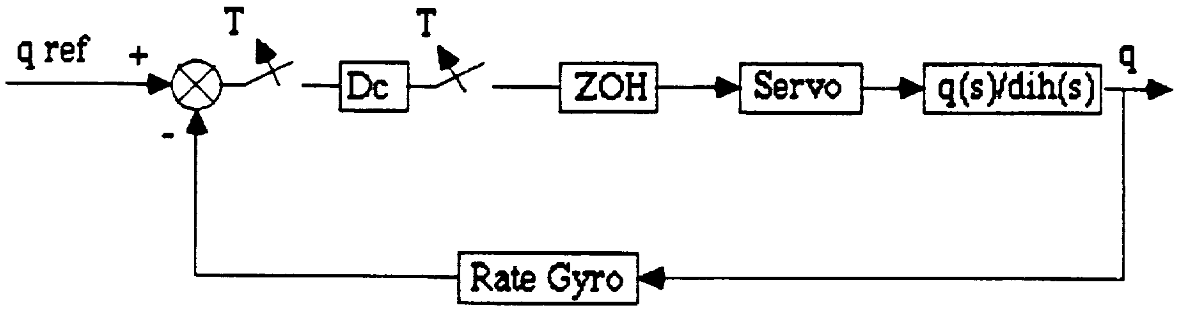

Figure 9.21 - Pitch SAS Block Diagram

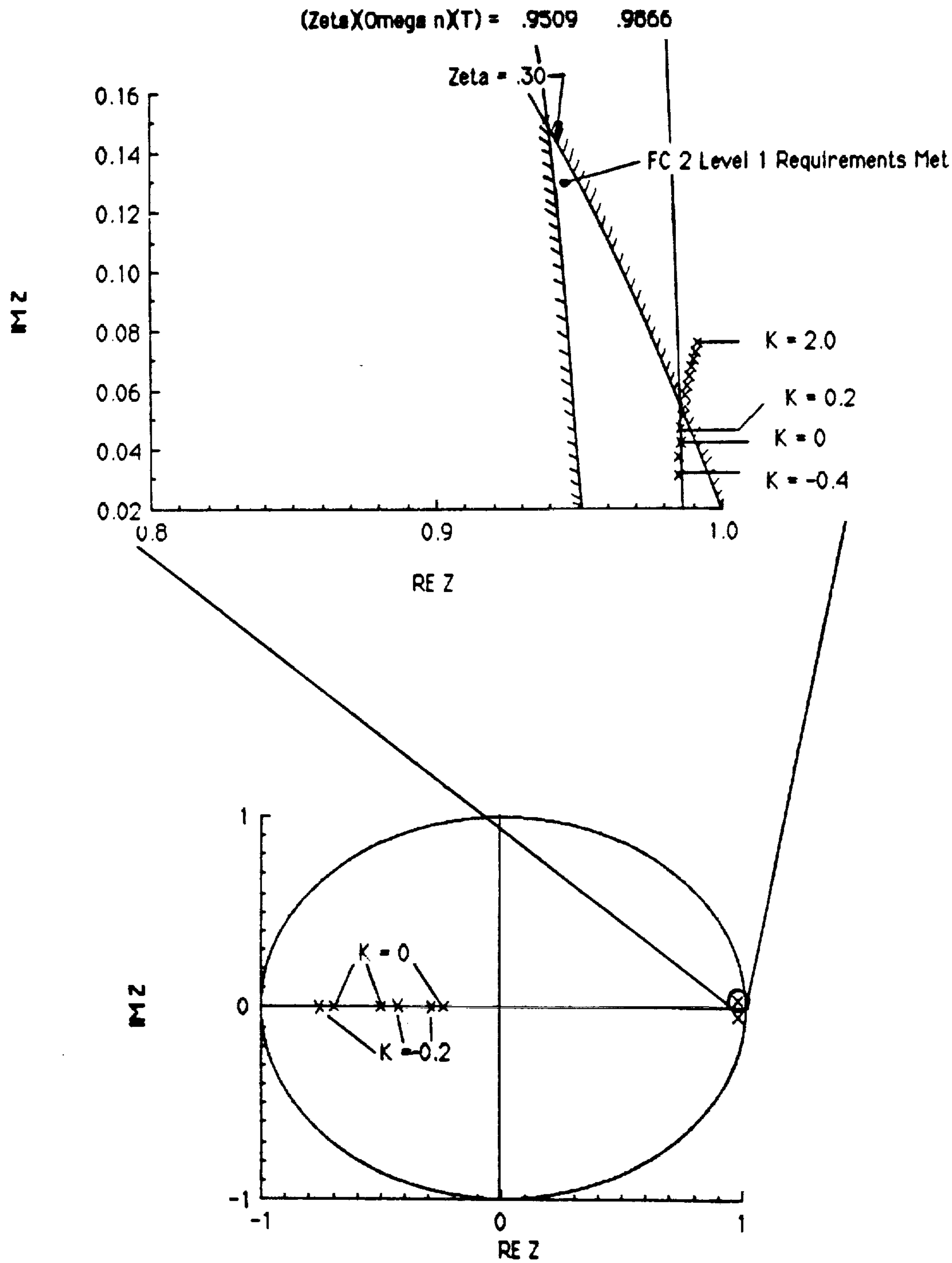

Figure 9.22 - Longitudinal Z-Plane Closed Loop Root Locus, Flight Condition 2

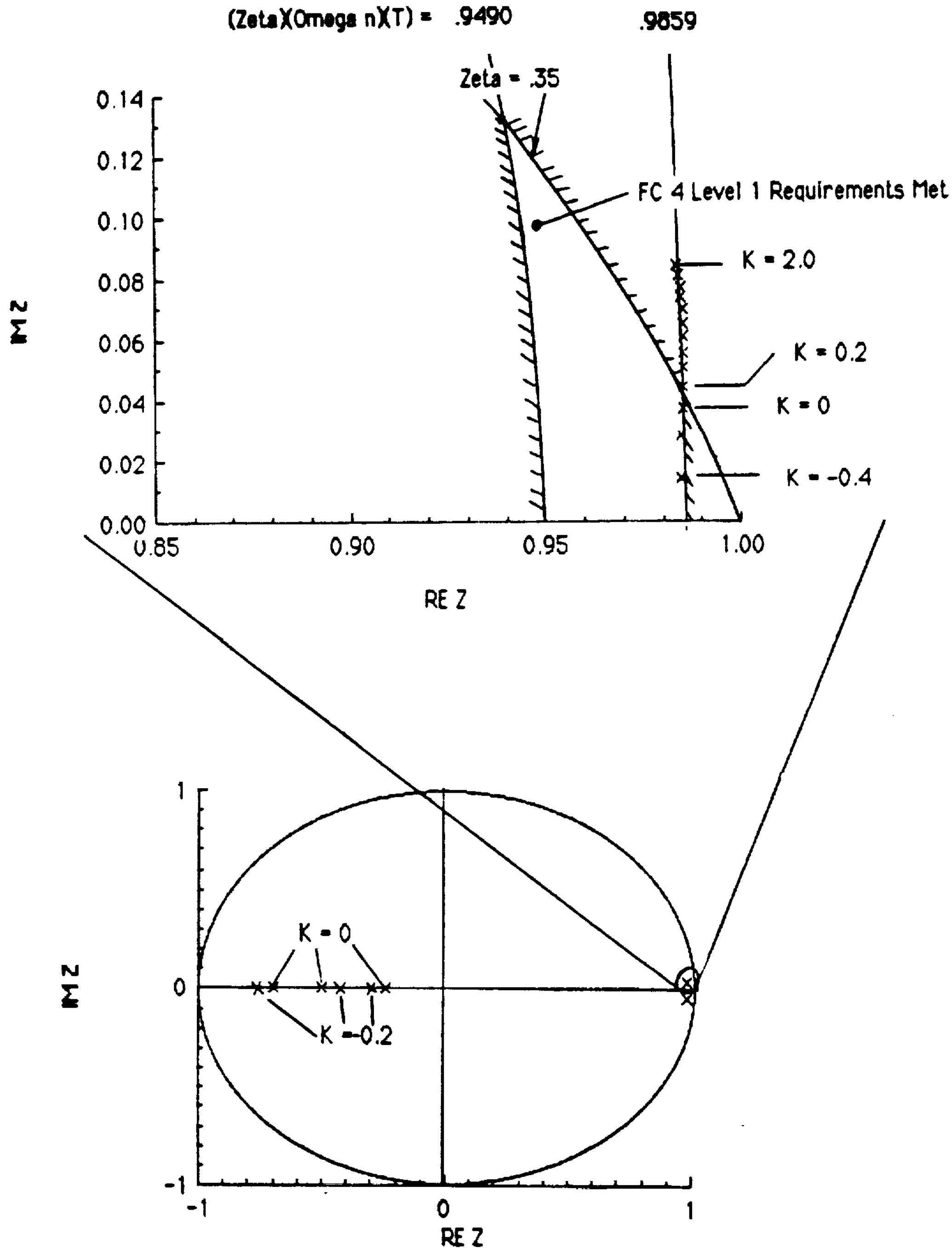

Figure 9.23 - Longitudinal Z-Plane Closed Loop Root Locus, Flight Condition 4

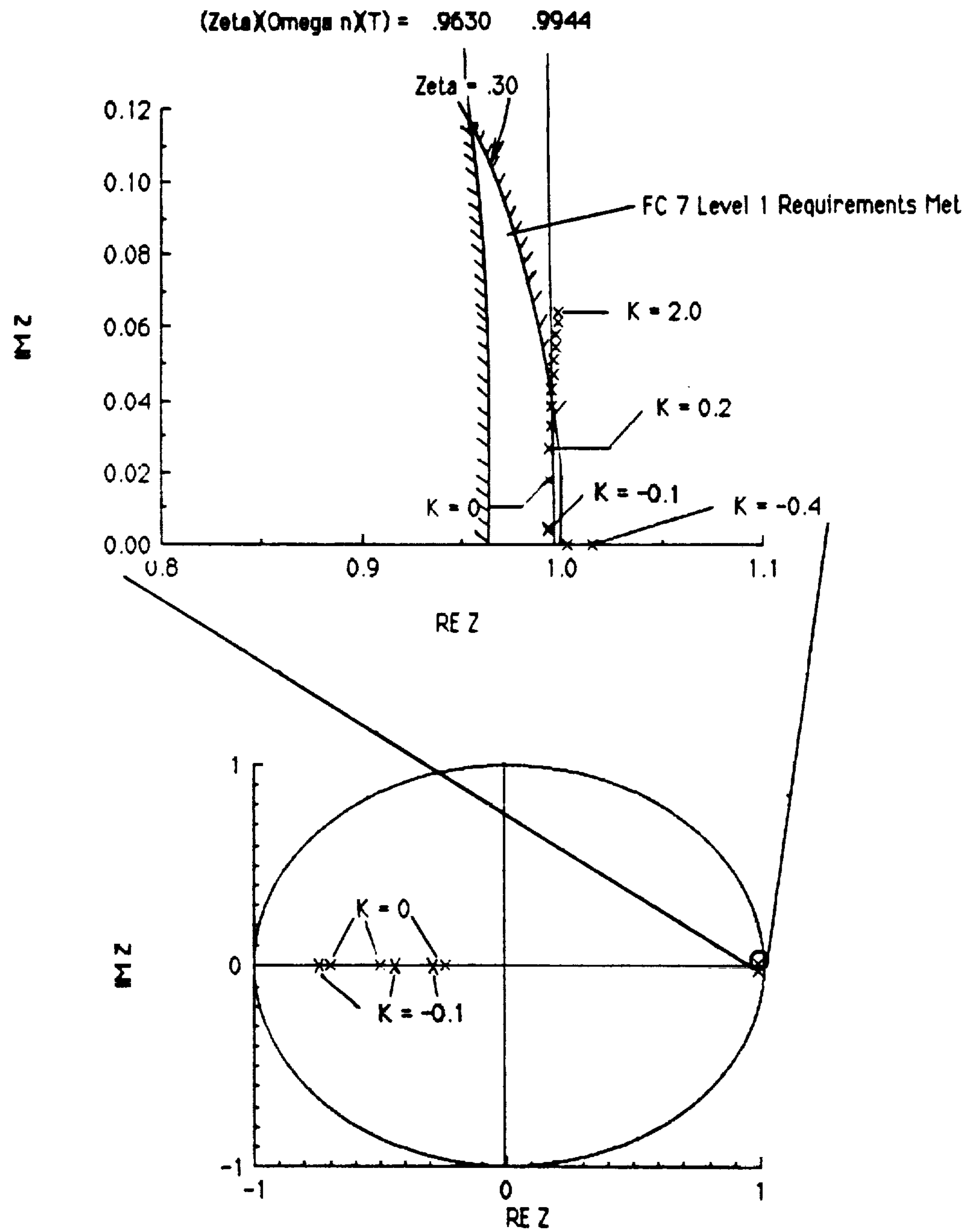

Figure 9.24 - Longitudinal Z-Plane Closed Loop Root Locus, Flight Condition 7

The augmented values of short period frequency and damping are shown in Table 9.14. The values of gain were selected to meet both the handling qualities requirements and the inertia coupling requirements. Details on inertia coupling appear in Section 9.5.

Table 9.14 - Augmented Longitudinal Stability Characteristics

Flight Condition	Gain	Omega sp (rad/sec)	Zeta sp
2	-0.2	4.35	.38
4	-0.2	3.96	.37
7	-0.1	1.88	.3

9.4.2 Lateral

Lateral stability in a fighter is very important. If sufficient roll time-constants can not be met, then a roll damping stability augmentation system (SAS) is necessary. The MIL-F-8785-C requirements, as in Reference 9.7, were examined to determine what the roll and spiral time constants needed to be. For flight conditions 2 and 4 the Monarch is in flight phase category A, terrain following (TF) and air-to-air combat (CO), respectively. In flight condition 7 the Monarch is in flight phase category B, cruise (CR). The Monarch is considered a Class IV aircraft due to its high maneuverability. According to these flight phase categories for a Class IV aircraft MIL-F-8785-C dictates the requirements of Table 9.15.

Table 9.15 - Lateral Dynamic Stability Requirements

FC	Max Roll Time Constant	Min Time to Double Amplitude
2	1.0 second	12 seconds
4	1.0 second	12 seconds
7	1.0 second	12 seconds

The basic roll damping SAS block diagram is shown in Figure 9.25. The bank angle to aileron transfer function was determined using the matrix method of Reference 9.9, using the stability and control derivatives of Section 9.3. The open loop transfer function was determined using the Laplace variable s, and then a total pulsed transfer function was determined in the z domain. PC-Matlab was used to perform the z transform. The z plane root locus was used to find the root locations, and these in turn were used to determine where the spiral and roll roots needed to be. The z plane root locus was then looked at to see whether using a different gain would make a difference. If gain could not solve the problem then a compensator had to be implemented to move the roots to the desired locations.

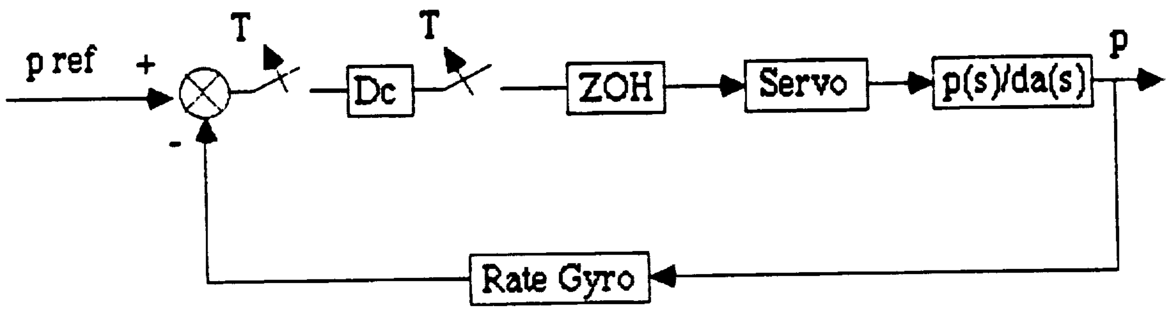

Figure 9.25 Roll Damping SAS Block Diagram

For flight condition 2 it was found that by using a gain of 0.25 that both the roll and spiral time constant requirements for Level 1 could be met. Flight condition 4 inherently met Level 1 handling qualities for the lateral modes, so no stability augmentation was necessary.

For flight condition 7 the uncompensated system was seen to have too small a value of T_2_s. This meant that the amplitude of the bank angle was being doubled too quickly for Level 1 handling qualities. From the MIL-F-8785-C requirements it was known that the spiral time constant root had to be increased without making the roll time constant greater than 1.4 seconds. The z plane root locus was examined to determine where this was possible. A compensator was then designed to make the sytsem meet the Level 1 requirements. Detail design of the compensator can be seen in Appendix 3. The discrete transfer function of the compensator which is to be implemented in the digital flight control computer is:

$$Dc(z) = \frac{(z - 1.0009) * (z - .9050)}{(z - .9903) * (z + .10)}$$

The uncompensated z plane root locus for flight condition 7 is shown in Figure 9.26. The compensated z plane root locus is shown in Figure 9.27. A design gain of 0.3 was found to give sufficient roll and spiral time constants for Level 1 handling qualities. Detailed development of the roll damping SAS can be found in Appendix 3. The gains that are necessary for lateral dynamic stability are summarized in Table 9.16.

Table 9.16 - Lateral Control Gains Necessary for Level 1 Handling Qualities

FC	Gain	T 2s	T R
2	0.25	23.1 sec	.1327 sec
4	0.0	22.1 sec	.4888 sec
7	0.21	28.85 sec	.0778 sec

9.4.3 Directional

The directional stability of the Monarch will be enhanced with the use of a digital yaw Stability Augmentation System (SAS). The yaw SAS will, when required, improve the dutch roll characteristics of the airplane. Figure 9.28 illustrates the block diagram of the digital yaw SAS system. The unaugmented dutch roll characteristics and the corresponding handling level are listed in Table 9.10. The handling level requirements are based on MIL-F-8785C specifications and can be found in Reference 9.7. As stated, only flight conditions 2, 4 and 7 are investigated.

For the digital controller, a sampling rate of 100 cycles per second was selected as suggested from Reference 9.8. For flight condition 2 the unaugmented dutch roll discrete root locus is shown in Figure 9.29. The lines of constant danping and constant (σT) for Level 1 requirements are shown. The figure shows that dutch roll Level 1 handling qualities can not be met for any value of gain.

Detailed development of the compensator for flight condition 2 can be found in Appendix 3. The following implementation equation was developed to achieve Level 1 qualities for flight condition 2:

$$D_c(z) = \frac{z^2 - 1.9978z + .9978}{z^2 - 1.9766z + .9773}$$

The augmented dutch roll discrete root locus is illustrated in Figure 9.30. For gain ranges of 0 to -1.5, Level 1 handling qualities are achieved. A gain of - 0.1 is selected to give a dutch roll damping ratio of .60 and a frequency 2.25 rad/sec. These values were selected to help achieve favorable inertia coupling characteristics as discussed in Section 9.6.

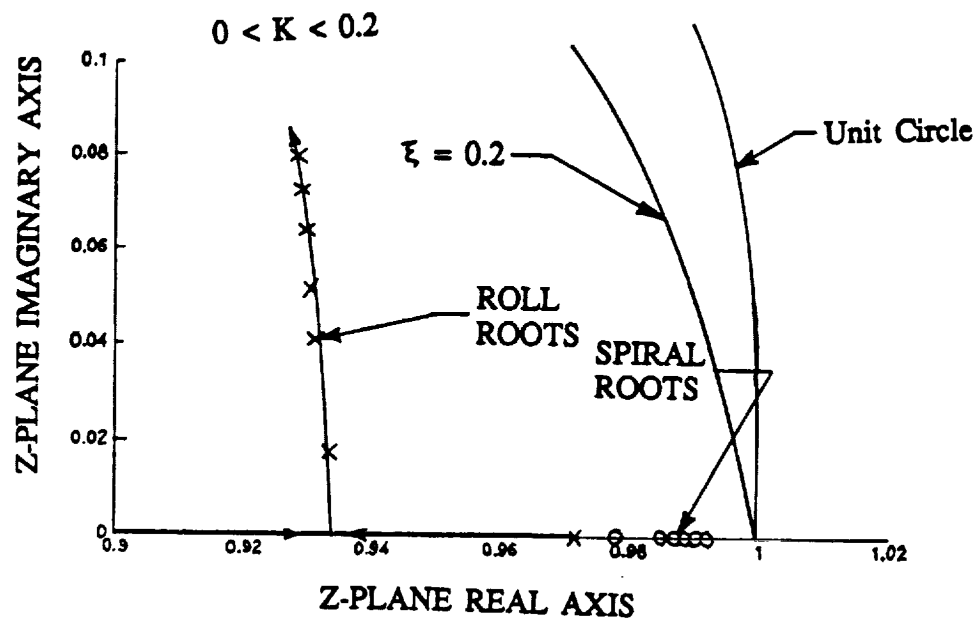

Figure 9.26 Uncompensated Roll Z-Plane Root Locus for Flight Condition 7.

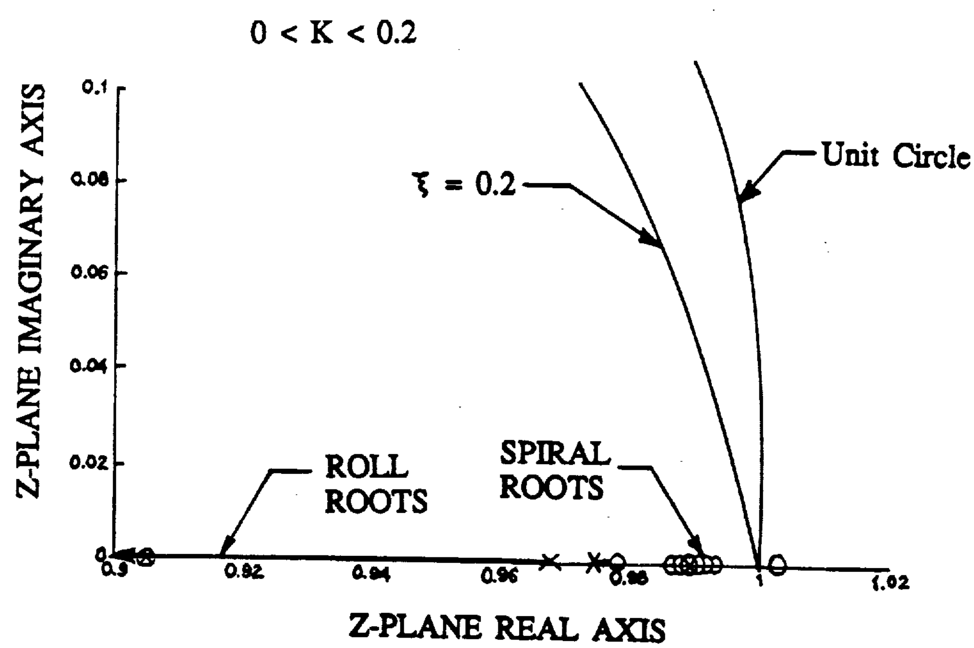

Figure 9.27 Compensated Roll Z-Plane Root Locus for Flight Condition 7

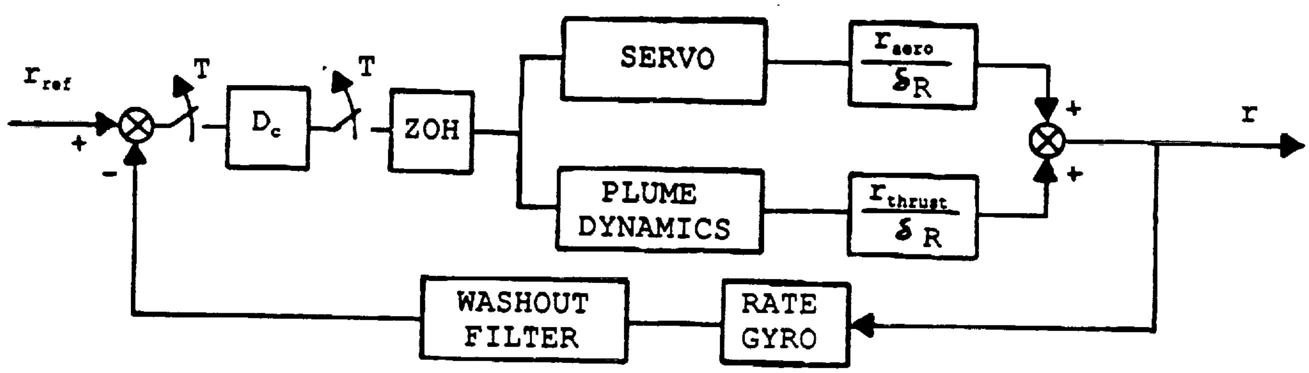

Figure 9.28 Yaw SAS Block Diagram

Table 9.17 Dutch Roll Characteristics and Corresponding Handling Levels

Flight Condition	Zeta D	Omega D	Zeta*Omega	Handling Level
1	.085	1.5922	.135	2
2	.183	.5665	.104	2
3	.1895	2.6564	.503	2
4	.3769	4.1945	1.580	1
5	.6543	6.1797	4.040	1
6	.1223	3.1533	.386	1
7	Two Real Roots outside the unit circle in the Z-Plane - Unstable			
8	.1299	1.7693	.230	1

As indicated in Table 9.17, flight condition 4 dutch roll characteristics meet level 1 handling requirements. Therefore no augmentation is required.

The unaugmented discrete dutch roll root locus for flight condition 7 is illustrated in Figure 9.31. For increasing negative gain, it is seen that the roots meet on the real axis and split to form the oscillatory dutch roll pair. Level 1 handling qualities are still not obtained, as illustrated by the contant damping and (σT) lines.

Appendix 3 documents the full development of the discrete compensator for this flight condition. The following implementation equation was developed to achieve Level 1 handling qualities for flight condition 7:

$$D_c(z) = \frac{z^2 - 2.0088z + 1.0088}{z^2 - 1.9968z + .99707}$$

The augmented discrete dutch roll root locus for flight condition 7 is illlustrated in Figure 9.32. For gains ranging from 0 to 2 it is seen that the dutch roll roll does not move much. A gain of 2 is selected to give a dutch roll daping ratio of .10 and a frequency of 1.65 rad/sec. These values were selected to help achieve favorable inertia coupling characteristics as discussed in Section 9.6.

As can be seen from the development of the discrete compensators for the yaw SAS, a different compensator is required for the two flight conditions investigated. Therefore compensator, as well as gain scheduling will be required. This is possible when using a digital computer to implement the discrete compensator in the flight control system.

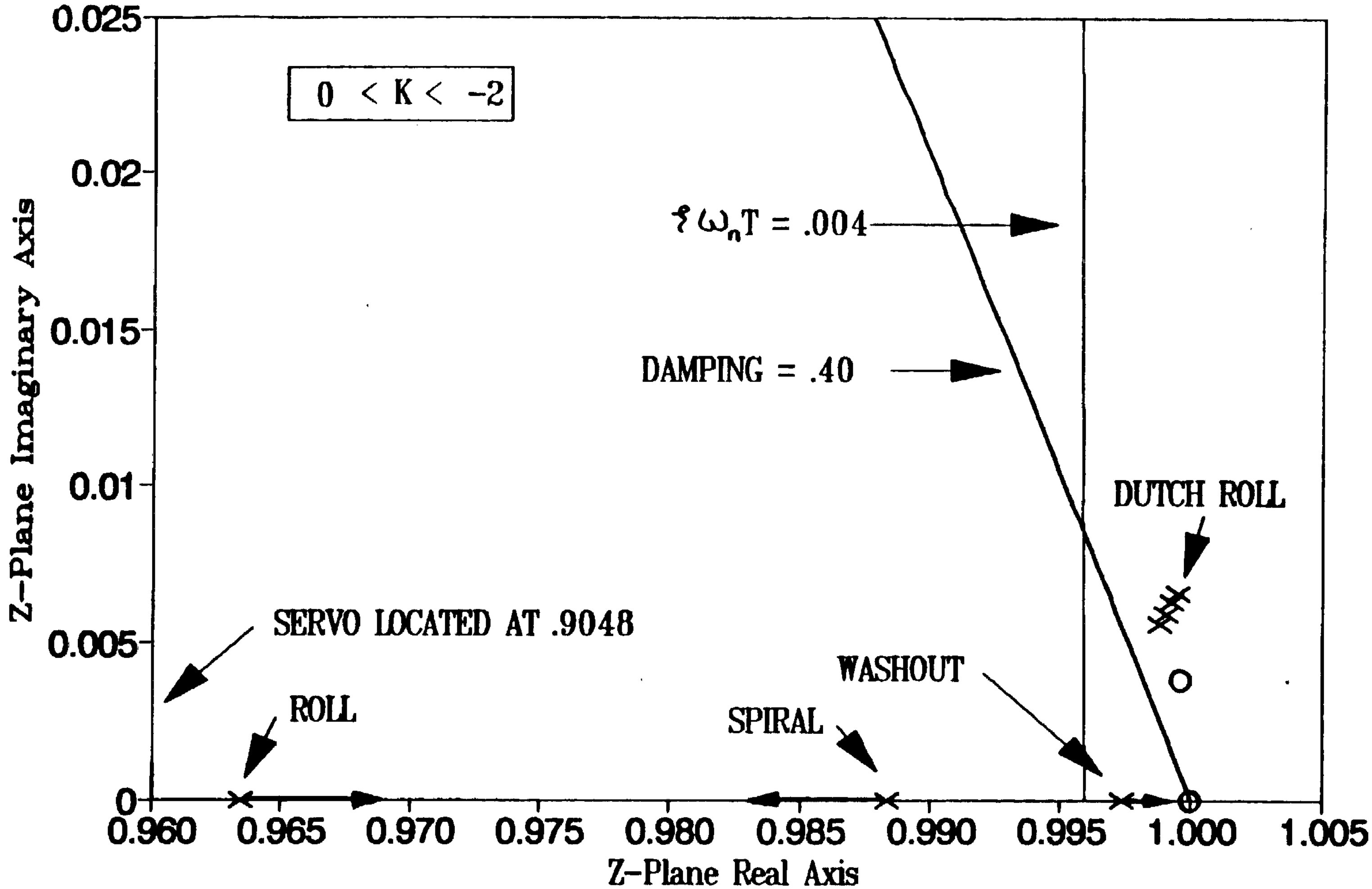

Figure 9.29 Unaugmented Discrete Dutch Roll Root Locus for Flight Condition 2

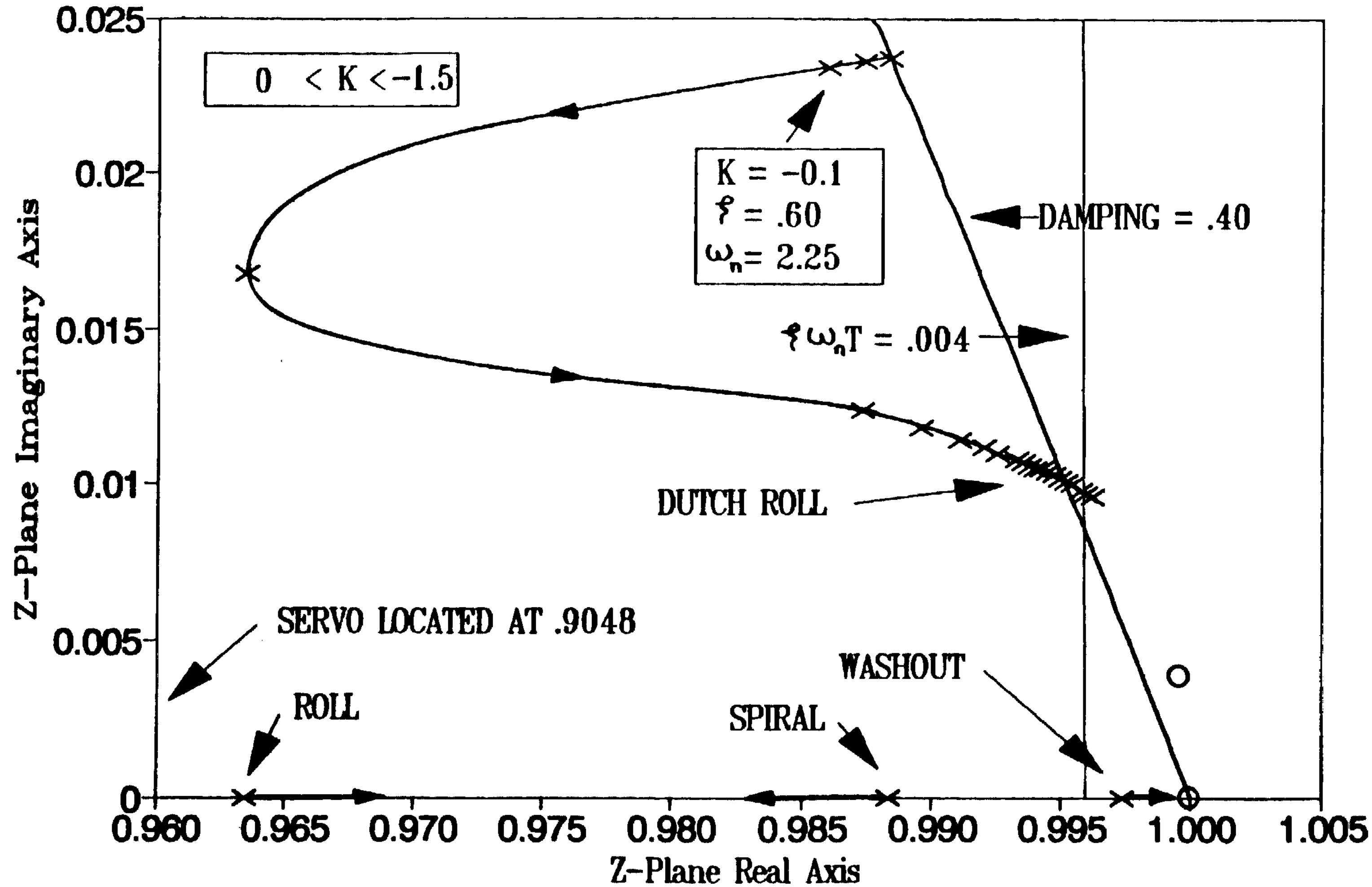

Figure 9.30 Augmented Discrete Dutch Roll Root Locus for Flight Condition 2

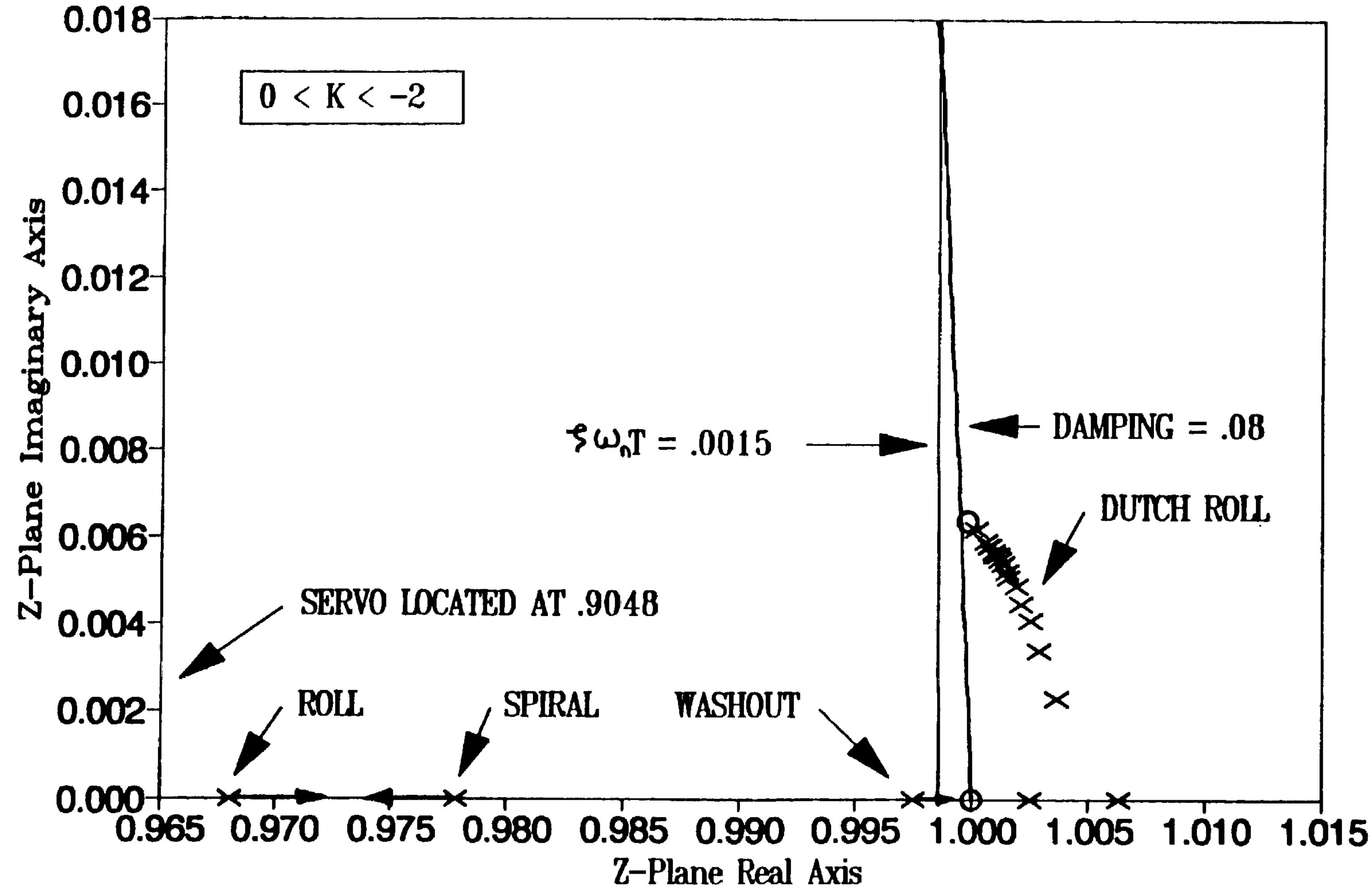

Figure 9.31 Unaugmented Discrete Dutch Roll Root Locus for Flight Condition 7

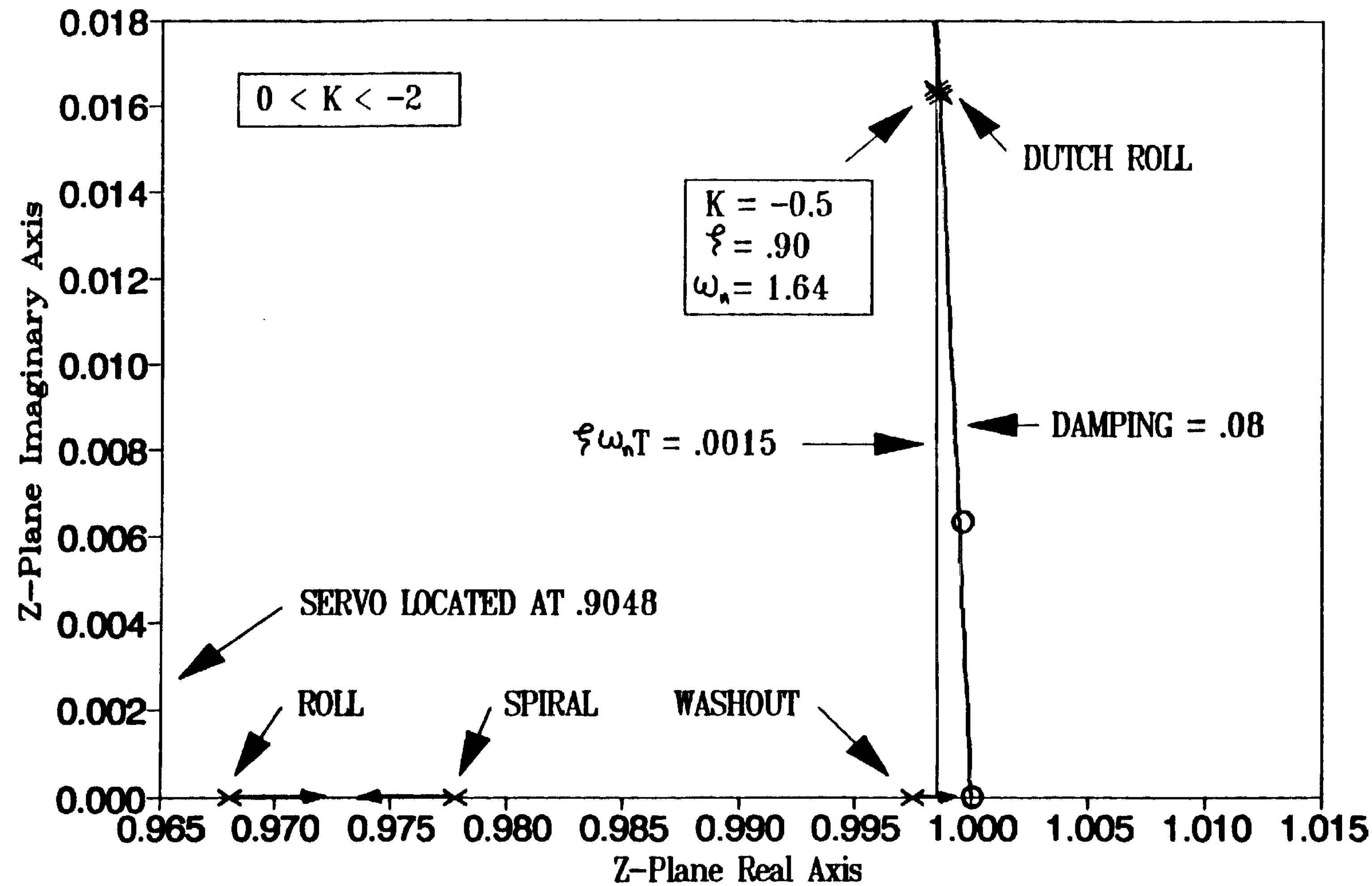

Figure 9.32 Augmented Discrete Dutch Roll Root Locus for Flight Condition 7

9.5 ROLL PERFORMANCE

Roll performance is vital to the success of a fighter. A pilot needs to be able to roll his aircraft rapidly to avoid enemy fire and to point at his enemies so as to lock on ordnances. The maximum roll rate isusually around 150-180 degrees per second, depending on the role of the fighter, according to Reference 9.10. If this roll rate is too high there is the possibility of inertia coupling, and this would keep the plane from being able to roll consecutive loops.

The procedure of Reference 9.7 was used to determine the roll control power derivative due to lateral cockpit control, C_l_del_cpt. This analysis assumes that the Monarch only uses its ailerons for roll control. The maximum deflection of the ailerons is 25 degrees.

The Level 1 requirements according to MIL-F-8785-C are shown in Table 9.18.

Table 9.18 - Maximum Roll Rate Requirements

FC 2: must go through 90 degrees of bank in 1.3 seconds
FC 4: for 360 deg rolls: 90 deg in 1 sec, 180 deg in 1.6 sec, 360 deg in 2.8 sec
normal flight phase: 90 deg in 1.1 sec, 180 deg in 2.2 sec
FC 7: 50 degrees of bank in 1.1 seconds

The equations of Reference 9.7 were solved to find the maximum roll rates of the Monarch. For the known parameters of each flight condition, the time was put into the equations to see the level of bank angle response that resulted. The bank angles that the Monarch could go through are shown in Table 9.19, along with the roll time constants.

Table 9.19 - Bank Angle Response and Roll Time Constants for the Monarch

FC	M	q bar	I xx s	time, t	phi(t)	T r
2	.65	1067	15370	1 sec	18.7 deg	.244
4	.90	678	10631	1.0 sec	282 deg	.257
				1.6 sec	2789 deg	.257
				2.8 sec	296213 deg	.257
7	1.60	1127	10152	1.1 sec	31.7 deg	.264

The roll performance of the Monarch does not meet Level 1 in all flight phases. The effect of using the stabilators for roll control for meeting the Level 1 should be investigated. This was not done because it was assumed that the ailerons could provide adequate roll control power. Using the stabilators for roll control would have to be looked at with respect to inertia coupling and roll damping SAS.

9.6 INERTIA COUPLING

An additional constraint on the selection of gains for the Monarch flight control system was the susceptibility of the aircraft to inertia (roll) coupling. Because of the high roll rates and rapid maneuvers that fighters must execute, these aircraft are vulnerable to excursions in pitch and yaw while performing combat rolls. Thus, the selection of short period and dutch roll frequencies and dampings were coordinated for the three flight conditions to avoid this problem.

The method used for the inertia coupling analysis comes from Reference 9.11. Plots indicating the vulnerability of the unaugmented aircraft to inertia coupling appear in Figures 9.33 - 9.35. Calculation of these data appear in Appendix 3. The width of the "throat" between the two hyperbolic boundaries on these plots varies with the product of short period and dutch roll damping. The slope of the line which starts at the origin and passes between the boundaries is the ratio of the dutch roll frequency to short period frequency. An inertia coupling incident occurs if this line intersects one of the hyperbolic boundaries. The roll rate at which this departure occurs can be calculated from the frequency to roll rate values on the axes and the corresponding dutch roll or short period frequency. As shown in the calculations in Appendix 3, the Monarch suffers inertia coupling in Flight Condition 2 at roll rates below 28 degrees per second. In flight condition 4, the aircraft departs at roll rates below 160 degrees per second. As shown in Section 4.5, the roll rate capabilities of the unaugmented aircraft place the Monarch in the unstable region of the inertia coupling plots.

Figures 9.36-9.38 show the inertia coupling diagrams for the Monarch after implementation of the compensators described in Section 9.4. As seen from the plots, the frequency and damping ratios selected for the Monarch do not produce any instances of inertia coupling. This was made possible by keeping the ratio of short period frequency to dutch roll frequency as close to one as was feasible, pending the restrictions of the handling qualities requirements. The minimum required dutch roll frequency was much less than the minimum required short period frequency for the flight conditions analyzed for the aircraft. As this produced inertia coupling problems, the short period frequency was held at its minimum allowable value and the dutch roll frequency was increased until the ratio of the frequencies moved the line shown in the plots out of the unstable region. The compensators chosen for the digital flight control system used these frequencies as design points.

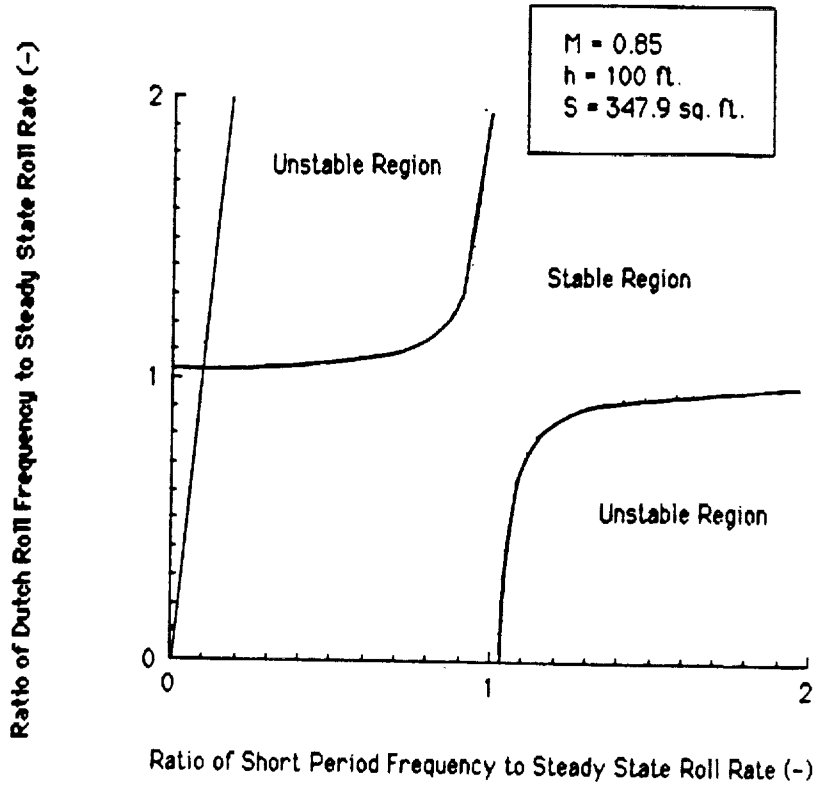

Figure 9.33 - Inertia Coupling Diagram for Flight Condition 2, Open Loop

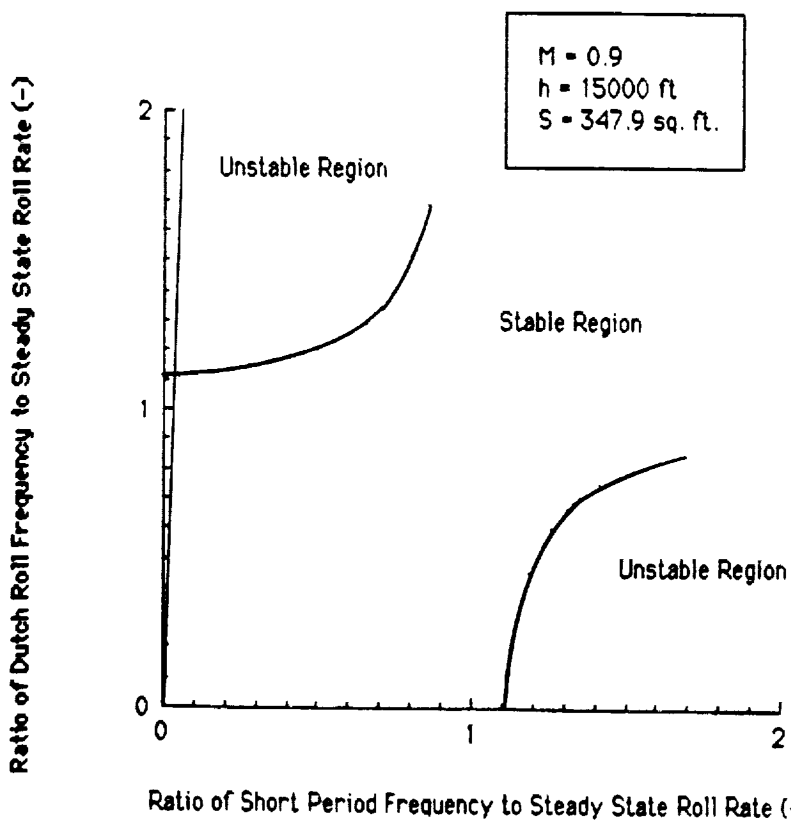

Figure 9.34 - Inertia Coupling Diagram for Flight Condition 4, Open Loop

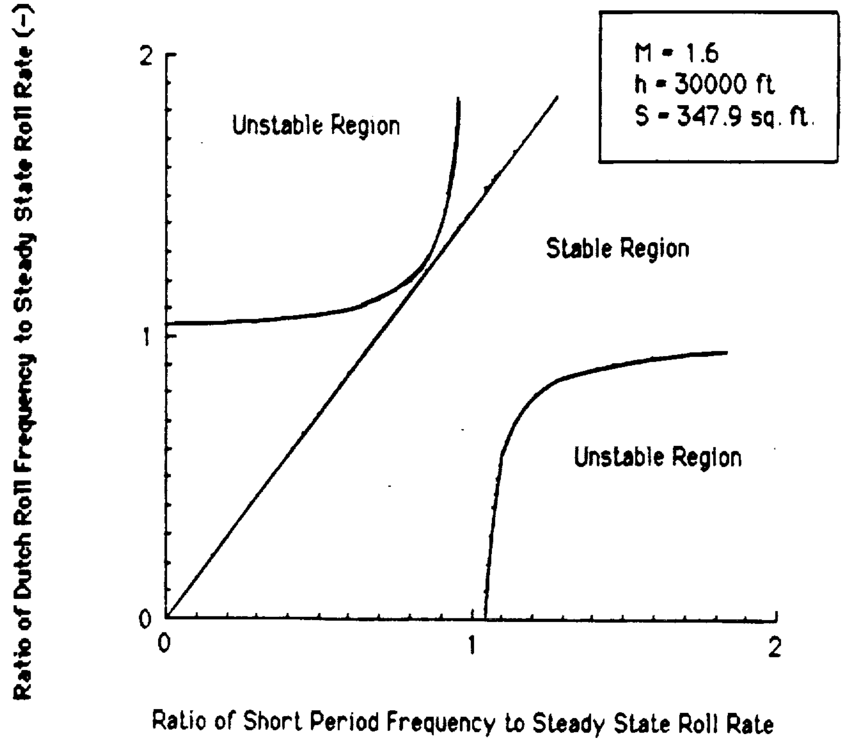

Figure 9.35 - Inertia Coupling Diagram for Flight Condition 7, Open Loop

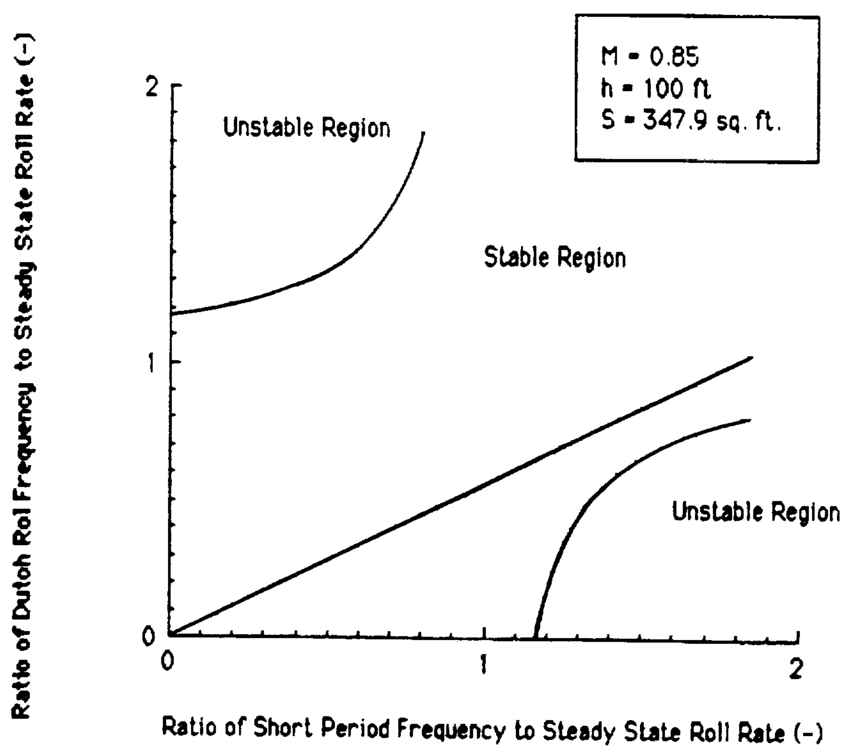

Figure 9.36 - Inertia Coupling Boundaries for Flight Condition 2

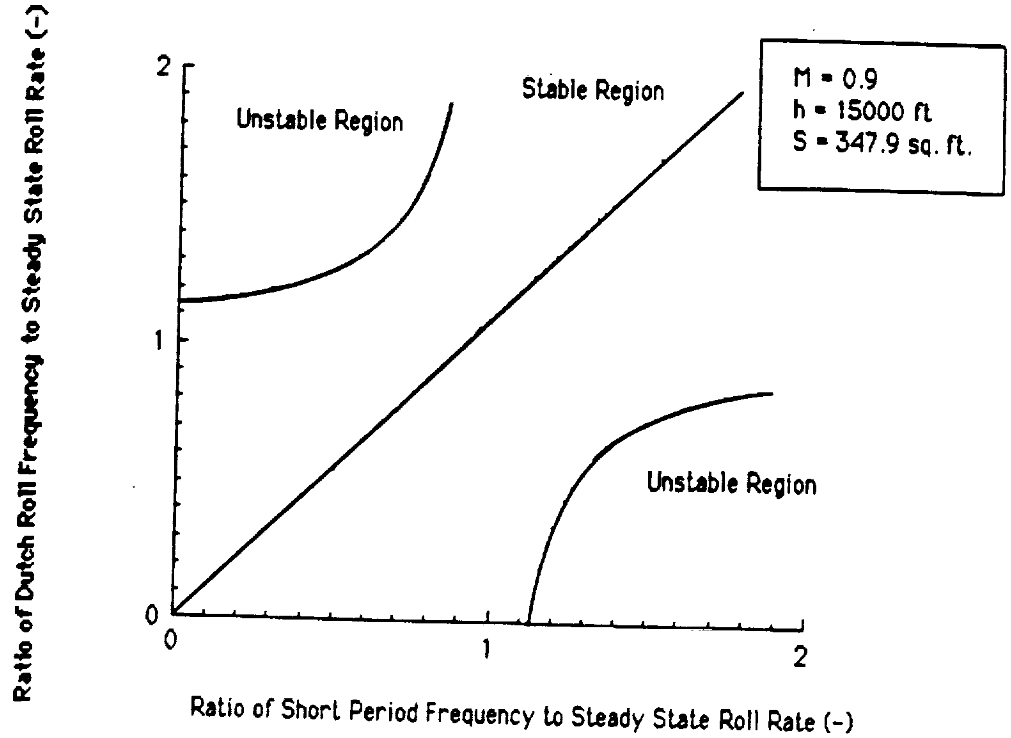

Figure 9.37 - Inertia Coupling Boundaries for Flight Condition 4

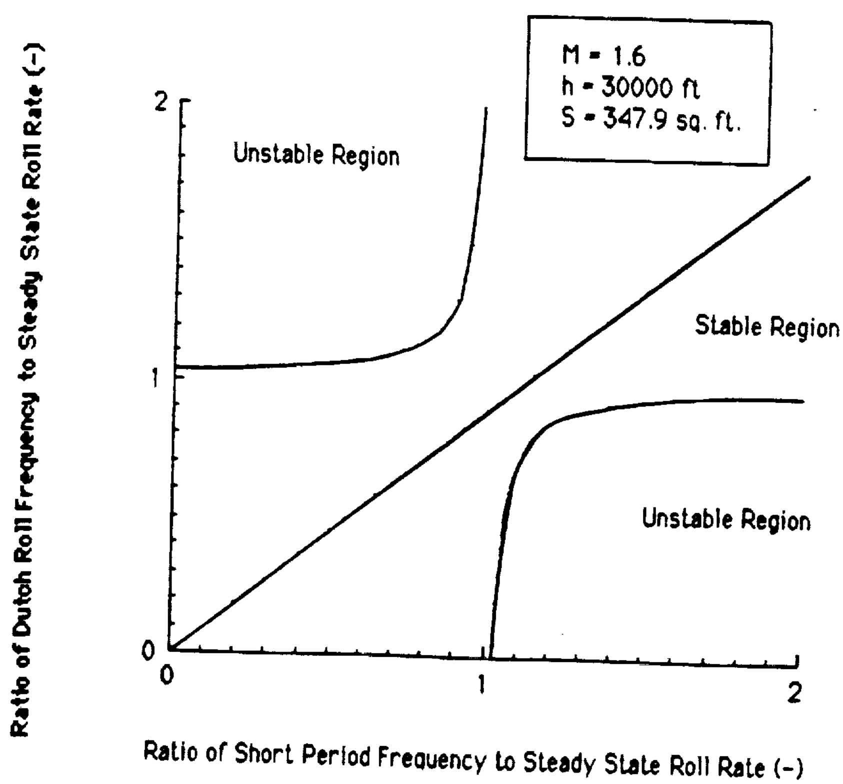

Figure 9.38 - Inertia Coupling Boundaries for Flight Condition 7

9.7 SPIN DEPARTURE

One measure of aircraft spin departure at stall angle of attack is the value of Cn_beta dynamic. Using the methods of Reference 9.11, the aircraft has adequate resistance to departure when the sign of Cn_beta dynamic is positive. Calculations of this parameter appear in Appendix 3. The results are summarized in Table 9.20. These data indicate that the Monarch does not enter a divergence at the stall angle of attack. This analysis gives no indication of the spin tendencies of the aircraft in the post-stall flight regime.

Table 9.20 - Results of Spin Departure Analysis

Flight Condition	h, ft.	M	Cn_beta dynamic
1	0	.20	.4532
2	100	.85	.0955
3	10000	.90	.2099
4	15000	.90	.3789
5	30000	.90	.4770
6	30000	1.20	.3285
7	30000	1.60	.0908
8	40000	.80	.2291

According to Reference 9.11, a means of aiding aircraft spin recovery is to locate the most aft center of gravity of the aircraft ahead of the centroid of the total aircraft planform. This guarantees a form of stability margin at an aircraft angle of attack of 90 degrees. As shown in Figure 9.39, the most aft center of gravity is ahead of the planform centroid. This margin of stability for the aircraft is 8.12 inches, or 5.64% of the mean geometric chord. Therefore, these analyses indicate that the Monarch does not have any inherent spin tendencies.

9.8 LOW LEVEL RIDE QUALITIES

Attack mission require fighter aircraft to fly low level, high speed profiles to the target. The aircraft and the pilot must be capable of accurately delivering ordnance in this flight regime. An assessment of the low level ride qualities of the Monarch was completed to determine if the aircraft required a ride quality augmentation system.

The method for this analysis comes from Reference 9.7. A "root mean squared g-level" per foot per second gust level ($\bar{A}$) was calculated for the aircraft in Appendix 3. While this analysis is usually done only for low level, high-speed flight, the values of A

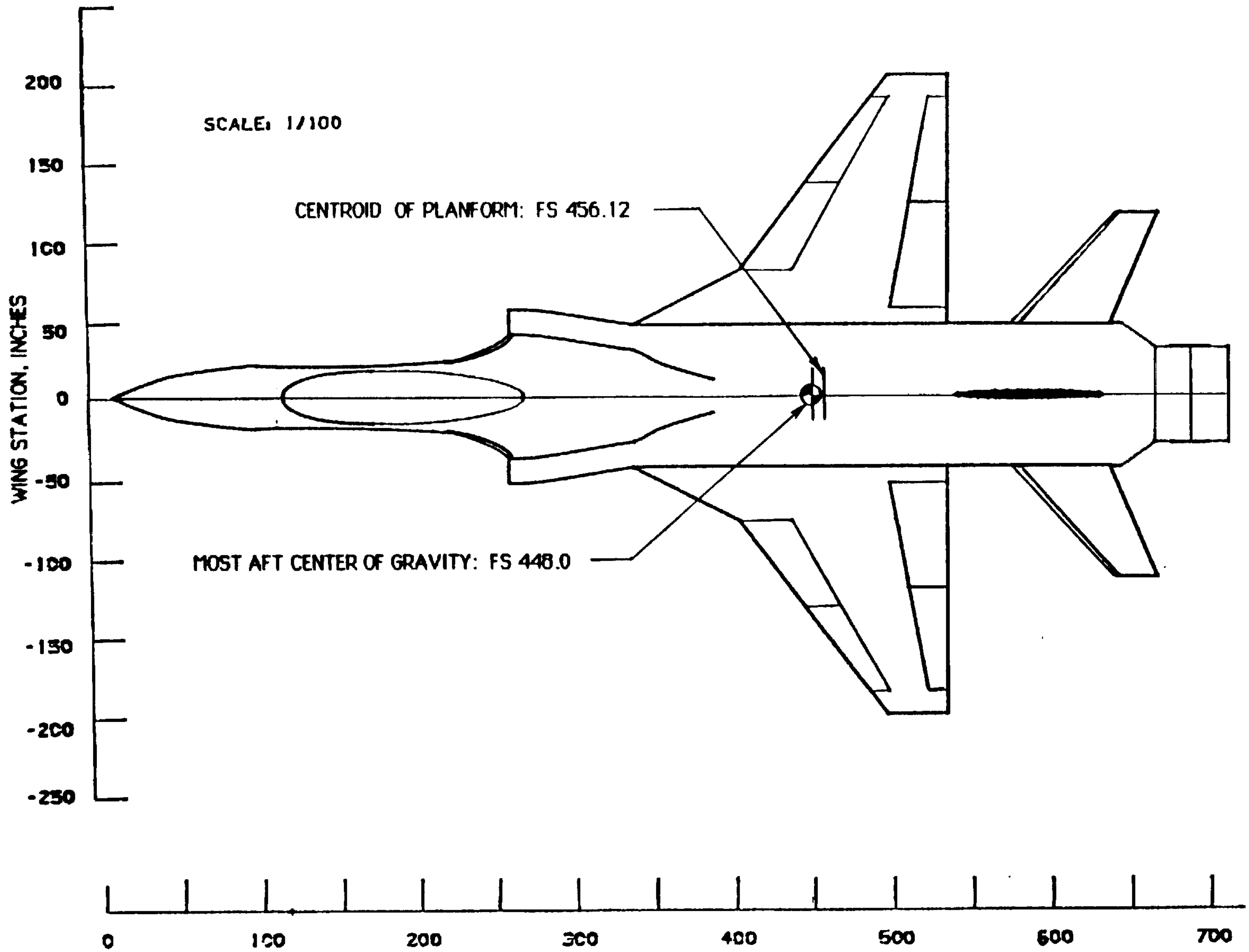

Figure 9.39 - Static Stability at Most Aft C. G., 90 Degrees Angle of Attack

appear in Table 9.21 for the eight flight conditions selected for the stability derivative calculations.

Table 9.21 - Monarch Ride Qualities

Flight Condition	h, ft.	M	Weight, lbs.	$\bar{A}$, g's/fps
1	0	0.2	30744	.00497
2	100	0.85	28395	.027774
3	10000	0.90	23980	.02742
4	15000	0.90	23980	.02385
5	30000	0.90	28223	.01546
6	30000	1.20	24534	.01818
7	30000	1.60	27239	.01465
8	40000	0.80	29917	.009324

Values of $\bar{A}$ greater than 0.005 generally indicate deficiencies in ride qualities. This analysis indicates that the aircraft may need a ride quality augmentation system throughout most of its flight envelope. The extensive augmentation of the standard flight control system may provide some improvement in the ride qualities of the aircraft, although this would have to be verified in flight test.

9.9 VERTICAL TAIL/RUDDER REMOVAL STUDY

A study was done on the Monarch to replace or reduce the vertical tail of the airplane with a thrust vectoring control that would provide partial or full directional stability. Eliminating the vertical tail would reduce the radar signature of the aircraft, reduce or eliminate interference drag between control surface at the aft end of the aircraft, and may reduce the complexity of some of the flight control system.

The preliminary design of the Monarch used a vertical tail with a 63 sq. ft. area. Using the lateral gust handling qualities requirements of MIL-F-8785C, which allow for a 30 knot gust, equivalent sideslip angles were produced for the eight flight conditions developed for the stability and control analysis. A proposed thrust vectoring location (FS 625) was used to determine:

* the sideforce that would be needed to control an aircraft without a vertical tail,

* the sideforce that would be needed to control an aircraft with the original vertical tail but without a rudder.

The calculations for this analysis appear in Appendix 3. The results are summarized in Tables 9.22 and 9.23.

Table 9.22 - Sideforce Required for Adequate Static Directional Stability

Flight Condition	Sideslip, rad	Sideforce at FS 625, lbs
1	0.2233	2583
2	0.0534	13077
3	0.0522	9712
4	0.0532	8038
5	0.0566	4231
6	0.0424	5926
7	0.0318	7939
8	0.0654	2339

Table 9.23 - Sideforces Needed to Replace the Rudder

Flight Cond.	Sideslip, rad	Rudder Defl., deg	Sideforce at FS 625, lb
1	0.2233	-0.478	-61.5
2	0.0534	0.309	962.8
3	0.0522	1.49	4318.3
4	0.0532	1.78	4219.5
5	0.0566	2.19	2663.2
6	0.0424	1.83	4232.7
7	0.0318	-0.161	-441.4
8	0.0654	1.47	729.9

In the case of vertical tail removal, high dynamic pressures in flight condition 2 produced unreasonably large sideforces to compensate for the lack of inherent directional stability. The design for the yaw thrust vanes for the Monarch, as shown in Chapter 6, Figure 6.11, did provide adequate directional stability for removal of the rudder. The correlation between sideforce and yaw vane deflection appears in Table 9.24. The physical limit of the yaw vane deflection was 25 degrees. The calculations appear in Appendix 3.

Table 9.24 - Sideforces Produced by Yaw Vane Deflections

Flight Condition	Sideforce at FS 625, lb	Vane Deflection Angle, deg.
1	61.5	1
2	962.8	4
3	4318.3	12
4	4219.5	6
5	2663.2	10
6	4232.7	17
7	441.1	1
8	729.9	9

Additionally, yaw vane deflections allowed for a reduction in the original vertical tail area of the Monarch. The tail area was reduced from 63 square feet to 40 square feet. The calculation for this analysis appear in Appendix 3. This empennage configuration was successfully integrated into the dircetional stability analysis of Section 9.4.

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10. STRUCTURAL LAYOUT, MATERIALS SELECTION, MANUFACTURING PROCESS, ACCESSIBILITY AND MAINTAINABILITY CONSIDERATIONS

The purpose of this chapter is to present the preliminary structural design and materials selection for the Monarch aircraft. The manufacturing process and accessibility and maintenance considerations are also presented here. Structural design is based on the Class I methods presented in Reference 10.8 and a data base of current fighters. Materials selection is based on a desire to create opportunities for weight and life-cycle cost reductions. The manufacturing process is presented with a shop flow diagram and a description of the processes used in forming the aircraft. The accessibility discussion presents the engine removal schemes and other systems maintenance considerations.

10.1. STRUCTURAL ARRANGEMENT

The purpose of this section is to present the preliminary structural arrangement for the Monarch aircraft. The work presented here is used to indicate where primary structural members are located to provide stiffness and component mountings for the Monarch. More advanced structural design requires information on aircraft loads. Loads information was not calculated for the Monarch aircraft. For this reason the structural arrangement of the Monarch is currently based on data for structure of other fighters and assumptions of primary load paths. The design considerations and the comparative data base for each primary structural component are also presented.

10.1.1. Fuselage Structural Arrangement

The driving design considerations for the fuselage structure were to reduce the number of primary frames and achieve synergism wherever possible. Materials used for frames and longerons are aluminum and titanium. The skins use aluminum and composites. The location of secondary frames and longerons was based on the fuselage layouts in Reference 10.12.

The location of major frames for the Monarch is shown in Figures 10.1 and 10.2. These frames are made of titanium. Lesser frames are made of aluminum. These frames are spaced at intervals of 18 inches aft of FS 116.5. Synergism was achieved at the major fuselage frames as shown in Table 10.1. Note the location of jack points on frames FS 364 and FS 552. These points are used during the production stage for the testing of systems and can be used during service life as securing points during tire and landing gear replacement.

Longerons are also used to stiffen the fuselage and support components. Major longerons are placed so that landing gear bays, the weapons bay, canopy, and nozzle openings receive large amounts of local stiffening. Lesser longerons are placed along the aircraft at 12 inch intervals.

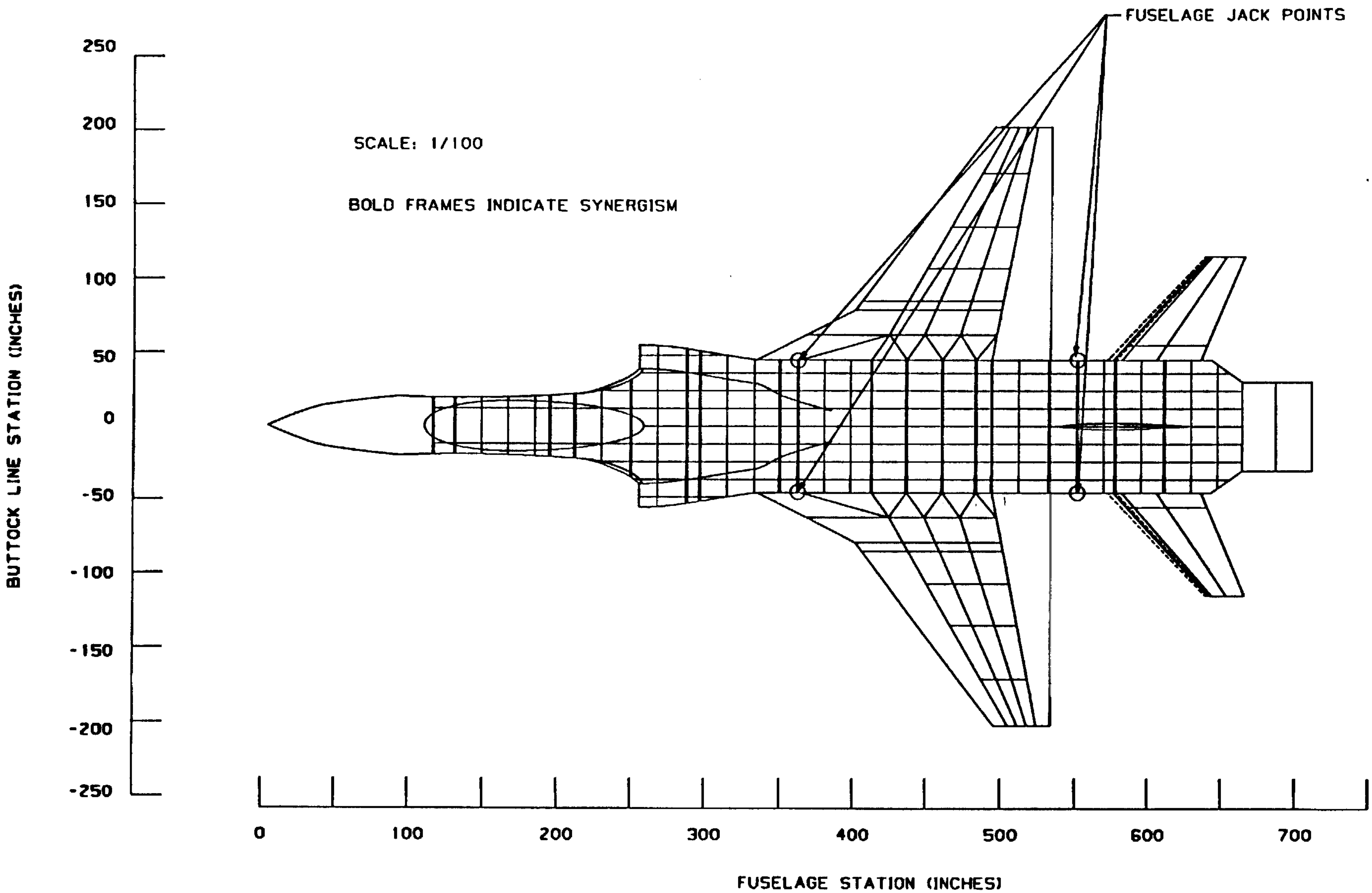

Figure 10.1 Monarch Fuselage Structural Layout

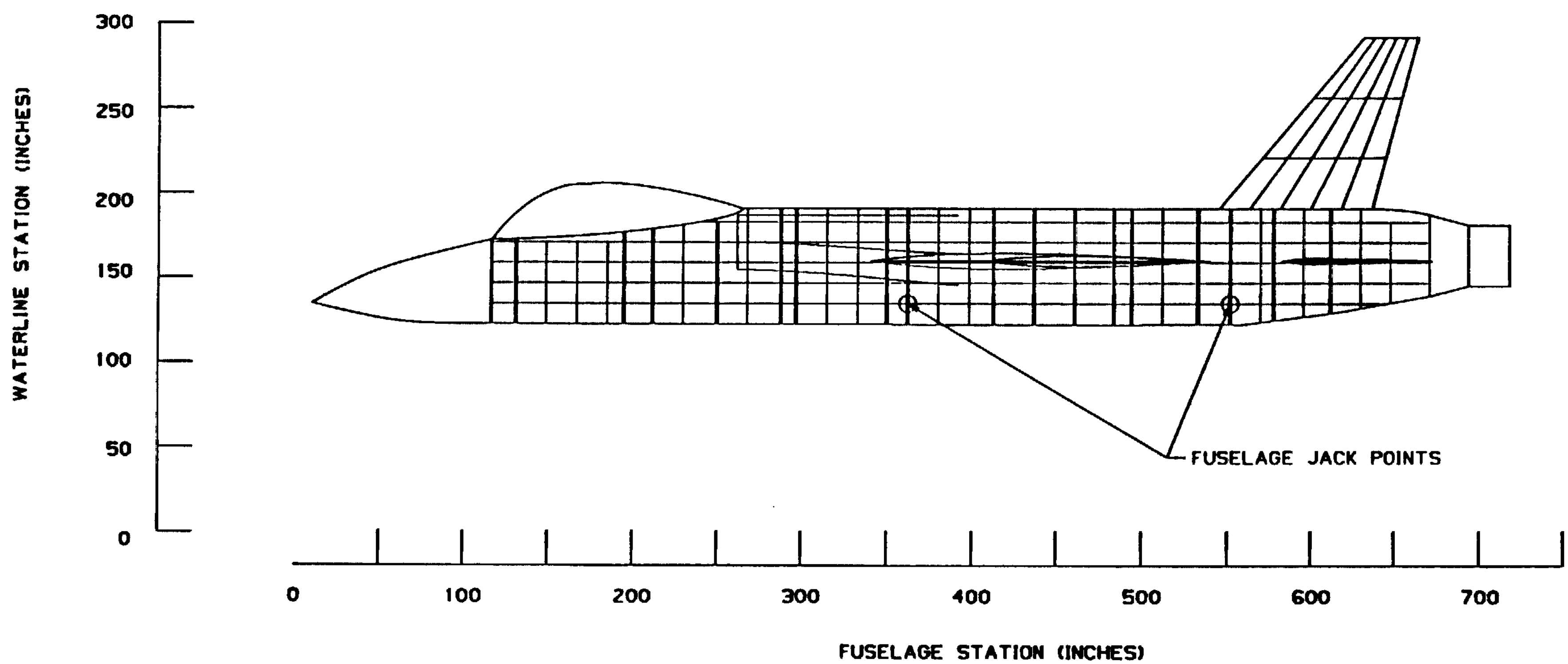

Figure 10.2 Monarch Structural Fuselage Layout

Table 10.1 Fuselage Frame Synergism

1. FS 116.5
   - Radar Mount
   - Nose Gear Support
   - Canopy Attachment
2. FS 131
   - Support Nose Gear Bay
   - Forward Pressure Bulkhead
3. FS 195
   - Nose Gear Attachment Pt
   - Seat Attachment
   - Cockpit Support
4. FS 212
   - Front Engine Mounts
   - DL Inlet Support
   - Pilot Armor/Insulation
   - Rear Pressure Bulkhead
5. FS 250
   - Aft DL Engine Mounts
   - Front Weapons Bay
6. FS 288
   - Inlet Support
   - Fuel Bay support
   - Gun ring Supports
7. FS 297
   - AIM-7 Mount
   - Inlet Support
   - fuel Bay Support
8. FS 350
   - Aft Gun Mount
   - Aft AIM-7 Mount
   - Fuel Support
   - Wing Shelf Support
9. FS 412
   - Aft Weapons Bay
   - Inlet Support
   - Fuel Cell Support
10. FS 421
    - Wing Attachment Point
    - Inlet support
    - Fuel Support
11. FS 437
    - Wing attachment Pt.
12. FS 461
    - Inlet/Compressor Interface
    - Front Main Gear Bay Support
    - Main Gear Attachment
13. FS 493
    - Shelf Attachment
14. FS 533
    - Main Engine Mounts
    - Shelf Support
15. FS 552
    - Vertical Tail Mount
    - Rear Main Gear Bay Support
16. FS 578
    - Vertical Tail Mount
    - Ventral Nozzle Mount
    - Shelf Support
17. FS 612
    - Vertical Tail Attachment
    - Horizontal Tail Attachment
    - Engine Slip Mount

10.1.2. Wing Structural Arrangement

The design drivers for the wing structure of the Monarch are:

* The ability to sustain air loads
* Fuel storage volume
* Lack of wing carry through
* Weapons station requirements
* Control surface requirements
* Weight considerations

The wing structure must be able to support sustained loads of 9g's. The wing is also subjected to fatigue due to gusts and loads caused by deflections of the high lift devices. Locations for weapons carriage must also be provided.

The wing of the Monarch is to be used for fuel storage. This requires that the volume of the structure be kept to a minimum. Additional structure such as baffles and allowances for fuel tank access must also be made. These requirements act contrary to the requirement for a minimized component weight.

Wing spar carry-through was not possible as the Monarch is a mid-wing configuration and carry through spars would conflict with the engine section. This required that additional support be provided where the wing joins the fuselage. Lack of wing carry-though is not uncommon in fighters (see Table 10.2), but it does result in a weight increase at the fuselage/wing interface.

Table 10.2 presents data for wing structures used in other fighter aircraft. This information was used to determine the structural layout for the wing of the Monarch. Actual sizing of the wing members is not possible until loads are calculated.

The wing structural layout of the Monarch fighter is shown in Figure 10.3. The structure consists of four spars, seven ribs, and a "shelf". The number of spars is less than that used by most fighters. This is assumed possible through the use of titanium spars and highly stressed skins. The wing attaches to a "shelf" much in the manner of the F-16 (see Figure 10.4). Structural components indicated with letters in Figure 10.3 collectively make up the shelf of the Monarch. Spar attachment points and rib locations are given in Table 10.3. Ribs are used as divisions in the fuel tanks as well as to provided stiffness. Weapons hard points are installed at B.L. station +/-87 and +/-135 at fuselage stations 470 and 490, respectively (Reference 10.9).

Structural synergism was achieved at rib numbers 3, 5, and 7, and at spar numbers 1 and 3. Rib numbers 3 and 5 act as weapons hard-points as well as wing stiffeners. Rib number 7 act as both the spar cap and the mount of the AIM-9 launch rail. Spars 1 and 3 support the wing and provide mounting locations for control surfaces.

The materials used in the wing are titanium and composites. The upper wing skins are made of graphite epoxy. The spars and ribs are made from Ti-6Al-4V titanium alloy. The lower skins are made from boron epoxy. The leading edge devices are made with aluminum skins and an aluminum honeycomb core. Trailing edge devices are graphite

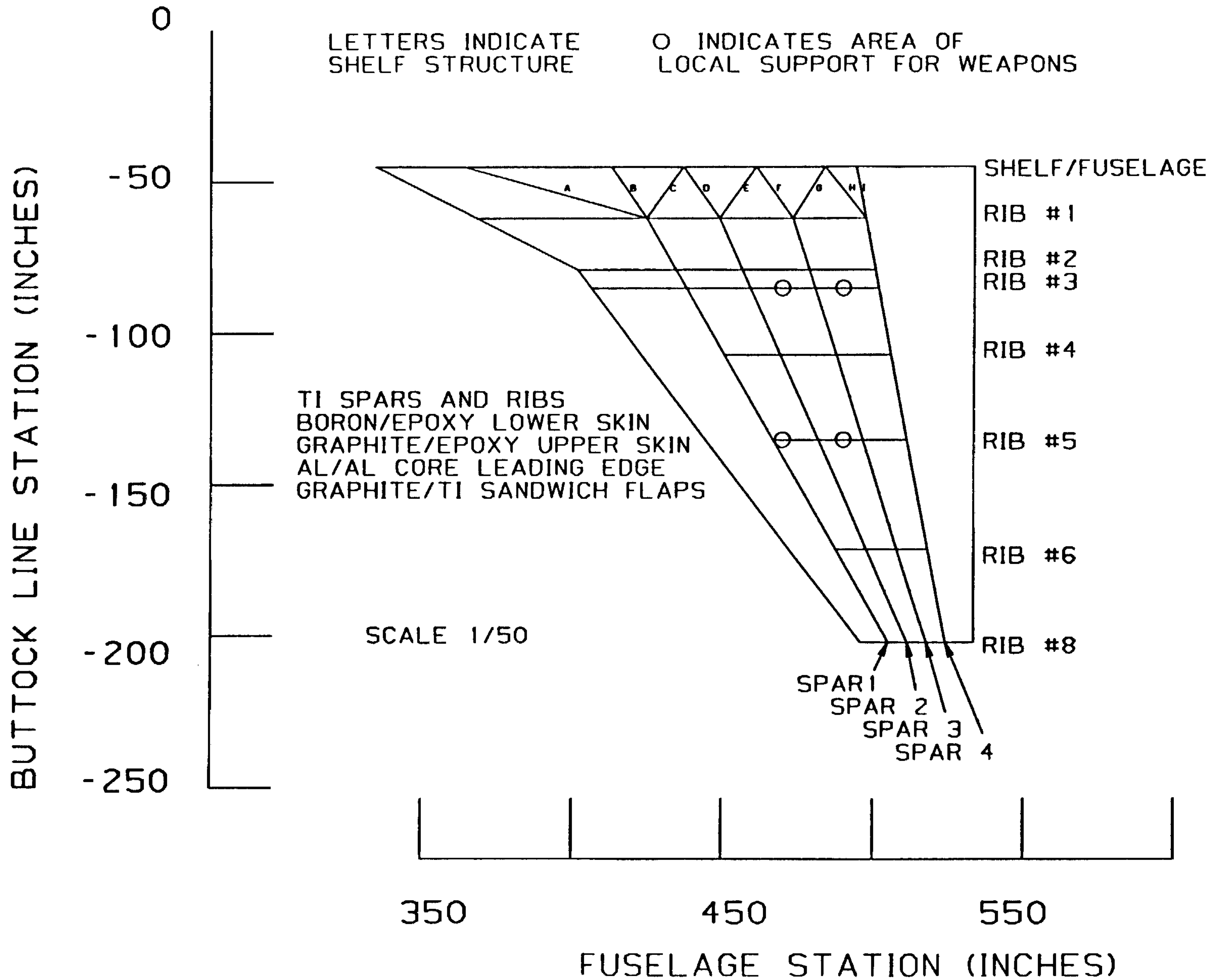

Figure 10.3 Wing Structure Layout

Source: Reference 10.2

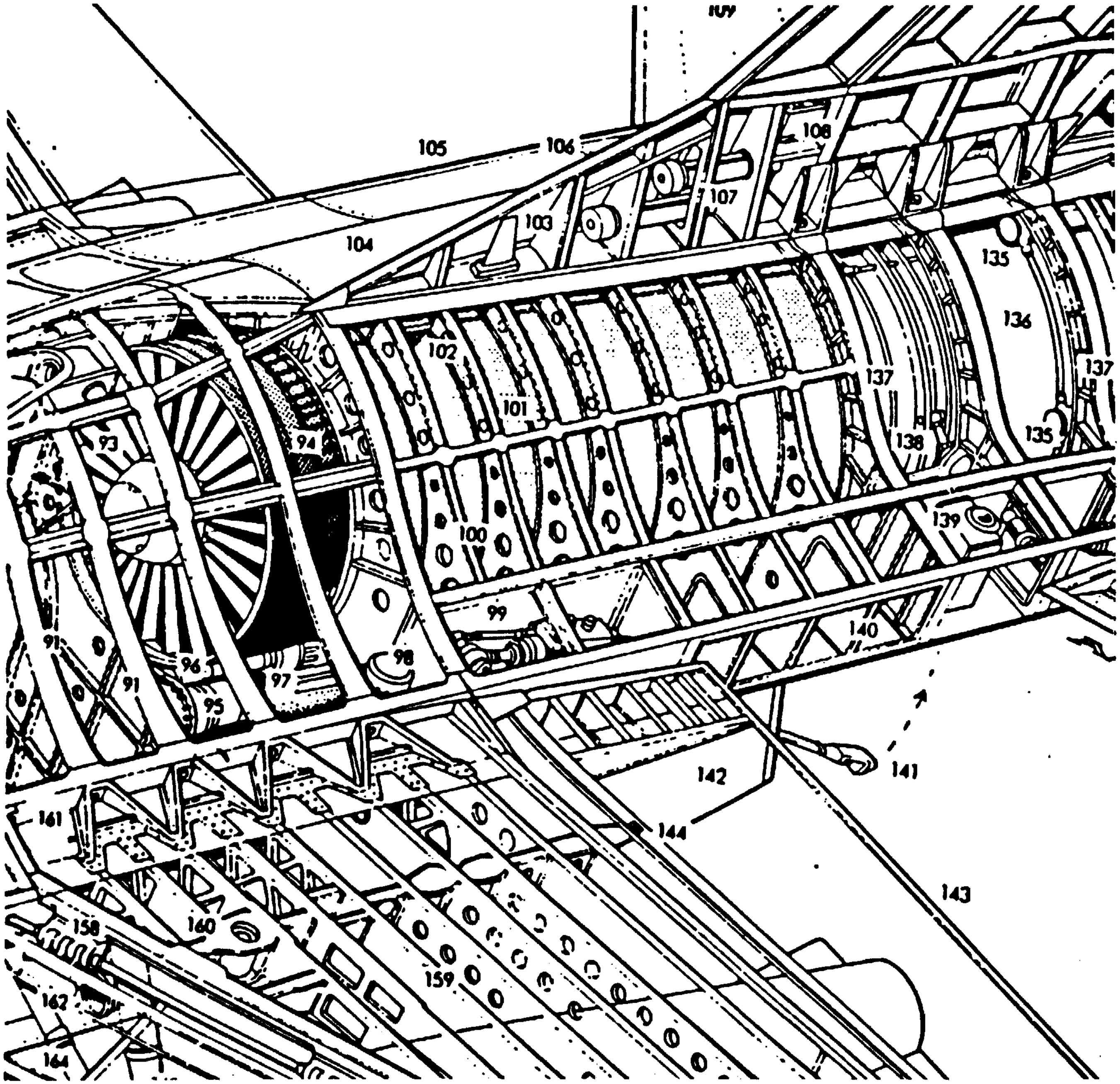

Figure 10.4 F-16 Wing Mounting Shelf

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epoxy over a titanium core. Justification for the selection is provided in section 10.2.

Table 10.2 Fighter Wing Structure Data

Aircraft	Spars	Ribs	Carry Through
F-15	5	10	No
F-16	9	11	No
AV-8B	8	6	Yes
MiG-21	5	NA	No
MiG-23	4	12	No
MiG-25	4	4	No
A-4	3	7	Yes
F/A-18	6	NA	No

Source: References 10.2 and 10.8

C-3

Table 10.3 Wing Structural Layout

Spar	Shelf Attachment (F.S.)	WL Station
1	425	160
2	450	160
3	475	160
4	500	160

Rib #	BL Station	WL Station
Shelf	+/-44	160
1	+/-61	160
2	+/-77	160
3	+/-87	160
4	+/-107	160
5	+/-135	160
6	+/-171	160
7	+/-202	160

10.1.3. Horizontal Tail Structure

Design drivers for the horizontal tail were the desire to reduce weight, increase heat resistance, and create structural synergism. Weight reduction is achieved through the use of composites and a sandwich/core structure. Synergism was achieved by placing the horizontal tail attachment points at the same fuselage frame as the vertical tail and engine slip ring.

Table 10.4 presents a data base of aircraft which use differential stabilizers. Based on this information, the structural layout of the horizontal tail was chosen.

Table 10.4 Fighter Stabilizer Data

Aircraft	Spars	Ribs	Material
F-111	5	4	Boron Epoxy
Mig-23	3	8	NA
F-14	NA	NA	Boron Epoxy
F-15	2	3	Al core Graphite Epoxy Skins
F/A-18	NA	NA	Al core Graphite Epoxy Skins

Source: References 10.1, 10.2, and 10.3

The horizontal tail structural layout of the Monarch fighter is shown in Figure 10.5. The structure consists of two spars and three ribs. The tail attaches to the fuselage at the second spar as indicated Table 10.5. The first spar acts as a re-enforcement for the leading edge and an attachment point for the fixed slat. Rib number one acts as a local stiffener at the attachment point. Rib number three acts as a spar cap. The basis for this layout is the F-15 which uses an almost identical arrangement (Reference 10.2).

The F-15 uses an aluminum honeycomb core with graphite epoxy skins (Reference 10.1). Aluminum spars and ribs, and a core of titanium honeycomb are proposed for the horizontal tail of the Monarch. The skins are made of a carbon/carbon composite. A justification of the materials selection is presented in section 10.2. A diagram showing the actuator mechanism for the horizontal tail is shown in Figure 10.6 and disucssed further in Chapter 11.

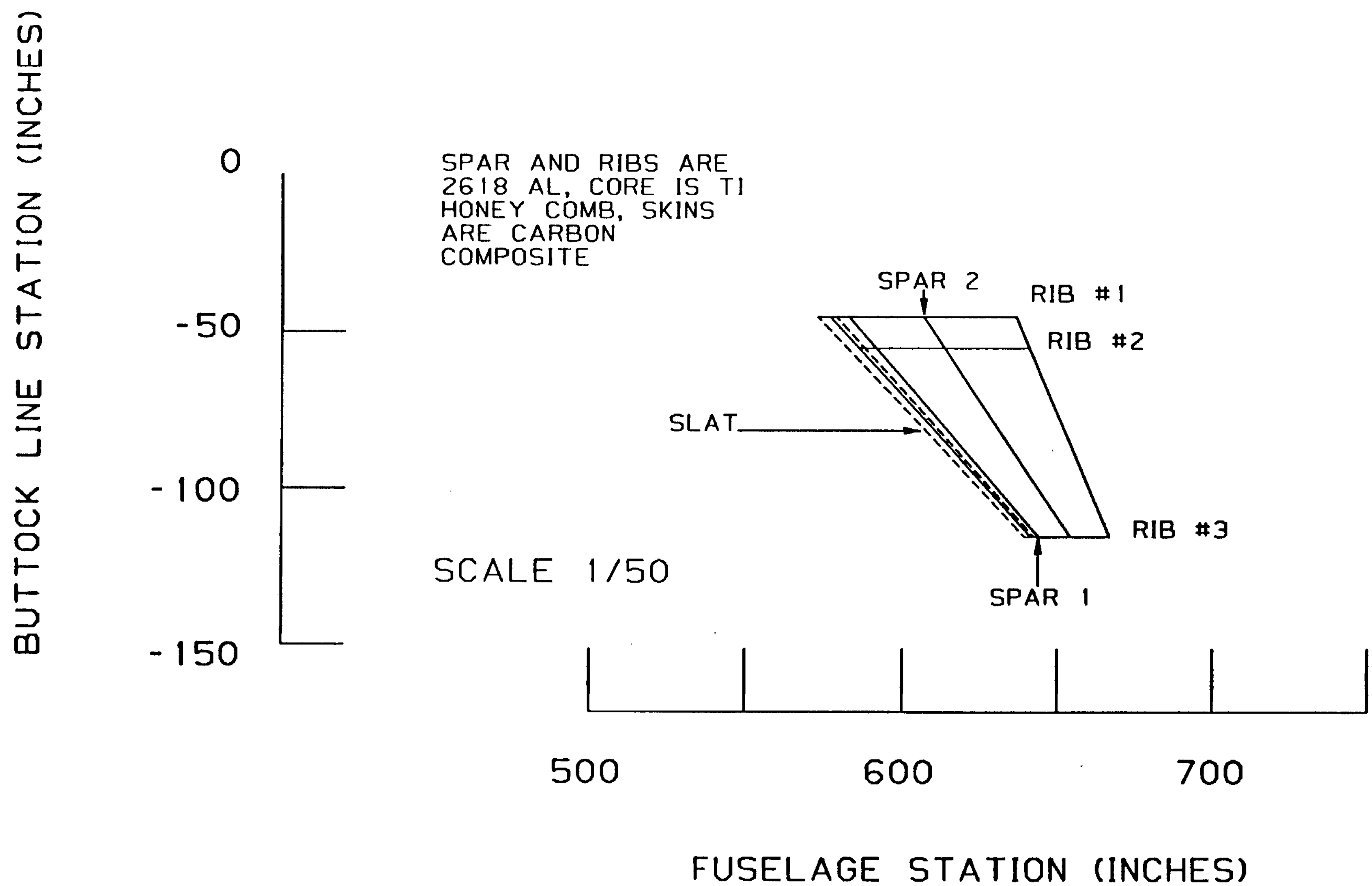

Figure 10.5 Horizontal Tail Structural Layout

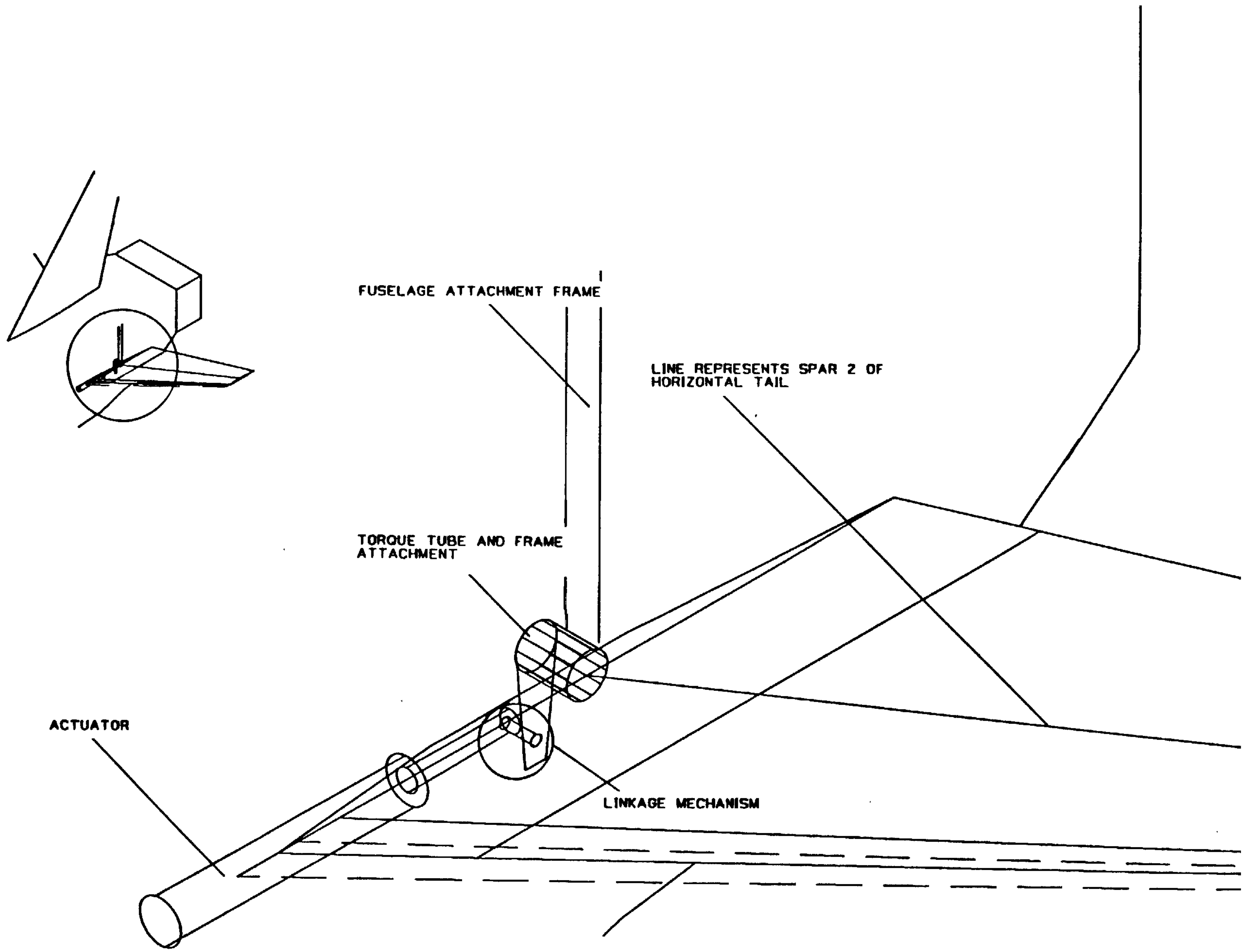

Figure 10.6 Horizontal Tail Actuation Mechanism

Table 10.5 Horizontal Tail Structural Layout

Spar #	WL Station (inches)	FS Station (inches)	BL Station (inches)	Root chord Location (%C, inches)
1	160	582.7	45	10% (6)
2	160	606.7	45	50% (30)

Rib #	WL Station	FS Station	BL Station	Span Location
1	160	587	45	7.5% (5)
2	160	595	55	9% (6)
3	160	648	115	100% (65)

10.1.4. Vertical Tail Structural Arrangement

The purpose of this section is to present the vertical tail structural arrangement for the Monarch aircraft. The design considerations and a data base for comparison are presented. The structural layout, materials used, and structural synergism are also given.

The primary design drivers for the vertical tail are:

* Ability to sustain air loads and provide control.
* Battle damage tolerance
* Structural Synergism
* Ability to store antennas, IFF gear, etc.

The vertical tail must be able to withstand sustained and repeated combat air loads. Fatigue due to gusts is part of this consideration. The vertical tail structure should also be able to tolerate the loss of some members without catastrophic failure. This requires that the structure be formed to transmit loads around severed members, or that members be made redundant. The second method results in a weight penalty. The first method requires additional detailed design. It was assumed that other fighters must meet these same requirements, so that by using a similar structure this requirement would be met for the Monarch. This design was coupled with composite skins which transmit loads around damage to create better short term damage tolerance than metal skins (Reference 10.6).

Synergism for the vertical tail was achieved by placing its spars such that the ribs support:

* Chaff and flare dispenser
* Fuselage frame/slip ring
* The horizontal tail attachment

The size of the vertical tail results in useful internal volume. Synergism is added by using the room to store components such as IFF and radio antennae in the vertical tail. This allows the receivers to be located away from the interference caused by aircraft systems.

Table 10.6 presents a data base of the vertical tail structures used in other fighter aircraft. The aerodynamic loads that the vertical tail will experience have not yet been calculated. For this reason, the vertical tail structure is based on this data.

The vertical tail structural layout of the Monarch fighter is shown in Figure 10.7. The structure consists of four spars and four ribs. This selection agrees with the data base. The tail attaches to the aircraft at the location indicated in Table 10.7. Attachment is to the "spine" longeron and fuselage frames. The ribs provide for tail stiffness at mounting points, an equipment mounting shelf, and as a means for loads to be transmitted in case of spar failure. Rib number four acts as a way to dissipate lightning strikes. The spars provide stiffness and redundancy for battle damage.

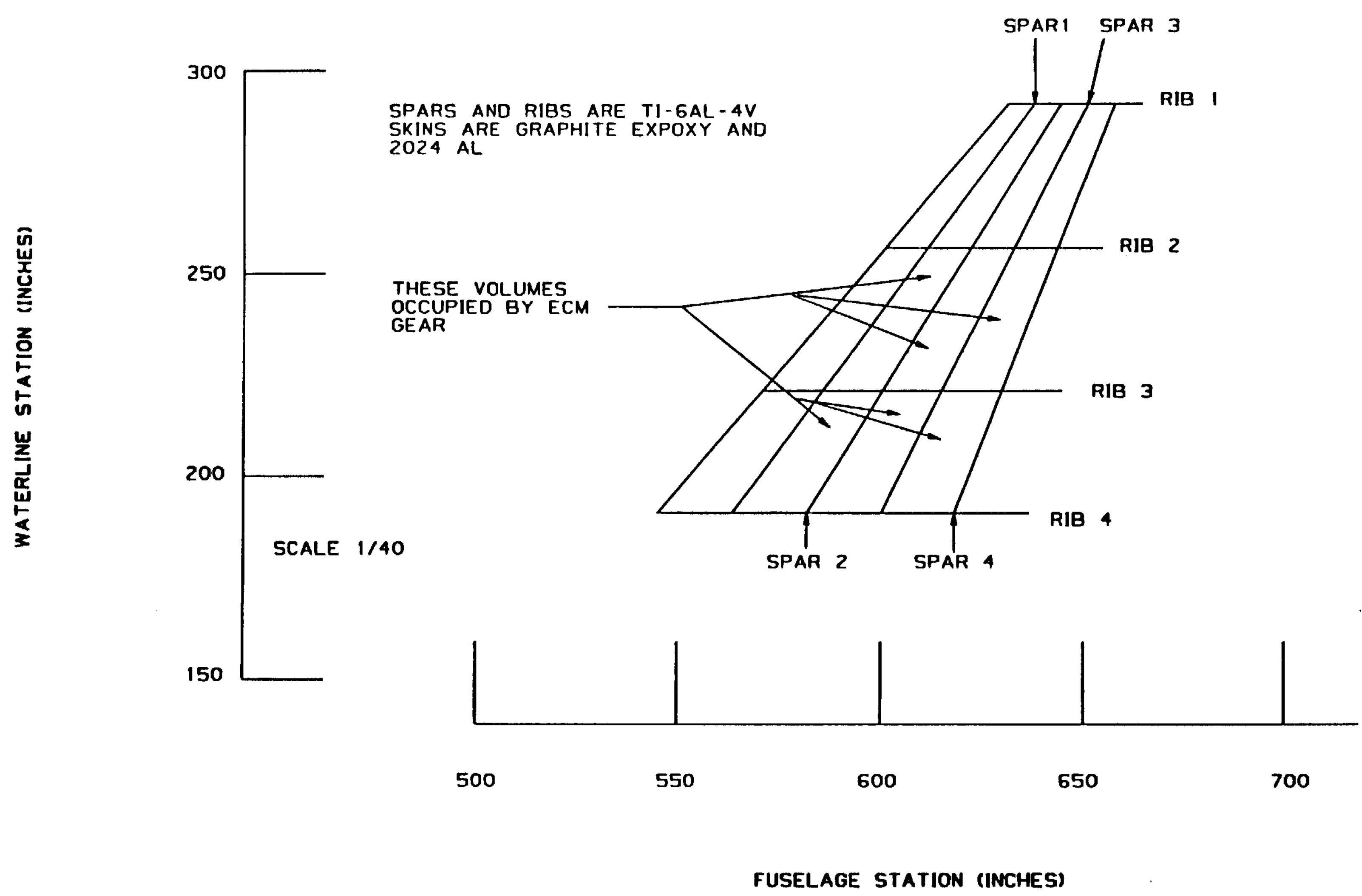

Figure 10.7 Vertical Tail Structural Layout

Table 10.6 Fighter Vertical Tail Structure Data

Aircraft	Spars	Ribs	Material
F-4	3	NA	Ti, Al
F-15	2	1	Ti, Boron Epoxy
F-16	4	3	Al, Graphite Epoxy
F/A-18	6	0	Ti, Graphite Epoxy
F-111	6	7	Al, Graphite Epoxy

Source: References 10.1 and 10.2

The primary materials used in the vertical tail are 2024 Aluminum, Ti-6Al-4V titanium alloy, and graphite epoxy. Aluminum is used from the leading edge of the vertical tail to spar #1. Titanium alloy is used in the attachment points, spars, and ribs. Graphite epoxy skin is used with the exception of the leading edge.

Table 10.7 Vertical Tail Structural Layout

Spar #	WL Station (inches)	FS Station (inches)	Root Chord Location (%C, inches)
1	192-291.8	563.5	15% (16.2)
2	192-291.8	581.8	32% (34.2)
3	192-291.8	600.1	48% (52.2)
4	192-291.8	618.4	65% (70.2)

Rib #	WL Station (inches)	Span Location (%b, inches)	Function
1	191	0 (0)	Attachment point
2	221	30% (30)	Equipment Shelf
3	234.3	66% (66)	Stiffness
4	291.8	100% (100)	Lightning Dispersal

10.2. MATERIALS SELECTION, JUSTIFICATION, AND LAYOUT

The purpose of this section is to present the materials layout and selection justification for the Monarch. Design criteria for materials selection are given by Reference 10.7 as:

* Mechanical Properties
  - -Static Strength Efficiency
  - -Fatigue
  - -Fracture Toughness and Crack Growth
  - -Environmental Stability
* Fabrication Characteristics
  - -Availability and Productibility
  - -Material Costs
  - -Fabrication Characteristics

Other considerations for the Monarch are weight savings, damage tolerance, and cost.

Weight savings are achieved through the use of composite materials and materials with high strength to weight ratios. Damage tolerance is achieved by using materials that have high toughness and redundant structure. Damage tolerance is further increased by using structural methods that redistribute loads well. An example of this method is sandwich/honeycomb structure which is used in the trailing and leading edge surfaces.

Many of the materials selected for the Monarch have high initial costs. However, it is possible that these costs are regained through the life cycle of the aircraft by better performance. References 10.4, 10.5, and 10.14 indicate that materials such as composites and titanium offer better fatigue characteristics, weight reductions or both. The materials of the Monarch are chosen to be light and have good fatigue properties. If it is possible to capitalize on these properties, life cycle cost may be lowered through reduced fuel consumption and maintenance requirements. The materials of the Monarch have been selected with these possibilities in mind.

This section is divided by materials type. The location of materials is as indicated in Figure 10.8. Table 10.8 and Figures 10.9-12 provides a data base for comparison of materials usage in the Monarch and current fighters. The materials used in the Monarch are as follows:

Aluminum Alloys

This alloy is used primarily in the fuselage and horizontal tail.

Fuselage Frames: Those frames which are not in engine heat fields or heavily stressed are made of 2219 Al-Cu. 2219 Al-Cu is relatively tough and resists corrosion cracking well (Reference 10.4). It has a yield strength of about 60,000 psi (in tension), and resists creep. 2219 does not retain strength well above 200 F. Manufacturing is relatively easy and a large number of suppliers exist.

Table 10.8 Aircraft Materials Breakdown

Aircraft	Al	Steel	Composites	Ti	Other
HIMAT	25	9	29	19	18
B-52	69.7	11.5	1.6	1.5	15.7
F-14	36	15	4	25	20
F-15	37.3	5.5	NA	25.8	NA
F-16	80	8	3	1.5	7.5
F/A-18	49.6	12.9	9.9	12.9	14.7
AV/8B	47.7	NA	26.3	NA	NA

All values in percentage of aircraft take-off weight
Note: Data for AV/8B is for skins only

Source Reference 10.1 and 10.2

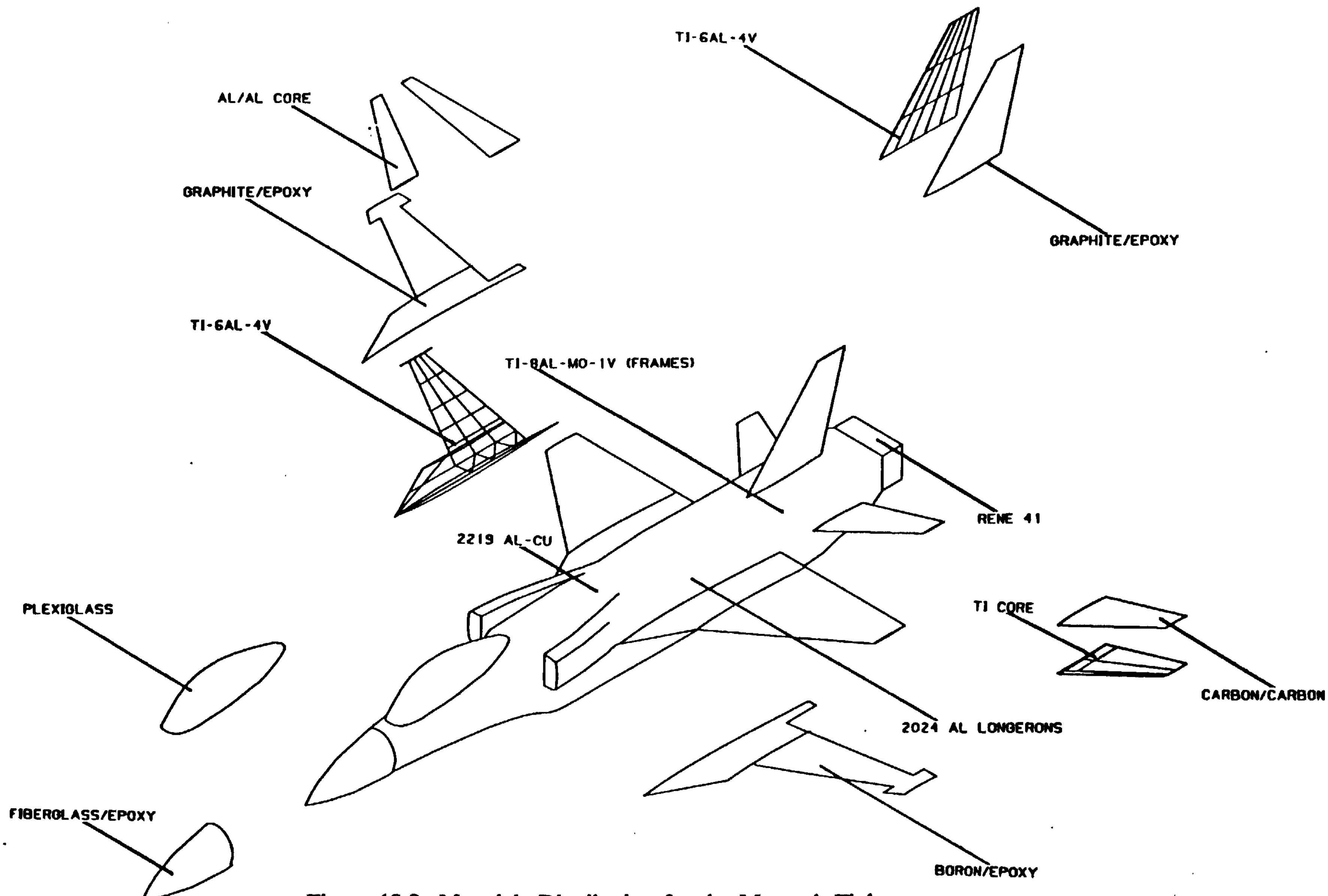

Figure 10.8 Materials Distribution for the Monarch Fighter

Source: Reference 10.2

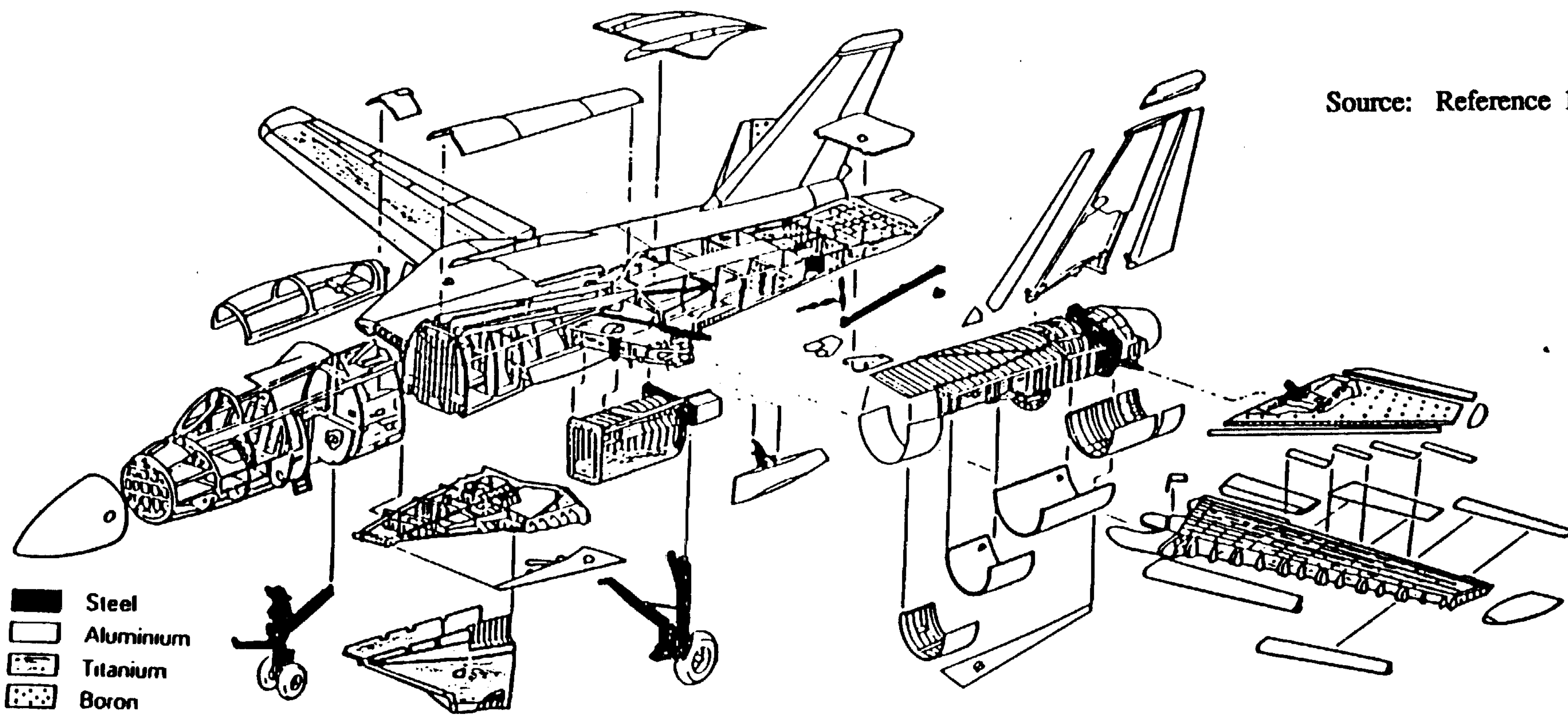

Figure 10.9 Materials distribution for the F-14

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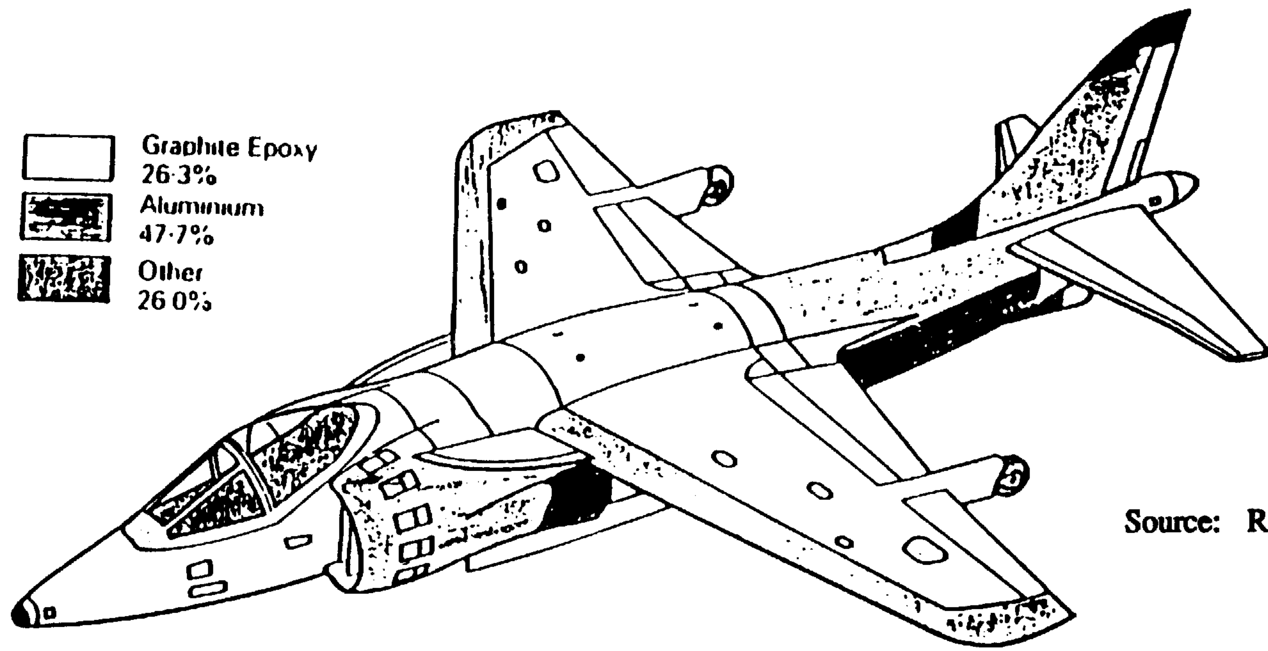

Source: Reference 10.2

Figure 10.10 AV/8B Structural Materials

Source: Reference 10.2

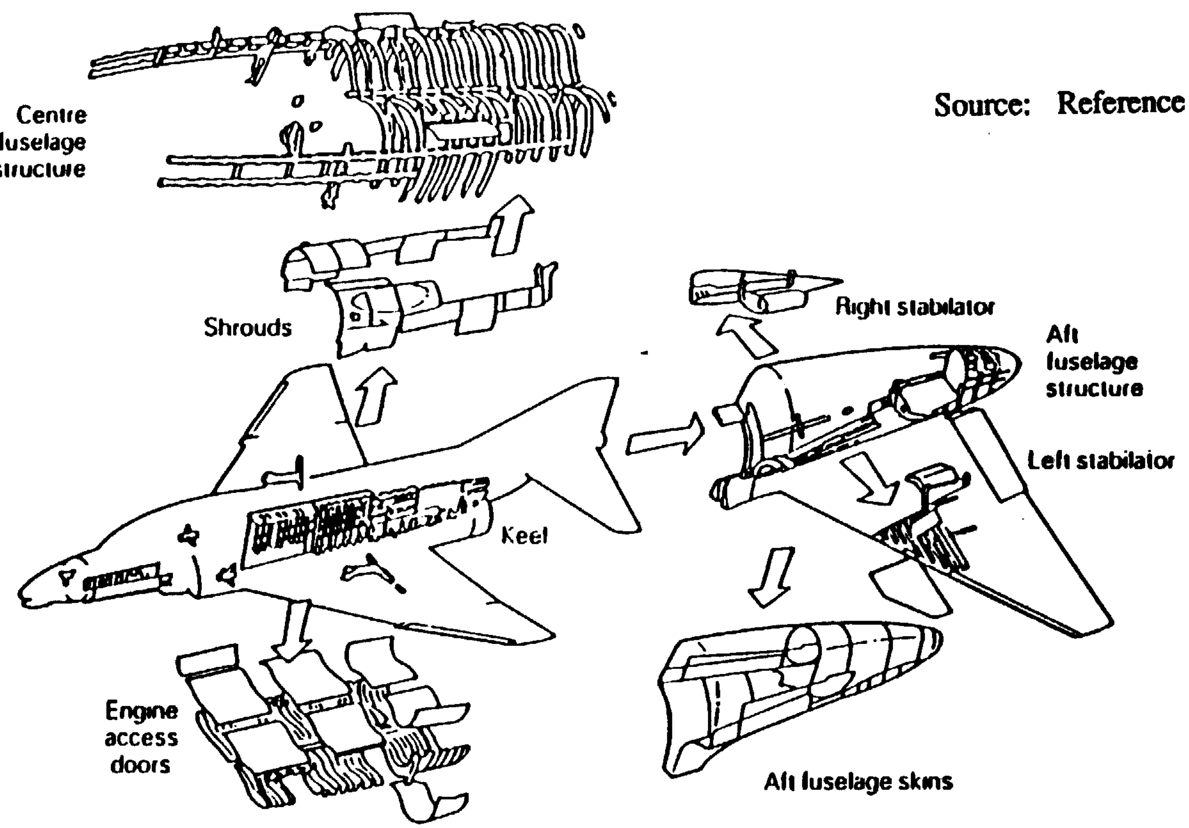

Figure 10.11 Titanium Components for the F-4

Source: Reference 10.12

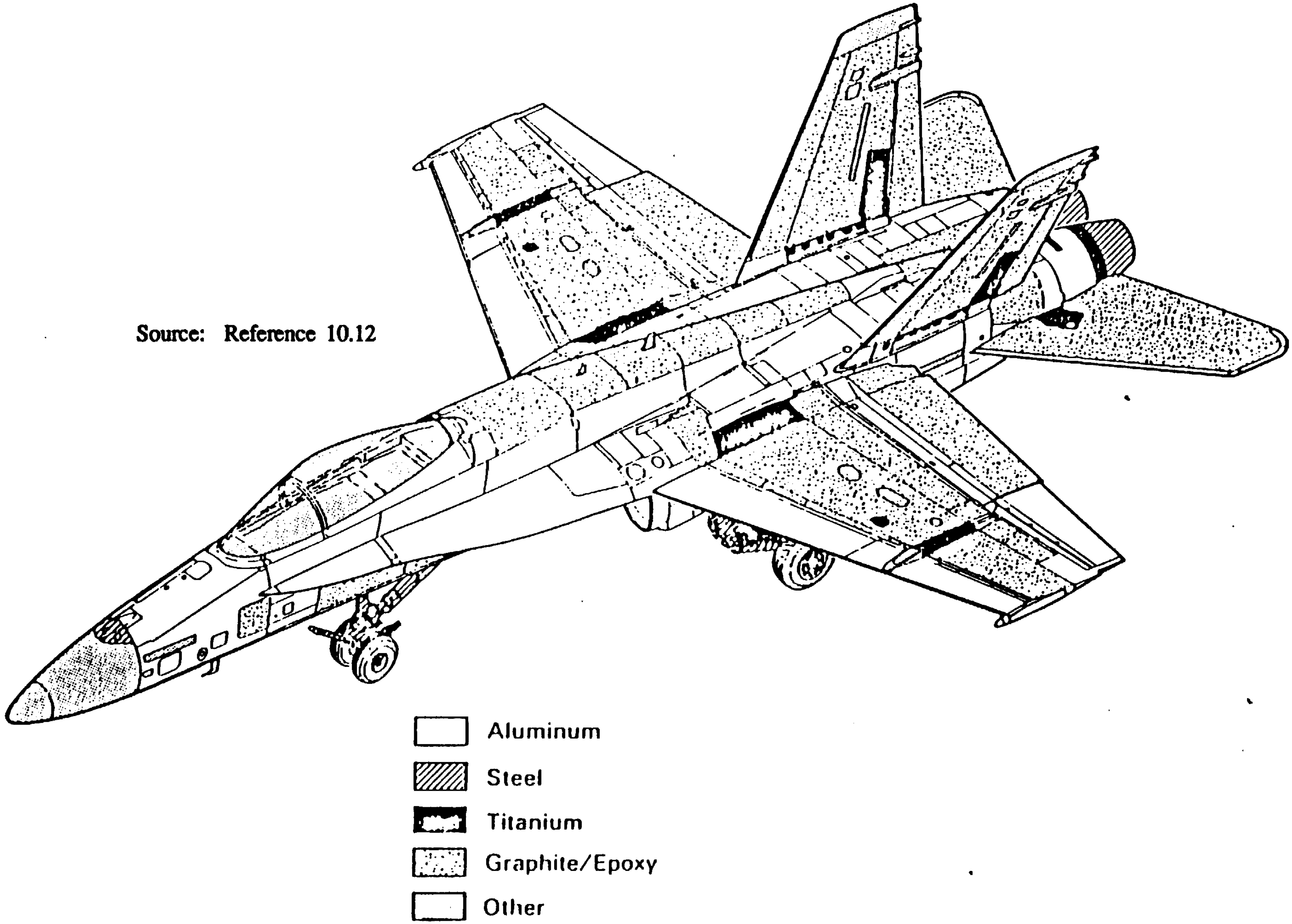

Figure 10.12 Materials Distribution of the F/A-18

Horizontal Tail: 2219 Al-Cu was also used in the spars and ribs of the horizontal tail. This material was chosen due to its resistance to creep and ability to tolerate temperatures up to 600 F (Reference 10.4). Although exhaust air through the yaw port is in the vicinity of 1400 F, loss of strength due to the heat effects of the yaw ports was not considered detrimental. This assumption was made due to the fact that the yaw ports are open only for short periods of time (see Chapter 7), the cooling effect of free stream air, and protection by the skins.

Stringers: 2024 was used in fuselage stringers. This material was chosen on the basis of its ability to be heat treated to high strengths and toughness, its cost, and availability. 2024 is also creep resistant at elevated temperatures (Reference 10.7).

Fuselage Skins: Large portions of the fuselage skins need to be aluminum due to heat fields and the possibility of foreign object damage. The exact alloy will depend on the local heat fields and strength requirements.

Titanium Alloys:

Titanium is used in the wing, vertical tail, engine section, horizontal tail, and fuselage frames.

Wing: Titanium is used in the wing spars and ribs. This material was chosen based on its high strength-to-weight ratio (1.3 that of Al, Reference 10.4) and good corrosion and fatigue characteristics. Titanium also has a low thermal expansion coefficient. This allows good bonding at metal/composite interfaces. Wing/Fuselage attachment points are made of Ti-6Al-4V prepared using powder metallurgy techniques. Reference 10.5 indicates that exceptional fatigue and crack stoppage is possible with this material. The penalty is a very slight decrease in tensile strength.

Vertical Tail: Ti-6Al-4V is used for the spars and ribs for the same reasons given in the wing description.

Engine Sections: Ti-8Al-Mo-1V is used in the structure surrounding the engine section of the Monarch. This material offers good creep and thermal stability characteristics up to 850 F (Reference 10.5). An additional benefit is that the material can be welded.

Horizontal Tail: A titanium honeycomb is used as filler between skins. This application offers high strength and good heat resistance. This style of application is more tolerant of battle damage than that with only spar/skin arrangements (Reference 10.6).

Fuselage Frames: Ti-6Al-4V is used in fuselage frames for attachment points. This choice was based on the strength-to-weight ratio of this material.

Steels:

Steel alloys are used in various applications including landing gear, fasteners, and other components which require high strength.

Composites:

Various composite materials are used in the skins of the Monarch fighter. In general, composites are light, strong, and have good corrosion resistance. Methods have been developed by McDonnell/Douglas for repairing battle damage to composites and the AV/8B uses large amounts of graphite/epoxy. A precaution against lightning and bird strikes exists in the form of metal leading edges. Given that composites are able to operate with field repairs and tolerate the heat fields found in hover, the precautions mentioned should make composite materials extremely serviceable for the Monarch.

Vertical Tail Skin: Graphite Epoxy is to be used for this application to achieve weight savings and high strength.

Horizontal Tail Skin: A carbon/carbon composite skin is used to provide heat resistance at low weight. Heat resistance is required due to the location of yaw RCS ports.

Wing: Boron Epoxy composites are used on the lower surface of the wing due to high strength/weight values in tension, heat resistance, and corrosion resistance. Graphite Epoxy is used in the upper wing surfaces because it is cheaper and heat resistance is not so crucial. Note that Boron poses some environmental problems which have been considered.

Fuselage: Graphite Epoxy is used in all access panels as a method of weight savings.

Other Materials:

Various other materials are to be used in the construction of the Monarch. A non-exhaustive list of examples includes:

* Rubber (tires)
* Plexiglass (canopy)
* Composites (radome)
* Rene 41 (nozzles)

10.3. MANUFACTURING BREAKDOWN AND PROCESS

The purpose of this section is to present the manufacturing process and shop flow for the Monarch fighter. The manufacturing breakdown is shown in Figure 10.13. The shop area required for the Monarch is presumed to be 50,000 sq. ft. based on comments from Reference 10.15. This area is to house all stages of production at a peak production rate of 10 aircraft per month.

Several different processes will be used in the manufacturing of the Monarch. Raw materials will be received in a storage and testing area where a quality control group will verify that the materials meet specifications. Required materials are then requisitioned from this stockpile as needed. Fuselage frames are milled using computer controlled milling machines, or forged and heat treated. Wing, vertical tail, and horizontal tail structures are created from standard bar stock. This stock is formed using a number of methods including milling, rolling, and drawing. It may be possible to purchase the wing and tail structures in finished, unassembled form from subcontractors. Fuselage skins are cut from

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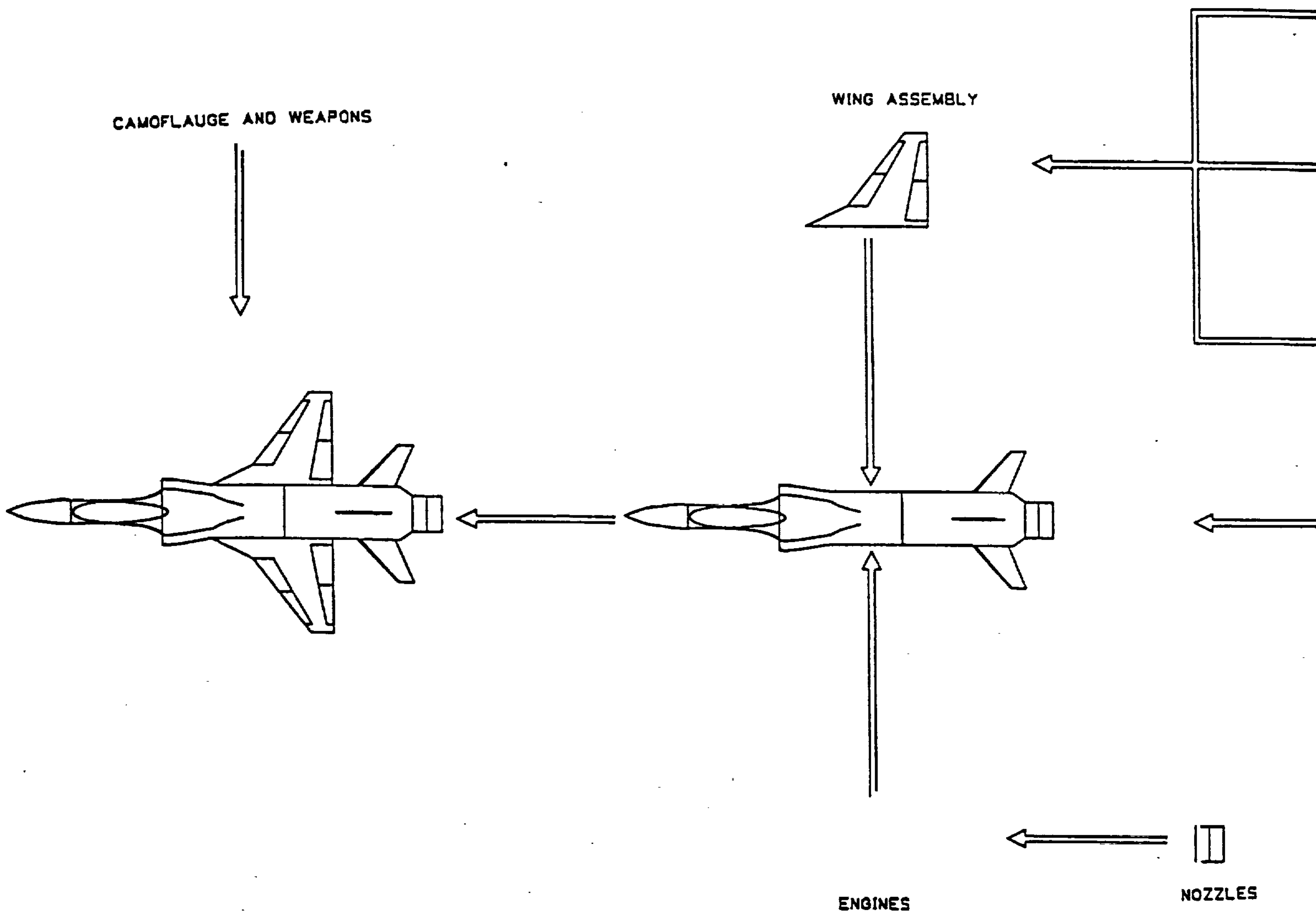

Figure 10.13 M

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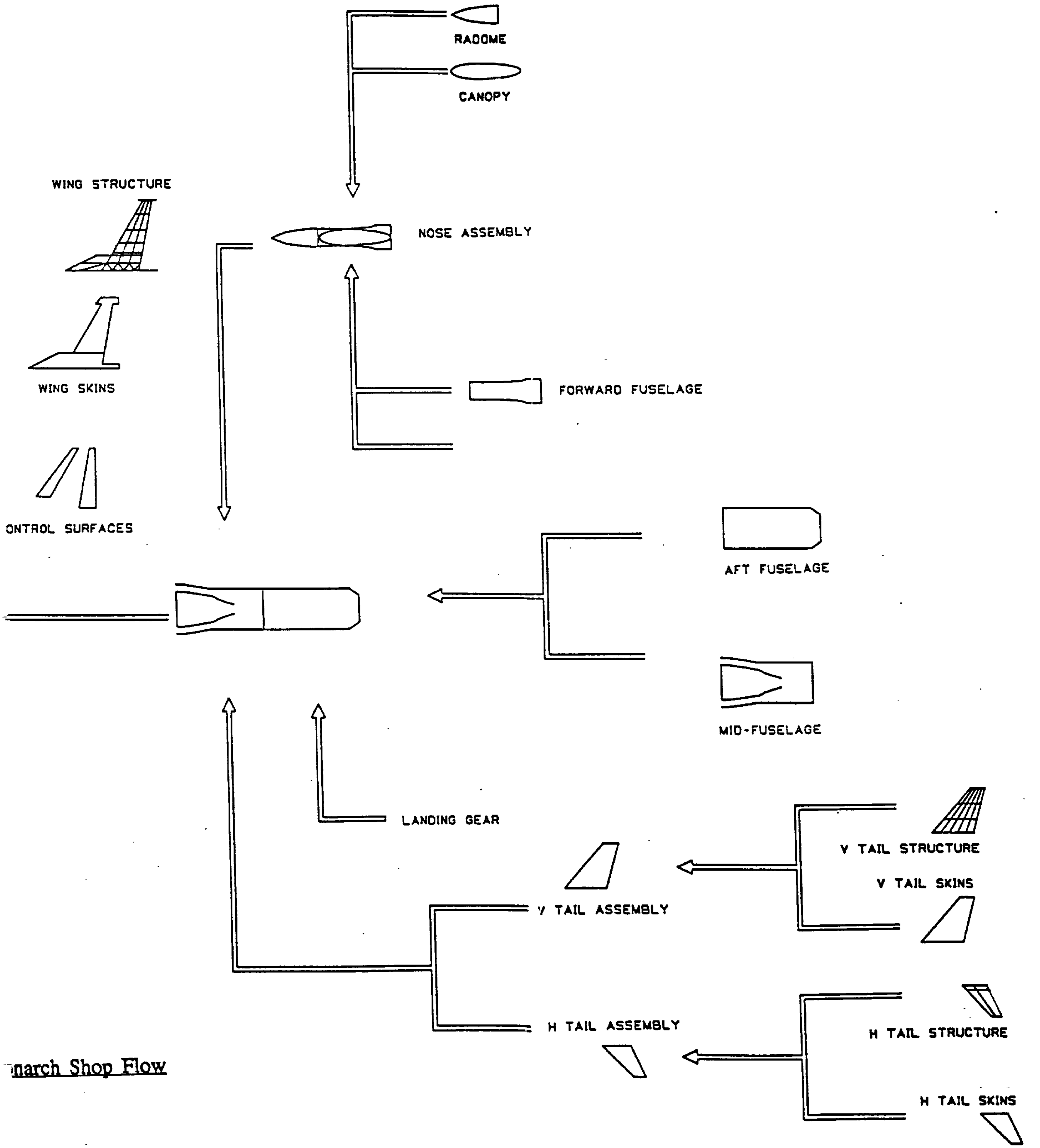

narch Shop Flow

sheet stock and formed using stamping or plastic forming methods. Wing and tail skins are made from autoclaving methods described in Reference 10.16. Due to the precise requirements and conical nature of the radome, it is created using a computer controlled filament winding process. Metallic skins will be joined to the fuselage using rivets. Composites will be joined to the structure using titanium/composite lap joints or by riveting through re-enforced holes. Other components such as the canopy, landing gear, and engine are supplied by subcontractors.

The shop flow of the Monarch follows a path consisting of:

1. Production of structure for all components.
2. Installation of systems in aft and mid fuselage sections.
3. Assembly of forward fuselage section including ejection seat, canopy, and skins.
4. Joining of fuselage components.
5. Installation of landing gear and systems.
6. Application of fuselage skins.
7. Addition of vertical and horizontal tail assemblies.
8. Addition of wing assemblies.
9. Installation of engines and nozzles.
10. Final systems check.
11. Application of camouflage paint and addition of armaments.

At each step in the manufacturing process quality control measures should be taken toinsure that all work has been performed correctly. Two types of quality control are currently being used in industry. The most dominant type of quality control establishes a separate department outside of manufacturing to perform checks. This tends to create resentment among the people responsible for manufacturing and adds to clerical and accounting cost by requiring a separate department and staff. An alternative method of quality control which is beginning to appear in U.S. industry is "Total Quality Management" or TQM. Under TQM, the manufacturing group is directly responsible for the quality control process. This reduces staffing costs, reduces worker tensions, and may instill better workmanship by making each individual directly responsible for his work. TQM does suffer the drawback of removing objectivity in the person checking the work. One compromise between these two systems may be to integrate quality control specialists directly into manufacturing groups. These specialists would be able to retain their objectivity as they would be checking the work of others, but they would also be an integral part of the team rather than an outsider. This would reduce clerical costs and departmental conflicts.

10.4. ACCESSIBILITY AND MAINTAINABILITY CONSIDERATIONS

The purpose of this section is to present the design considerations affected by accessibility and maintainability. These two factors heavily influence the life time costs and combat success of a fighter. Easy maintenance reduces manpower costs and increases combat effectiveness by allowing quick repair of battle damage. Examples at the extremes of this scale are the F/A-18 and AV-8B. The U.S. Navy record for engine removal and replacement on the F/A-18 is eight minutes under competition conditions. In comparison, the entire wing of the AV-8B must be removed in order for the engine to be changed.

The accessibility and maintainability features are divided between engine removal and system considerations. Only major access ports are mentioned here as examples. Mentioning all access requirements is not particularly useful at this stage of the design and would be extremely complicated. For example, some 80% of the skins on the F-15 are access panels.

The following discussion focuses on emergency access to primary systems. During the life of the Monarch, several complete overhauls will have to be performed under depot conditions. The work performed during these overhauls will be specified by military regulations and will change as the aircraft ages. For these reasons, depot maintenance is not addressed in this report.

10.4.1. Engine Removal

The engine removal is presented by showing a step by step procedure. Engine removal considerations are:

* Engine accessibility from ground level
* Structural soundness
* Accessibility in all types of NBC and Arctic gear

Removal of the lift engine is straight down through the nozzle opening. This avenue was chosen to take advantage of an existing structural opening, and to avoid breaking frames or disconnecting non-engine systems. For similar reasons, the cruise engine is removed in the aft direction. An engine removal jack is shown in Figure 10.14. This jack is a preliminary design driven by the following considerations:

* Able to remove both engines
* ~~Able to operate over rough ground~~
* Self-powered
* Remotely controlled

The ability to remove both engines is provided in the fork-lift type arrangement for the lift engine and the upper rails for the cruise engine. Rough ground operations are achieved through a wide wheel base and large, soft tires. The jack is to be powered by a 300 hp. diesel or gasoline engine. This size is estimated to be adequate for the powering of all jack systems and ground transportation of both engines. Remote control is desirable as it allows the mechanic to position the jack in the correct relationship with the removal devices without requiring that directions be relayed through a second party. This is accomplished by connecting a hand held device to the engine controls through electric cords.

Figures 10.15 and 10.16 show the mountings and access panels for the lift engine. Removal procedure for this engine is:

1. Disconnect engine fuel and coolant systems through access panel "1."

2. Remove access panel "2" around the engine nozzle.

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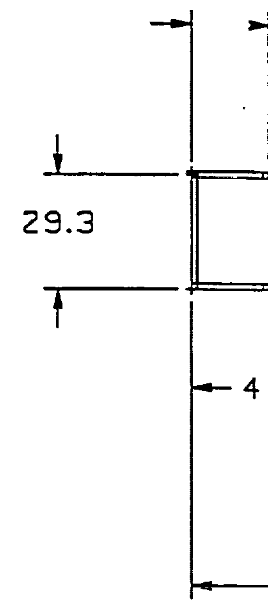
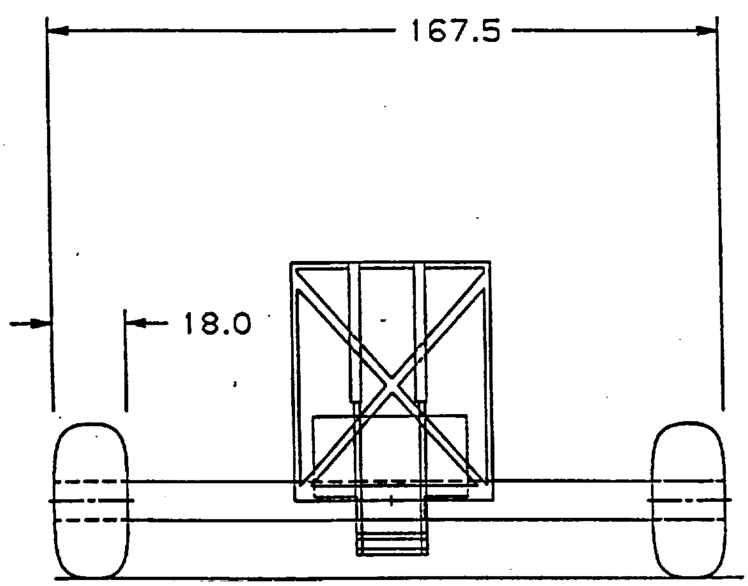
NOTES: ALL DIMENSIONS IN INCHES
POWER SUPPLIED BY 300 HP AUTOMOTIVE ENGINE
ALL MEMBERS CURRENTLY 1.5 IN STEEL BAR

SCALE: 1/50





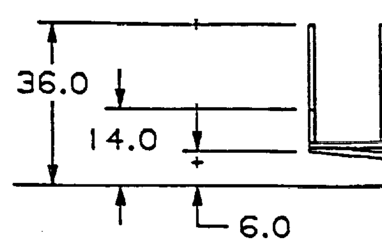


Figure 10.14 Engine Re

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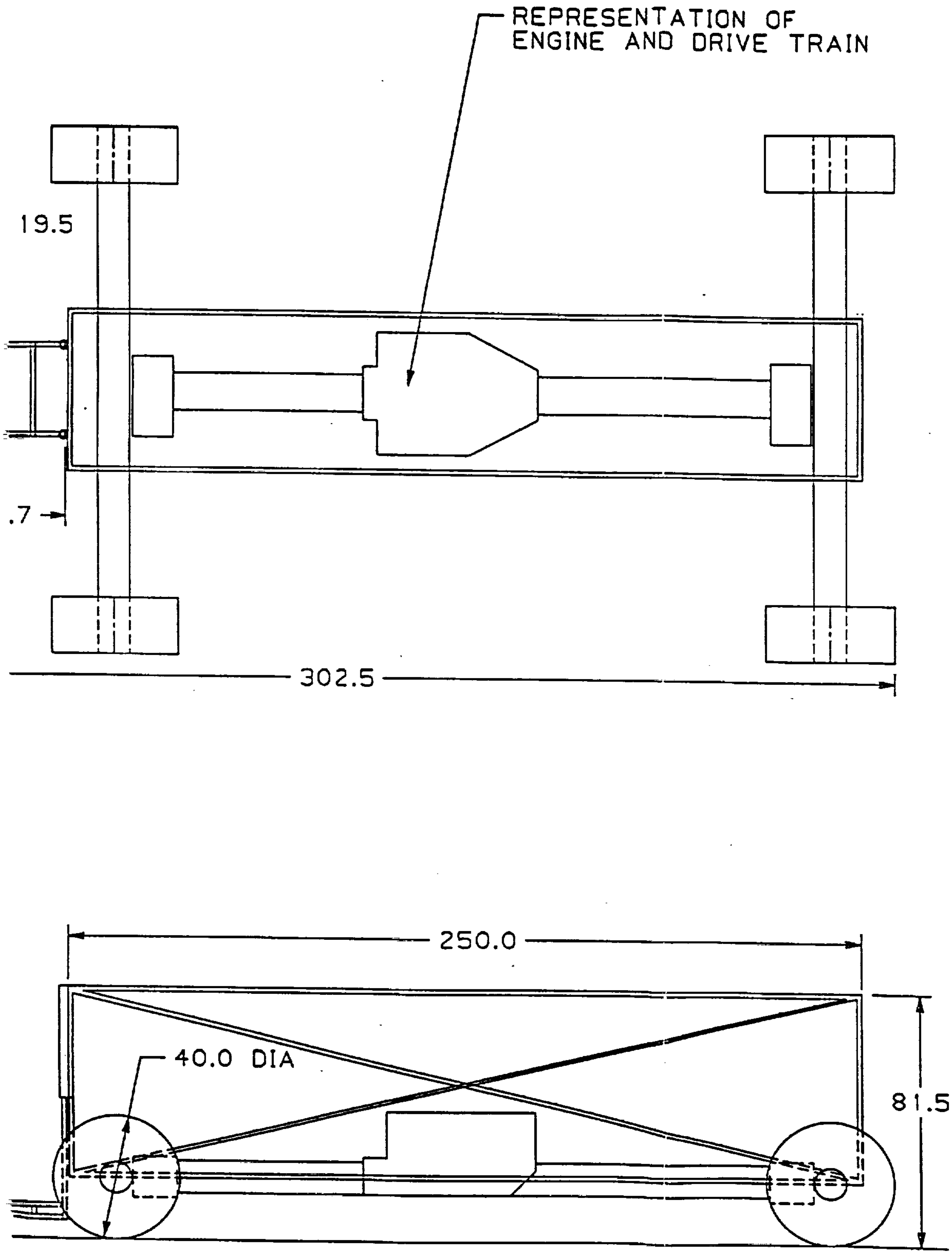

ıoval Jack

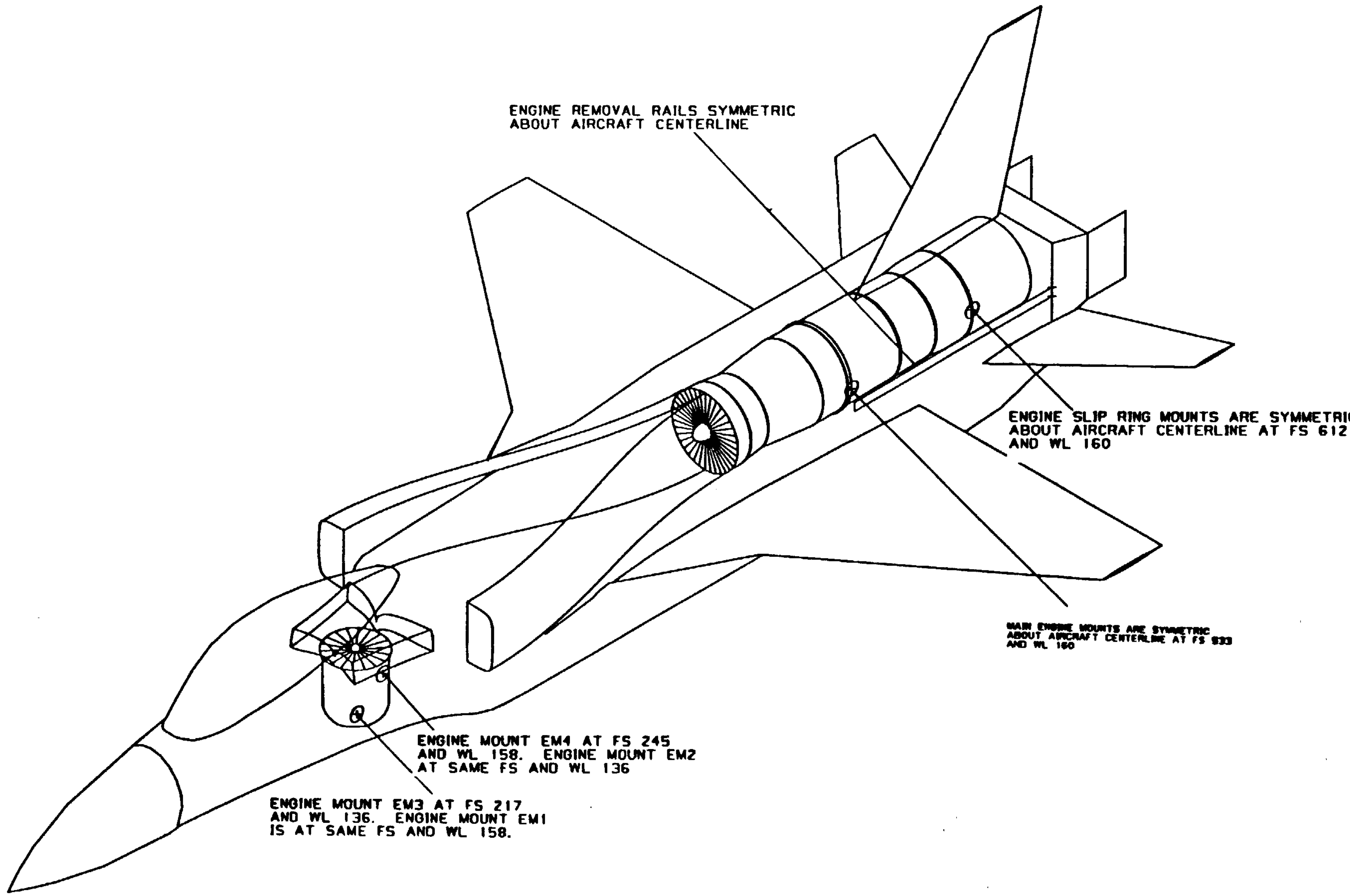

Figure 10.15 Monarch Engine Mountings

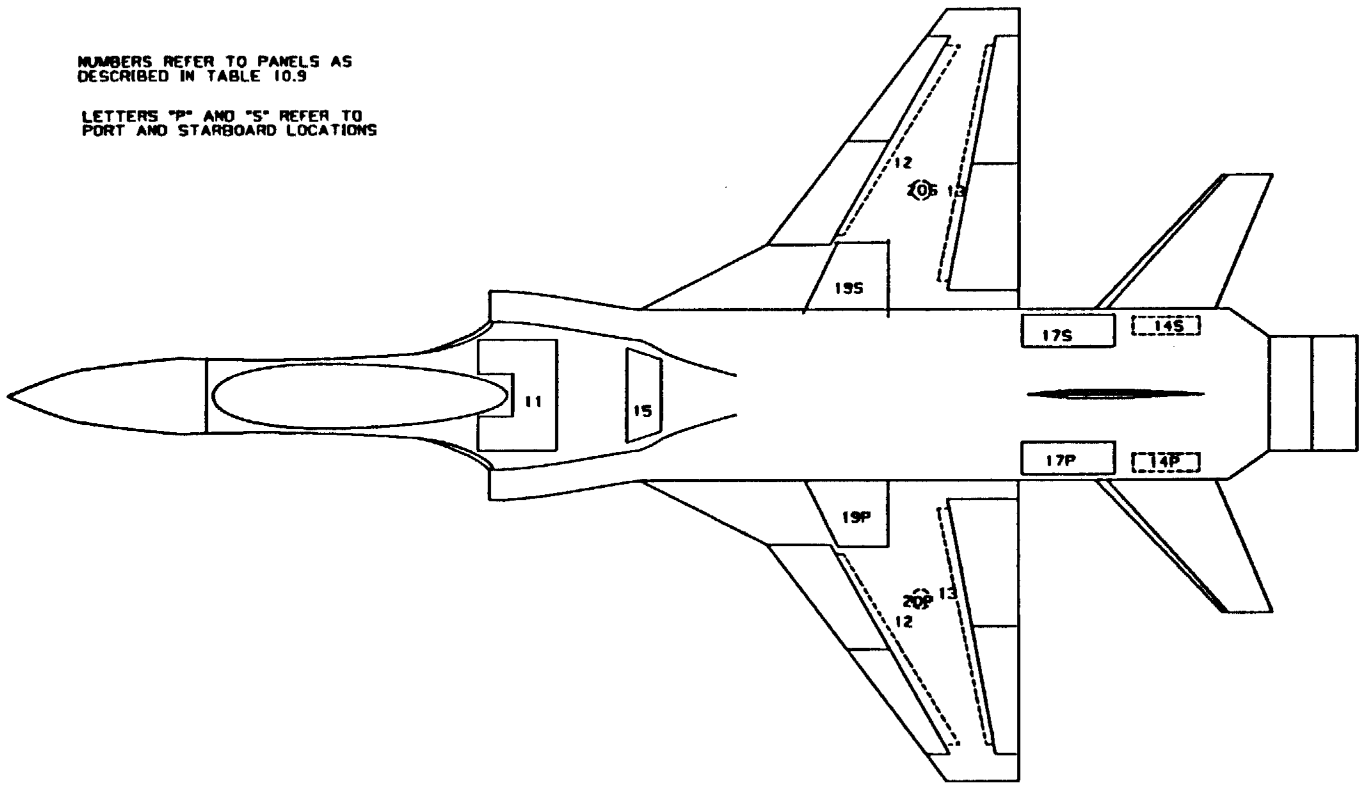

Figure 10.16 Monarch Access Panels

3. Use the engine removal jack to take the weight of the lift engine off engine mounts EM1-4.

4. Undo the inlet/compressor interface latches through access panel "3P" and "3S."

5. Remove bolts EB1-4 from engine mounts EM1-4 through access panels "1", "2", "3P", and "3S."

6. Lower engine and nozzle through access panel "2" using the engine jack.

This procedure is shown in 10.18. The engine is replaced by following the steps in reverse procedure. Note that access panels and bolts are sized so that they can be removed in arctic or other protective clothing. Access panels can be reached without the aid of the ladders (the highest reach required for either engine is 6.25 ft.).

Figure 10.15 shows the engine mountings for the cruise engine. Removal follows as:

1. Disconnect engine systems and pitch RCS ports through access panels through main landing gear bays (see Figure 10.17).

2. Disconnect inlet/compressor interface latches through landing gear bays.

3. Disconnect ventral nozzles through access panels "4P" and "4S."

4. Disconnect RCS yaw ports through access panels "5P" and "5S."

5. Remove access panel "6" around cruise engine nozzle and disconnect the nozzle.

6. Align the cruise engine removal jack rails with the engine rails of the aircraft.

7. Activate the grappling system of the removal jack so that it grips the removal posts at the nozzle/engine interface.

8. Disconnect the main mounts through landing gear bays.

9. Disconnect the engine slip ring through access panels "5S" and "5P."

10. Reverse the grappling system so that the engine is drawn out of the aircraft, along the engine rails, and onto the removal jack rail.

This process is diagrammed in Figure 10.18. Again note that the panels are sized so that they can be used in all types of protective clothing. The highest point a mechanic must be able to reach is 6.25 ft. from ground level.

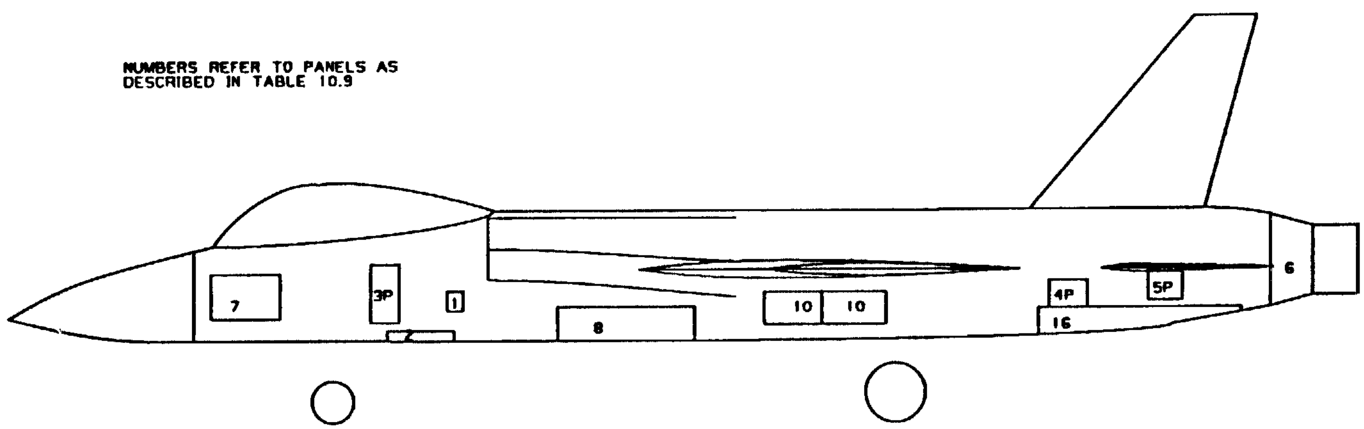

Figure 10.17 Monarch Access Panels

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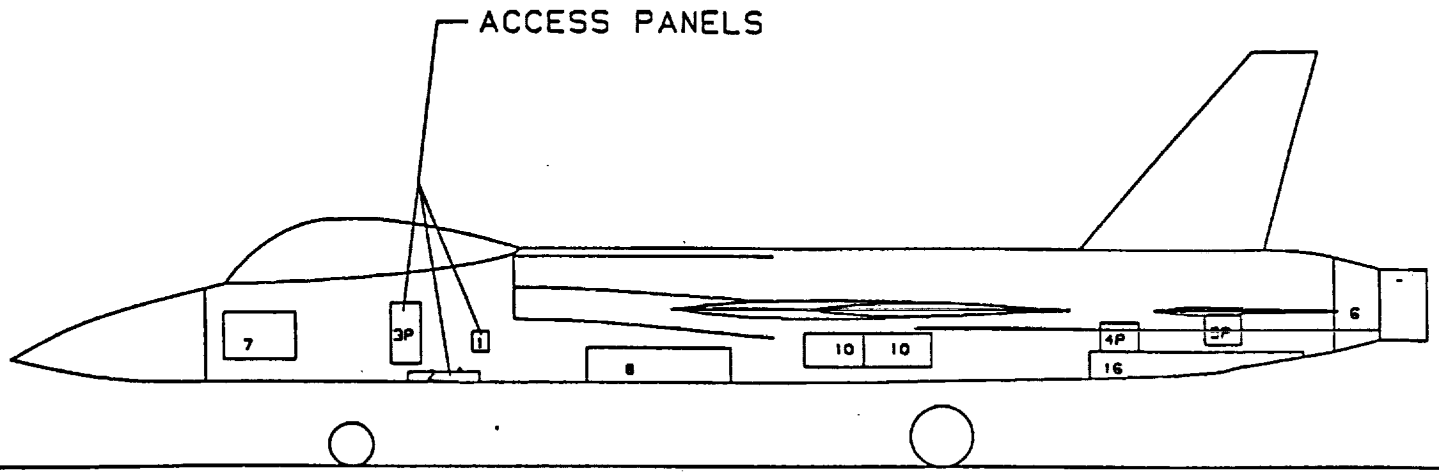
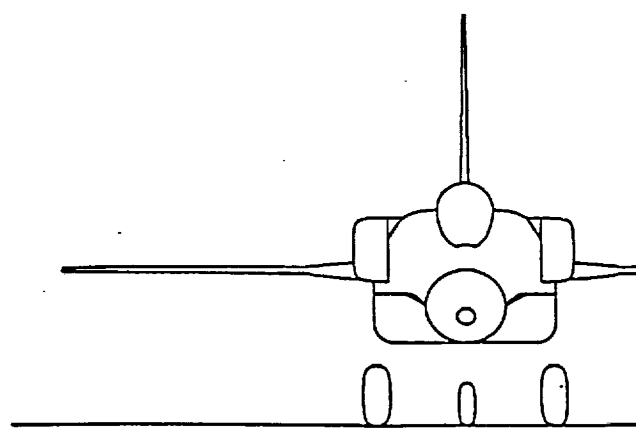


STEP 1 DISCONNECT SYSTEMS:
FUEL
CONTROL
NOZZLE FAIRING
INLETS



STEP 3 LOWER ENGI
REPLACE EN

Figure 10.18 Lift Engine

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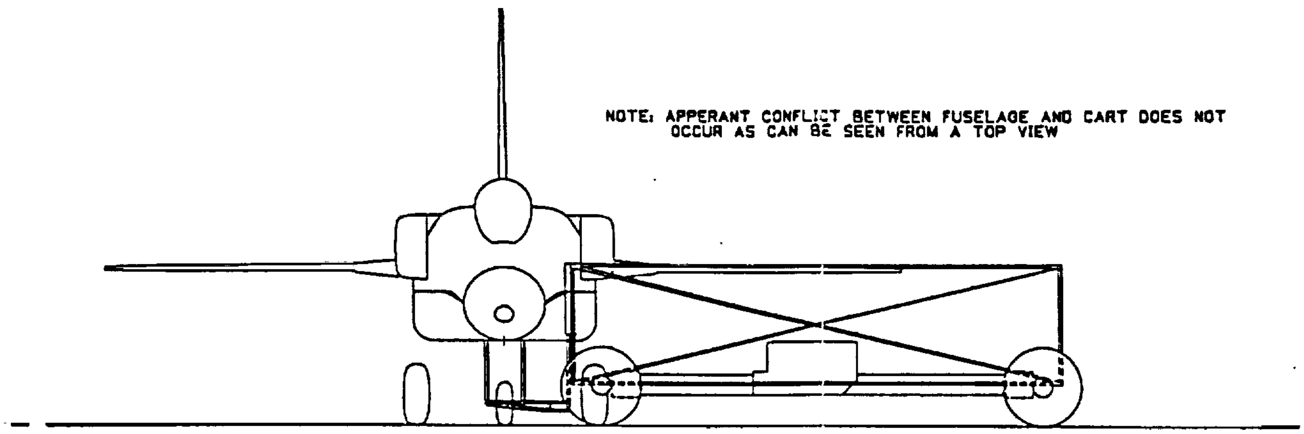
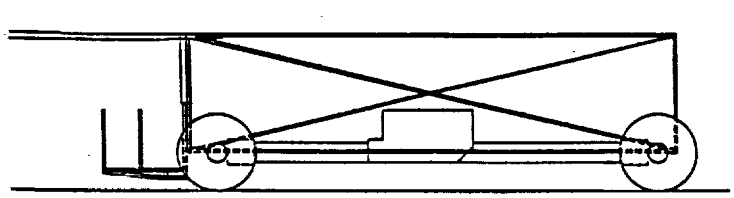


STEP 2 USE ENGINE JACK TO TAKE ENGINE WEIGHT
REMOVE ENGINE RETAINING BOLTS THROUGH
ACCES PORTS



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Removal

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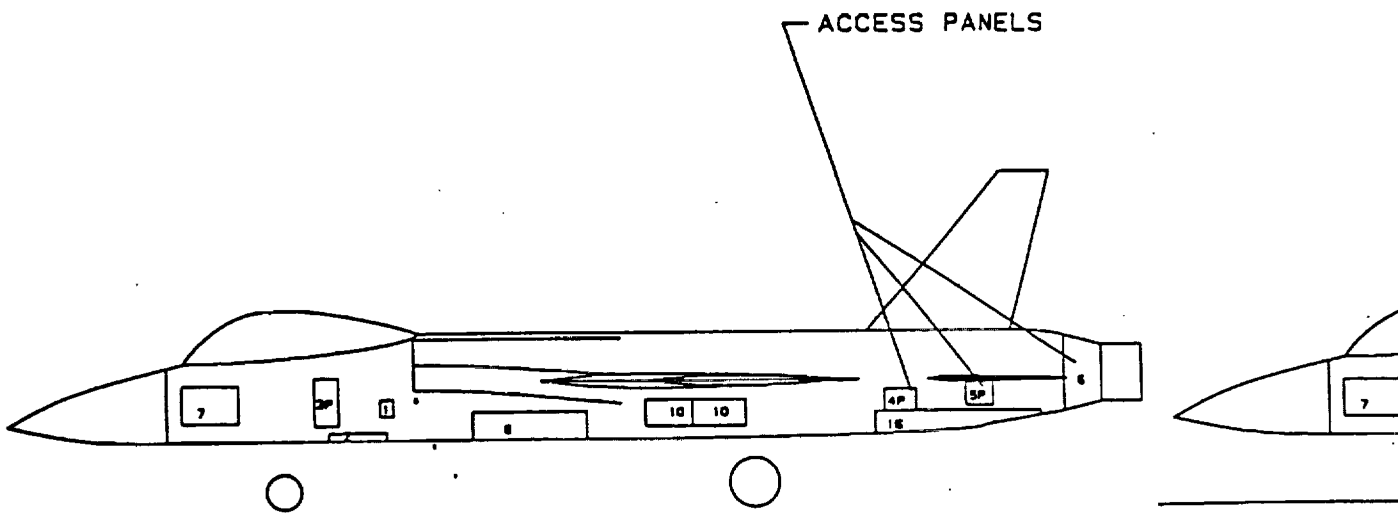
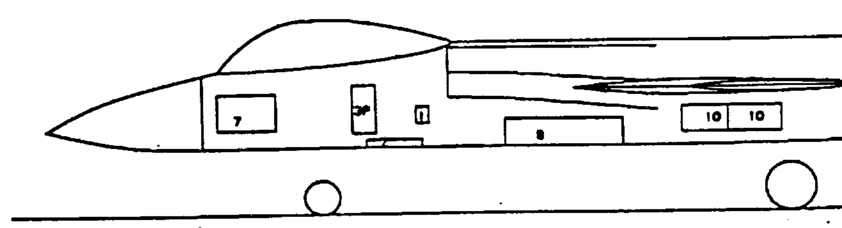


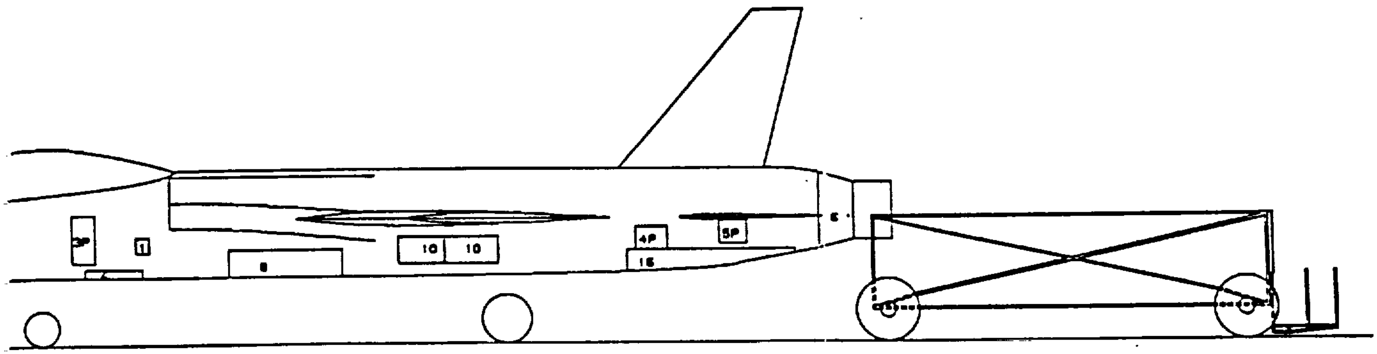
STEP 1 DISCONNECT SYSTEMS:
FUEL
CONTROL
NOZZLE FAIRING
INLETS



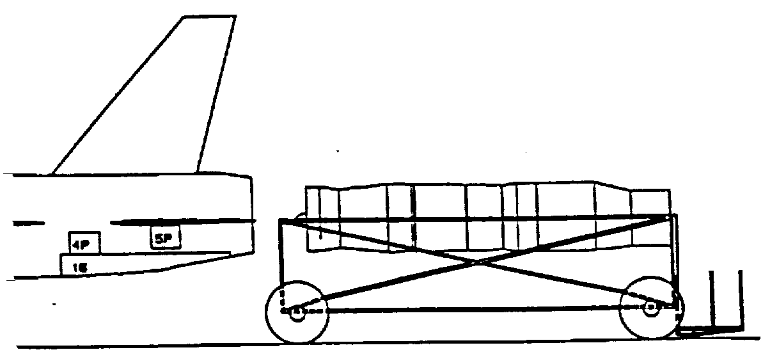
STEP 3 DRAW ENGINE AFT O
REVERSE AWAY FRO

Figure 10.19 C

FOLDOUT FRAME 2.



STEP 2 ALIGN ENGINE JACK WITH REMOVAL RAILS
ENGAGE GRAPPLING MECHANISM



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AIRCRAFT

ise Engine Removal

10.4.2 Systems Access

Primary systems access panels are shown in Figures 10.16 and 10.17. The numbers in these figures correspond to indicated panels in Table 10.9. Primary systems are accessed through the following panels as indicated in Table 10.9. Note that some systems access requires removal through the top of the aircraft. Based on systems placement, this was unavoidable, but it is not an ideal practice. Such access requires additional equipment such as ladders which increase cost, complexity, and maintenance time.

Table 10.9 Systems Access Panels

Panel #	Frame #	System	Comments
7	1	Radar	Access through side
8	15, 19	Gun	Port side, gun drops down
9	15, 19	Ammunition	Starboard side
10	21, 28	Avionics	Remove on trays
11	40, 41	ECM	Accessed from above
12	Wing spar 1	LE flight controls	Lower wing surface
13	Wing spar 2	TE flight controls	Lower wing surface
	21, 23	Flight control motor	Through landing gear bay
	26, 28	Flight control motor	Through landing gear bay
14P,S	35, 37	Stabilator Actuators	Remove downwards
15		Hydraulics	Remove upwards
16	32, 40	APU	Remove aft, down
17P,S	33, 35	Electric Drive	Remove upwards
		Fuel Pumps	Out through landing gear bays
		Landing Gear	Out through landing gear bays
		Oxygen system	Nose and weapons bays
18P,S		Fuel tank	Through weapons bays
19P,S		Inverted fuel tanks of wing	Remove through top
20P,S		Wing Tanks	Bottom wing ports

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<u>11. SYSTEMS LAYOUT</u>

The purpose of this chapter is to document the systems layout of the Monarch. The system selections are a result of the system research documented in Reference 11.2. The preliminary research was done to select the systems for the Monarch and to help in determining the layout of these systems. The following six aircraft were investigated:

* Fairchild Republic A-10,
* General Dynamics F-16C,
* McDonnell Douglas/BAe AV-8B,
* Dassault-Breguet Rafale,
* Saab JAS 39 Gripen,
* and the Eurofighter European Fighter Aircraft (EFA/Jf-90).

These aircraft were considered since they have modern systems with mission requirements similar to the Monarch. From this research, the systems of the Monarch were selected.

The survivability, maintainability, and reliability of a fighter aircraft is largely dependant upon the general arrangement of the systems. The following is a list of the causes of losses of single-engine aircraft in Vietnam and the Middle East; shown to exemplify the importance of designing survivability into the aircraft systems (Reference 11.3):

62% due to fuel system damage
18% due to pilot incapacitation
10% due to flight controls damage
7% due to engine power loss
3% due to structural damage

Much of the discussion of the system survivability is taken from Reference 11.4. The work in Reference 11.2 was completed to determine the existence of system conflicts. A thorough conflict analysis was completed and all system conflicts were deleted.

This chapter will discuss the general layout of the Monarch systems. These systems include:

11.1) landing gear,
11.2) fuel system,
11.3) flight control system,
11.4) electrical system,
11.5) environmental control system,
11.6) hydraulic system,
11.7) avionics selection,
11.8) and ECM selection.

11.1 LANDING GEAR LAYOUT

This section will discuss the layout of the Monarch landing gear. The material is organized as follows:

Methods used are from Reference 11.1 and calculations used for this section are presented in Appendix 4 .

11.1.1 Gear Loads and Tire Selection

Knowing the range of travel of the center of gravity, aircraft weight, and placement of the nose and main gear, the loads imposed on the landing gear are determined in Appendix 8. The determined loads of interest are:

* maximum static main gear load: 15,550 lb
* maximum static nose gear load: 5,400 lb
* minimum static nose gear load: 3,690 lb
* maximum dynamic nose gear load: 8,450 lb

Soft field tires were chosen for operation of the Monarch in austere rough field environments. The tire pressure on these tires is limited to below 140 psi. Since the tires chosen are designed for higher loads, the tires may be inflated somewhat below the design inflation pressure for better soft field operations. The specifications of the selected tires are selected for the Monarch are in Table 11.1:

Table 11.1 Monarch Landing Gear Tire Specifications

	Nose Gear	Main Gear
Outside Diameter	22 in	31 in
Width	8 in	13 in
Hub Diameter	10 in	12 in
Design Pressure	110 psi	135 psi
Maximum Load	8,500 lb	17,200 lb
Max Landing Speed	190 mph	210 mph
Loaded Tire Radius	9 in	12.4 in

11.1.2 Strut Sizing and Stroke Length Determination

The following stroke lengths are determined for the landing gear. It should be noted that liquid springs shock absorbers are used. A sink rate of 15 fps is used. Air Force requirements specify 10 fps. The higher sink rate was selected because it is envisioned that short no-flare landings at a steep approach angles will be employed during the service of the Monarch.

The following strut diameters and stroke lengths and are determined using landing gear load factors of 3.0 for the main gear and 11.3 for the nose gear:

Nose Gear:	3.25 in diameter	16 in stroke length
Main Gear:	4.23 in diameter	11 in stroke length

11.1.3 Landing Gear Layout

Nose Gear Description

The nose gear layout is illustrated in Figure 11.1. The length of the gear designed to produce 2.5 degrees of ground incidence. It retracts forward underneath the cockpit. It is designed with 3.0 inches of trail. The gear is retracted by actuating on the drag strut.

Main Gear Description

The main gear is designed with a triangulated structure much like that of the General Dynamics F-16 Fighting Falcon and is illustrated in Figure 11.2. This design has advantages in that:

* increase in energy absorption by the tire moving laterally across the runway,
* and it has a relatively large wheel stroke compared to the strut stroke.

The tire and hub section rotate 90 degrees about a line through the side strut upon retraction to lay the tire flat in the wheel well. The oleo shock strut will act as the radius link when it is in the extended position. When the gear is down and locked, the hub is locked to keep from rotating upon landing. The gear is actuated from the drag strut which is attached to the side brace.

11.1.4 Aircraft Tip-over Criteria

The tip-over angles measured for the Monarch from Figure 11.3 are:

Lateral Clearance Angle:	28 deg
Longitudinal Clearance Angle:	14 deg
Lateral Tip-over Angle:	65 deg
Longitudinal Tip-over Angle:	20 deg

A takeoff analysis has been performed and verifies that the 20 degree longitudinal tip-over angle is acceptable.

11.2 FUEL SYSTEM

As stated previously, the fuel system is the primary contributor to the vulnerability and survivability of an aircraft. The 'kill modes', or the types of failure, of a fuel system are (Reference 11.4):

* fuel supply depletion,
* in-tank fire and explosion,
* void space fire and explosion,
* sustained exterior fire,
* and hydraulic ram.

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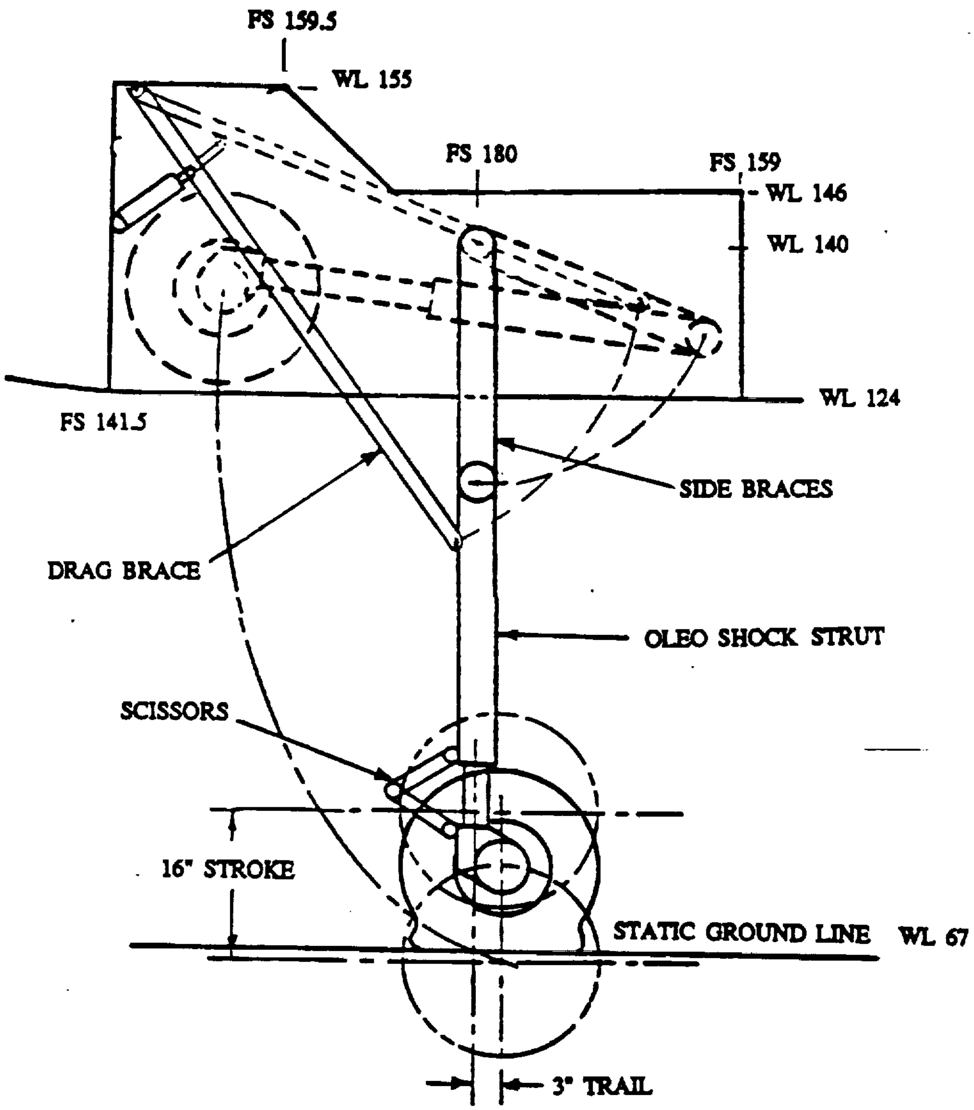

Figure 11.1 Monarch Nose Landing Gear Layout

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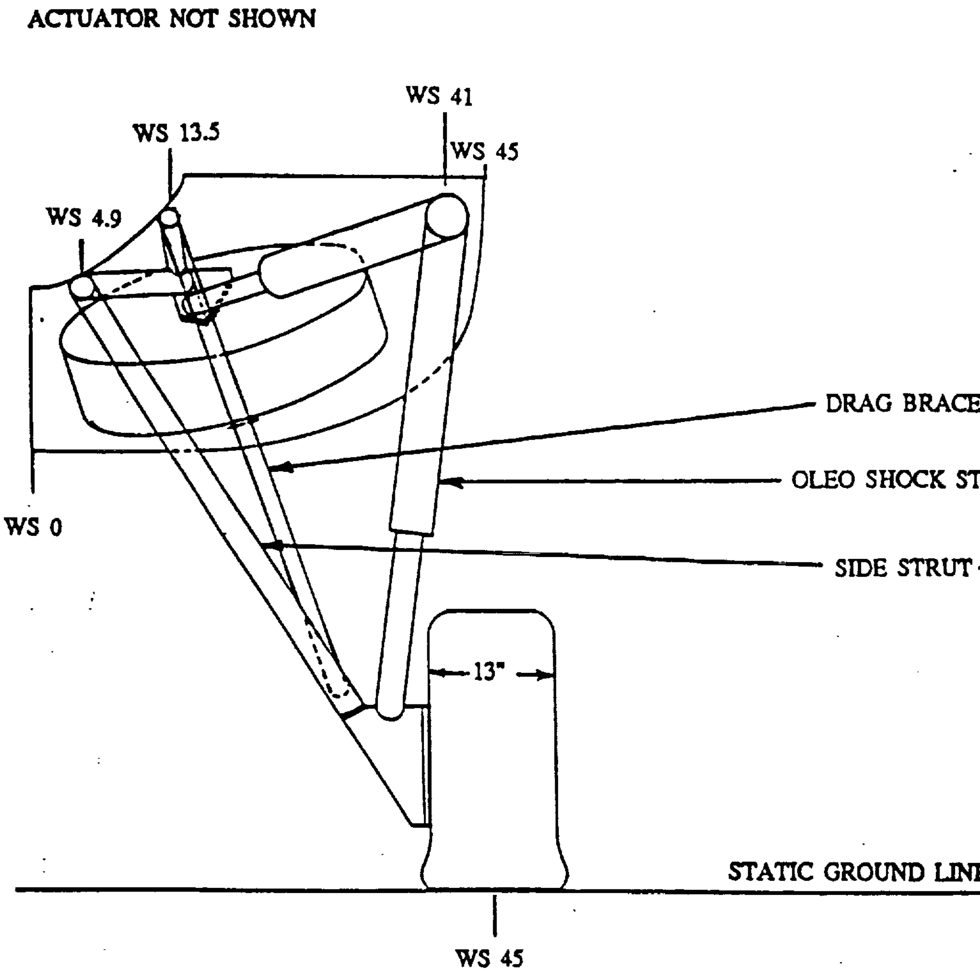
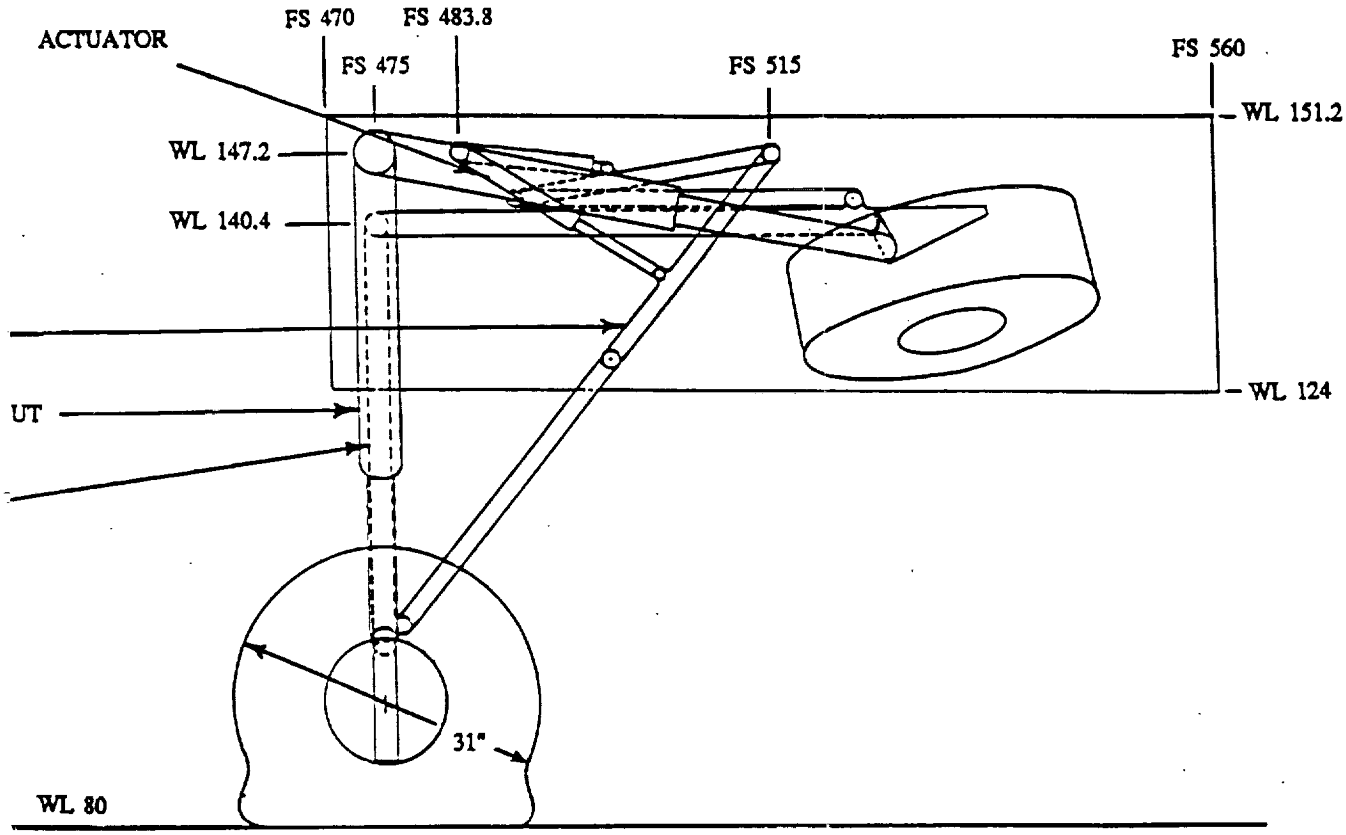


Figure 11.2 Monarch Mai

FOLDOUT FRAME 2



Landing Gear Layout

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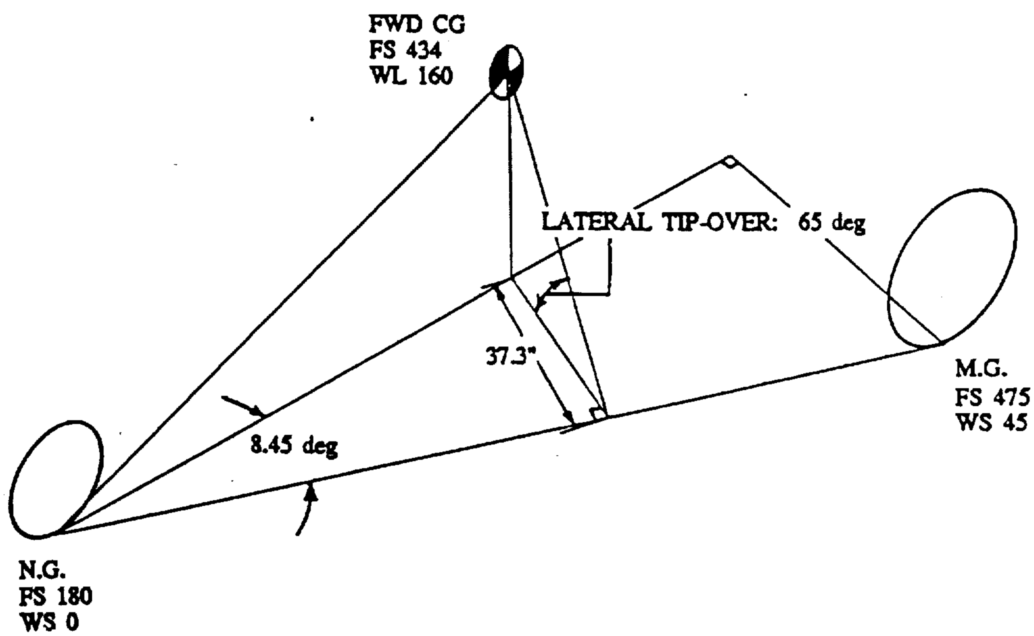

Figure 11.3 Monarch Tip-o

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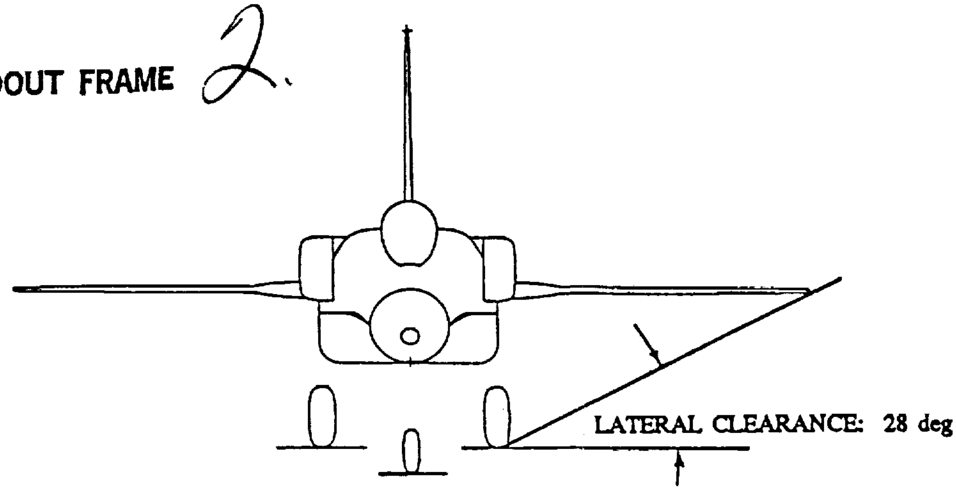

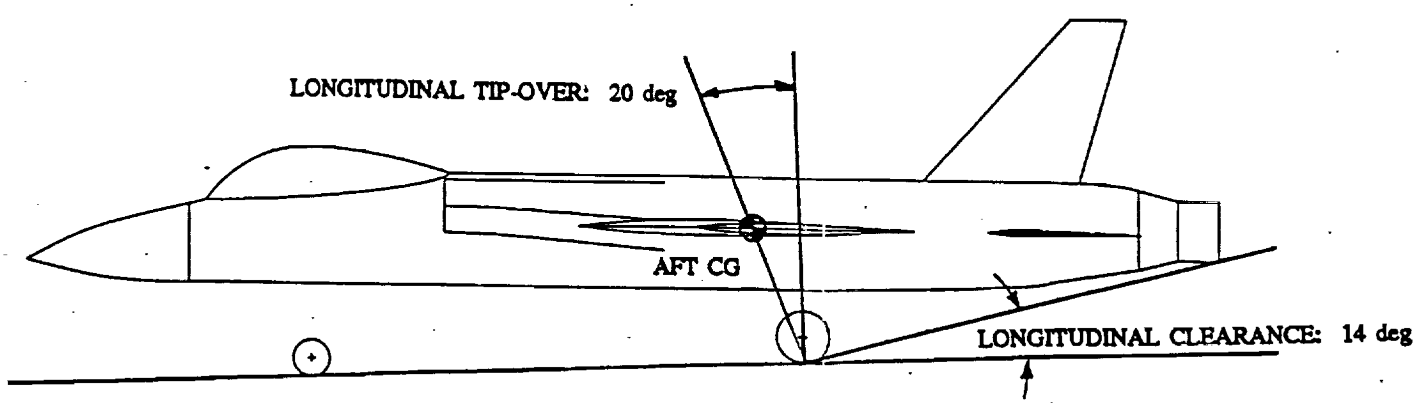

er Geometry

These factors were kept in mind when laying out the fuel system of the Monarch.

The fuel system layout is shown in Figure 11.4. The tank fuel volumes are as follows:

each wing	27.5 cu.ft. / 1348 lb
forward fuselage tank	50.0 cu.ft. / 2450 lb
aft fuselage tank	75.0 cu.ft. / 3675 lb
TOTAL	182.0 cu.ft. / 8918 lb

This accommodates the required fuel capacity of 8642 lb determined in Chapter 5.

These fuel tanks are self-sealing and tear resistant. They are also equipped with a reticulated foam filling (porous foam panels folded to fill the tanks) to prevent large ullage overpressures following ignition of the flammable vapor (Reference 11.4). As well as suppressing tank fires, the foam helps to relieve fuel surging and sloshing, and may reduce the effects of hydraulic ram. Hydraulic ram is the intense pressure waves generated in the contained liquid by penetrators or fragments passing through the liquid.

The system is set up such that there are essentially two separate systems. The forward fuselage tank is connected directly to the left wing tank, and the aft fuselage tank is connected directly to the right wing tank. The two 'separate' systems are also connected to allow for fuel management. This system allows for the complete shut-off of a damaged tank. The two systems have their own fuel pumps. These are located in the aft fuselage next to the engine. This allows for the use of a fuel suction system rather than a boost system. This is desirable since a boost system would tend to continue sending fuel through a damaged line, whereas a suction system would not be able to do this. The lift engine has its own fuel pump and line.

The Monarch is designed for single point refuelling on the underside of the left wing. It is also capable of in-flight refuelling, F-16 style, through the port in the upper fuselage behind the cockpit. Inverted flight tanks are located within the wing fuel tanks to ensure the availability of fuel in inverted conditions. The Monarch is also equipped with a fuel management system to control center of gravity travel, a fuel jettison system through the outboard section of each wing, and a fuel indicating and ventilation system.

The engine is started with the use of a jet fuel starter. This is located in the aft fuselage beneath the engine and also acts as an APU. The jet fuel starter, which is essentially a small jet engine, is started by a mechanical control from the cockpit. This releases pressurized hydraulic fluid which flows into the jet fuel starter gearbox, starting the small engine. This in turn drives the generators providing electrical power to start the main engines. The jet fuel starter is a self sustaining system; the hydraulic accumulators are self charging after engine start. This system requires no battery and incorporates a hand pump for hydraulic backup.

11.3 FLIGHT CONTROL SYSTEM

Because maintaining aircraft stability and control is one of the most critical factors affecting safety of flight, as well as the combat survival of the aircraft and crew, much

FOLDOUT FRAME /.

1. Fuel Pump
2. Forward Fuselage Fuel Tank
3. In-Flight Refuelling Port
4. Aft Fuselage Fuel Tank
5. Wing Fuel Tank
6. Inverted Flight Tank
7. Fuel Jettison
8. Jet Fuel Starter
9. Underwing Refuelling Port

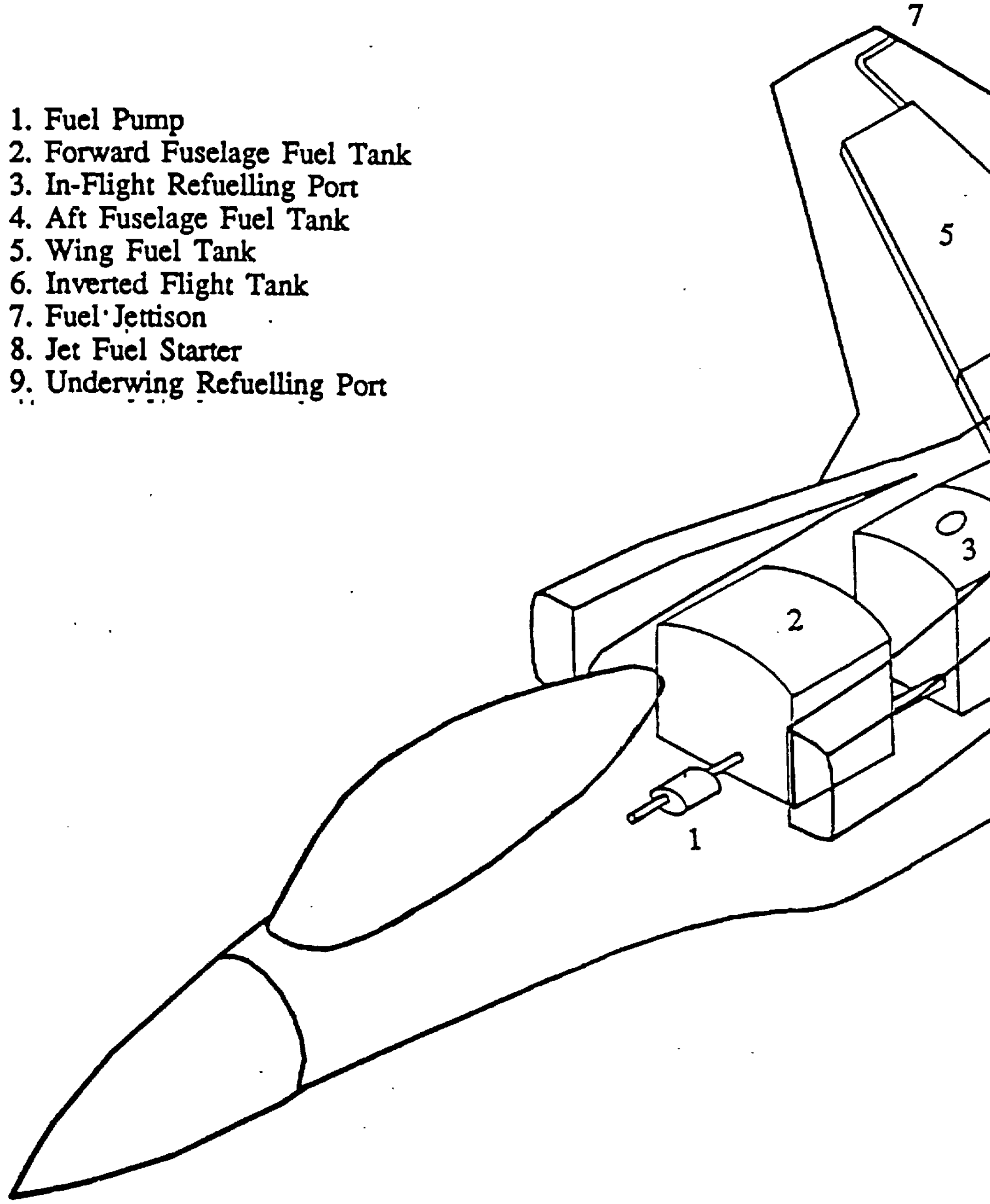

Figure 11.4 Fuel System L

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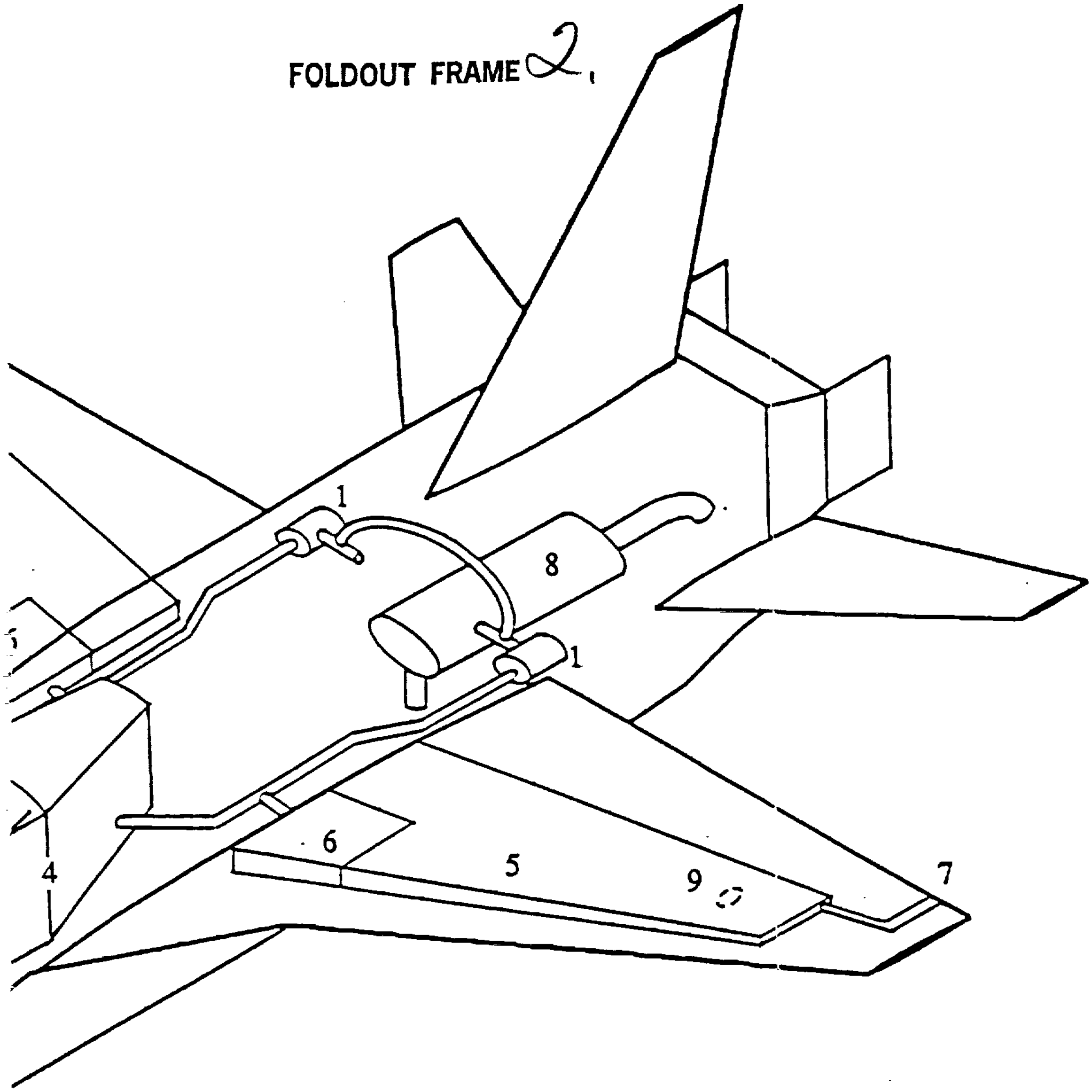

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attention should be given to the design of the control system to ensure that there is no unacceptable degradation of functional capabilities due to one or more component failures.

The flight control system of the Monarch is a quadruple-redundant fly-by-wire system. The flight control system layout is shown in Figure 11.5. The system uses a combination of rotary and electrohydrostatic actuators. The leading edge devices, the outboard ailerons, and the inboard flaperons use rotary actuators; the differential stabilizer uses linear electrohydrostatic actuators. The rotary actuators were determined to be 8 inches in length and the linear actuators are required to be 2.4 inches in diameter. A unique feature of the Monarch's flight control system is that it has no rudder. Directional control is obtained by thrust vectoring. This is explained in more detail in Section 9.9. For simplicity and maintainability reasons, one type of actuator should be used for all of the flight control surfaces. However, due to the proven performance of using linear actuators in the differential stabilizers of other fighters, this was not done. The advantage of using the electrohydrostatic actuators is that they are self contained. They need not be hooked up to the hydraulic system. This increases the survivability of the aircraft since there will not need to be hydraulic lines to all of the actuators. The flight control system is driven by the generators shown in Figure 11.6.

STOVL requirements create additional complexity in the flight control system. The aerodynamic controls must be linked to the reaction control system to be used in transition and hover. The reaction control system is discussed in greater detail in Subsection 7.3.1.

Although it was not incorporated in the design, it was determined that it would be desirable to utilize the separate surface control system concept (Reference 11.5) for the Monarch flight control system. The following description of this concept was taken from Reference 11.5. The conventional flight control surfaces are separated into segments. Some are driven directly by the pilot while the others are used for stability augmentation, autopilot control and attitude command applications. The servo-driven separate surface control can be used for stability augmentation functions as well as for autopilot functions. The pilot may elect at any time to fly the airplane through the wheel while retaining full benefit of stability augmentation. There is no feedback from the separate surface to the pilot.

11.4 ELECTRICAL SYSTEM

The electrical system of the Monarch is shown in Figure 11.6. The electrical system is dual redundant, powered by two 30 kVA engine driven generators. An electrical load analysis is shown in Figure 11.7. The phases listed refer to the mission phases found in Figure 9.1. The aircraft will still have the use of critical electrically powered components in the case that only one of the generators is operative. A 20 kVA battery is available for backup power in the event that both generators fail. The battery will supply adequate power for critical equipment, such as the flight control system. Since the landing gear is designed for gravity drop extension, backup power is not required to power the hydraulic system.

The auxiliary power unit (APU) is a jet fuel starter, and performs the dual role of engine starting and backup power. The jet fuel starter is discussed in more detail in Section 11.2. No battery is required for engine startup with the jet fuel starter system.

FOLDOUT FRAME /.

1. Control Stick
2. Drive Motor
3. Rotary Actuator
4. Linear Electrohydrostatic Actuator

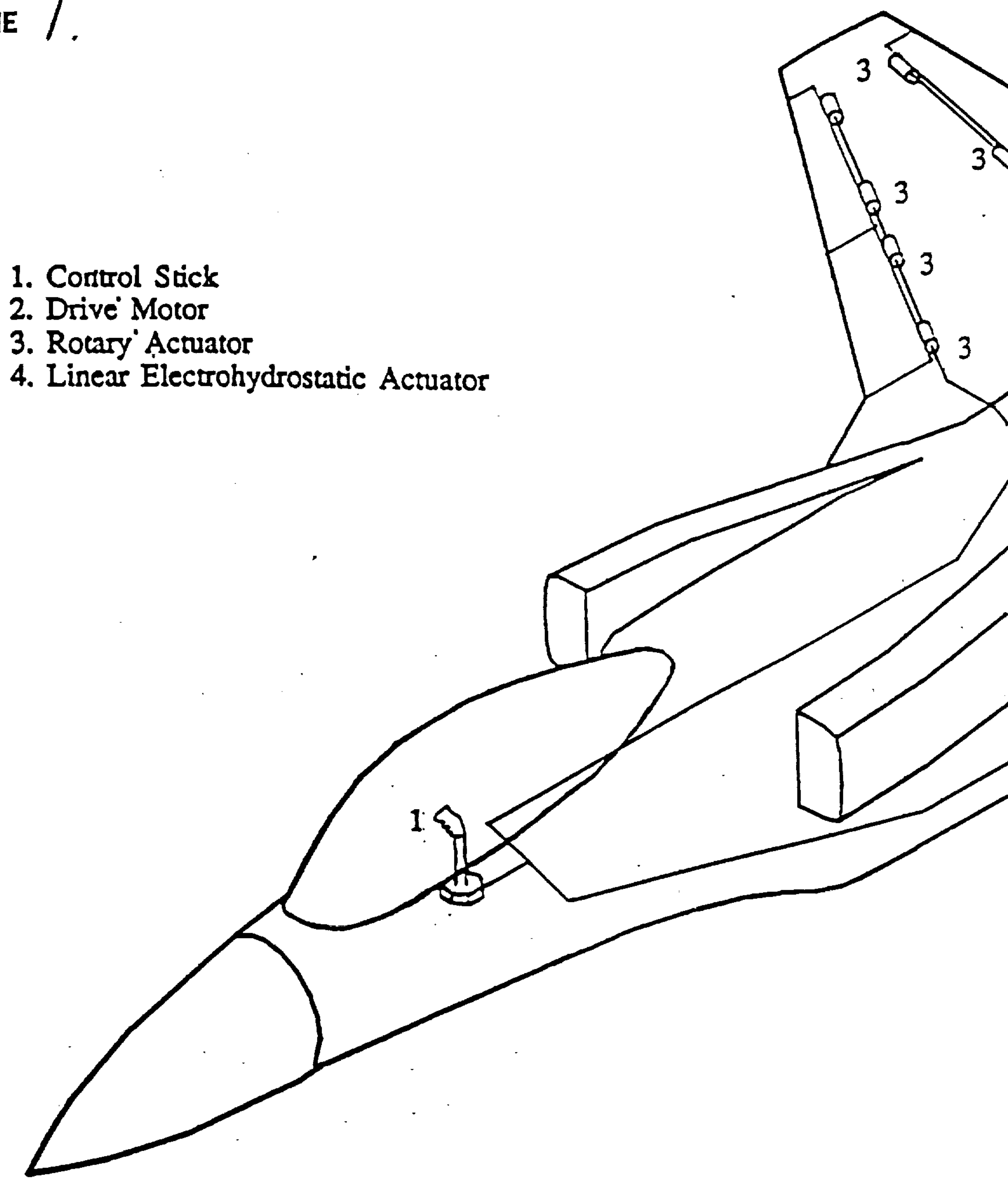

Figure 11.5 Flight Control System Layo

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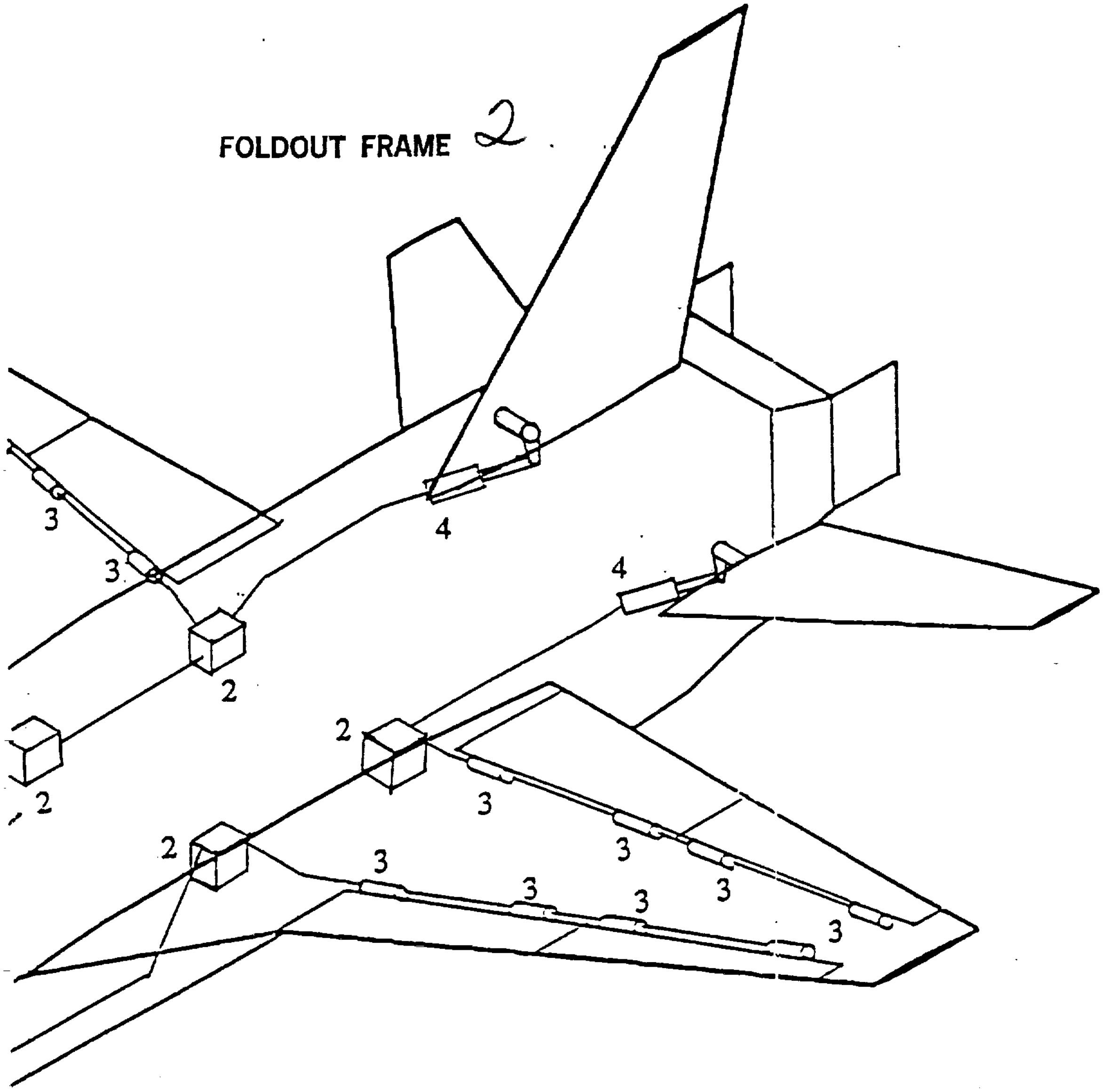

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FOLDOUT FRAME /

1. Radar, Equipment, and Forward Avionics
2. Cockpit Controls
3. Lift Engine Actuators
4. Auxiliary Inlet Actuators
5. Internal Weapons Bay Doors and Launch Mechanism
6. Aft Avionics Bay
7. Weapon Launch Mechanism
8. Ventral Nozzle Actuator
9. Engine Driven Generator
10. Chaff and Flare Dispenser
11. ECM and IFF Pod
12. ~~Pitch Vane Actuator~~
13. Yaw Vane Actuator

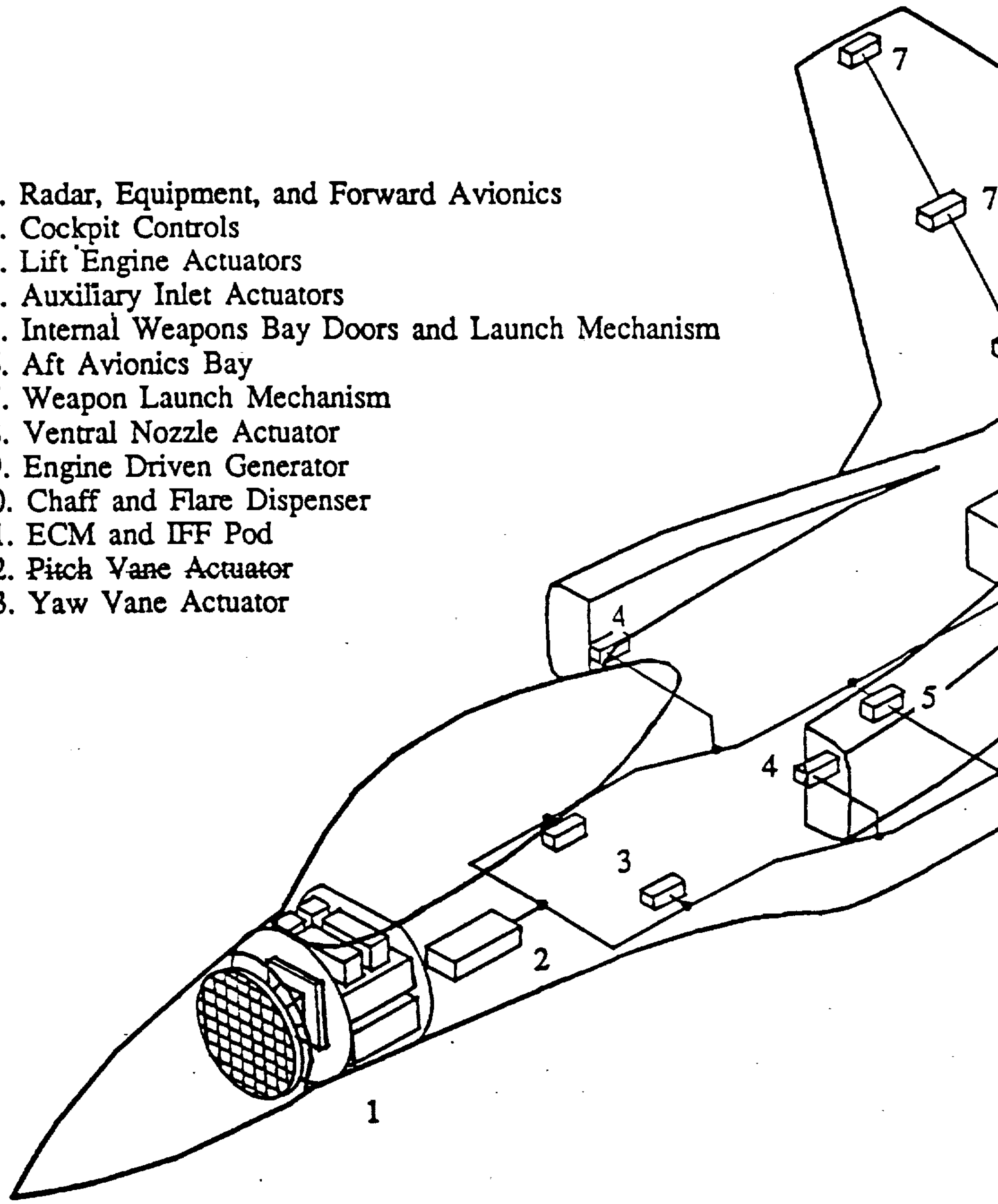

Figure 11.6 Electrical System Lay

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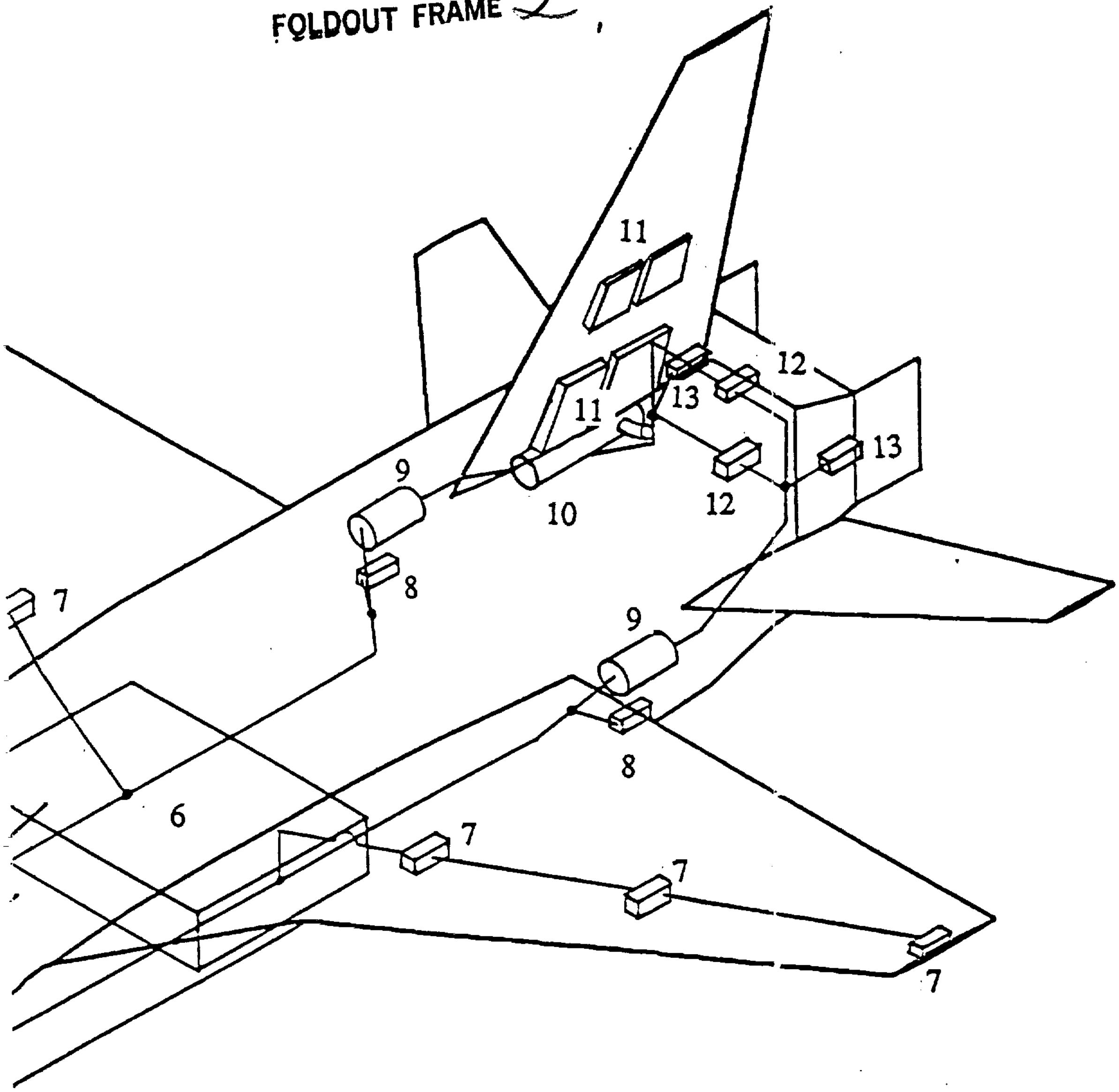

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All of the actuation mechanisms in Figure 11.6, represented by boxes, are either electro-hydrostatic or electromechanical actuators and are powered by the two generators, as are the flight control system actuators shown in Figure 11.5.

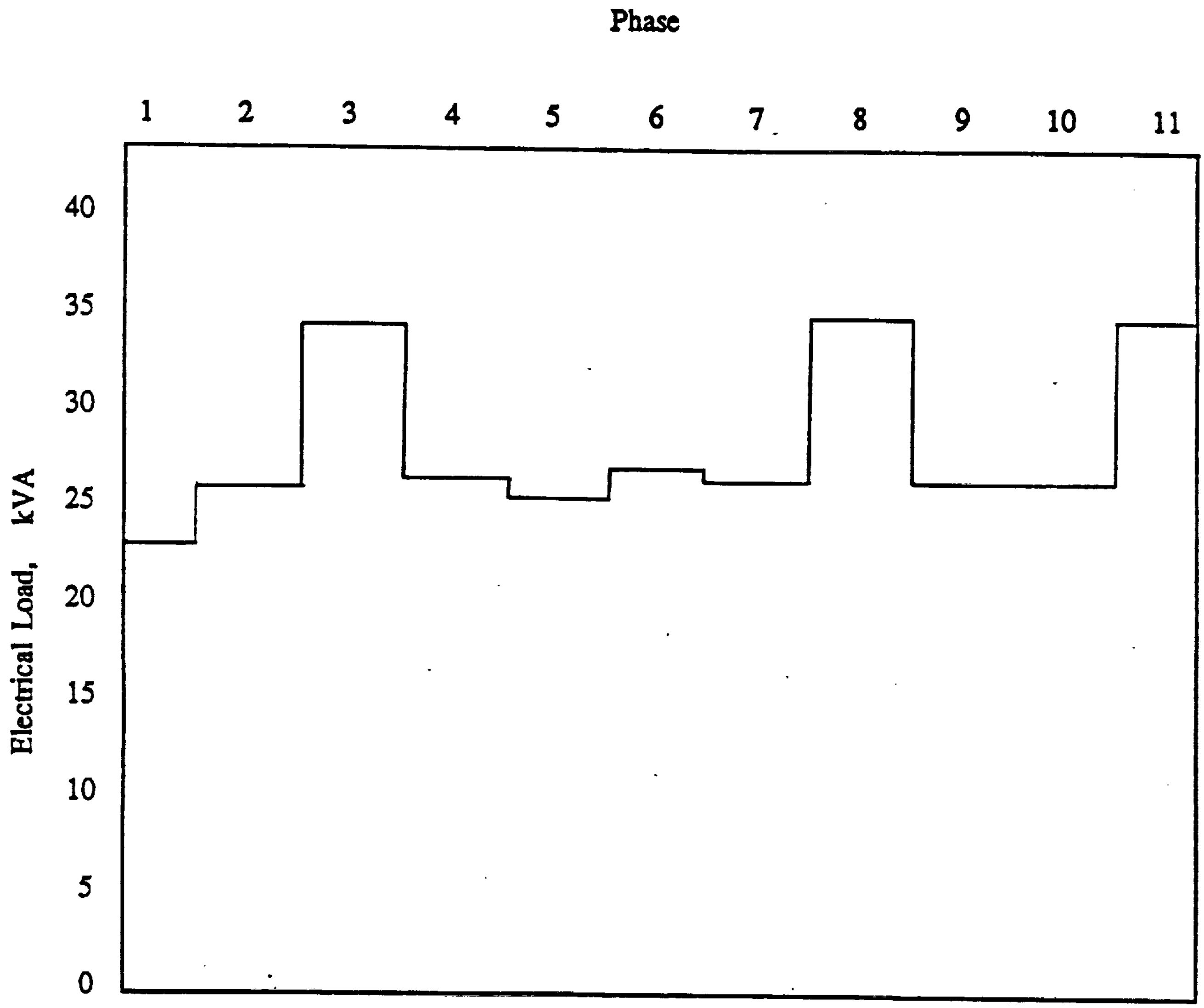

Figure 11.7 Electrical Load Profile Diagram for the Counter Air Mission

11.5 ENVIRONMENTAL CONTROL SYSTEM

The environmental control system of the Monarch, shown in Figure 11.8, uses engine bleed air. This air is piped forward from the engine compressor to the air conditioning unit and heat exchanger. The air is then used to cool the cockpit as well as the avionics compartments. This system is also used to provide for cockpit pressurization.

FOLDOUT FRAME /.

1. To Cockpit and Forward Avionics and Radar
2. On Board Oxygen Generating System
3. Heat Exchanger and Air-Conditioning Unit
4. To Aft Avionics Bay
5. Engine Compressor Bleed

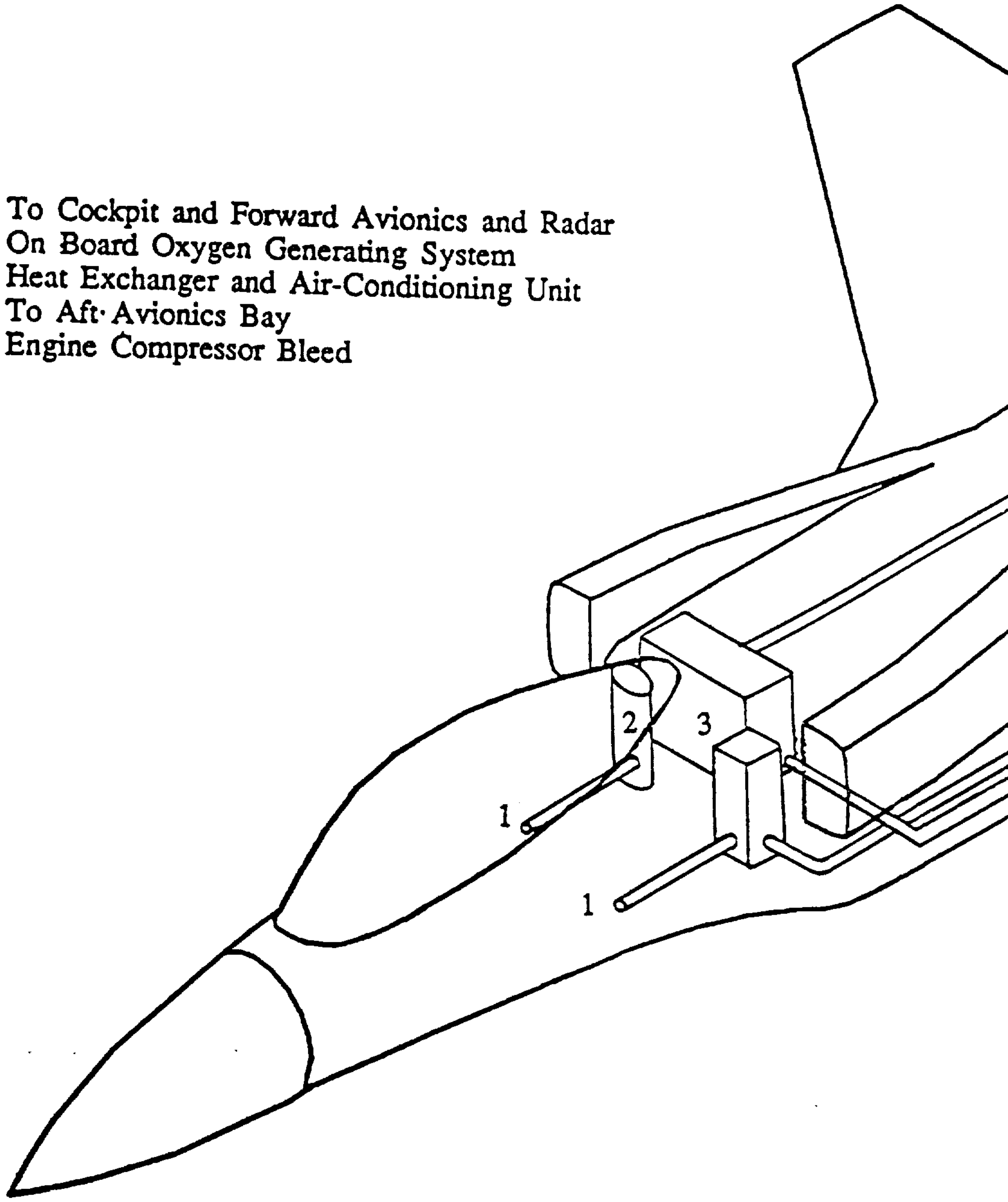

Figure 11.8 Environmental Control

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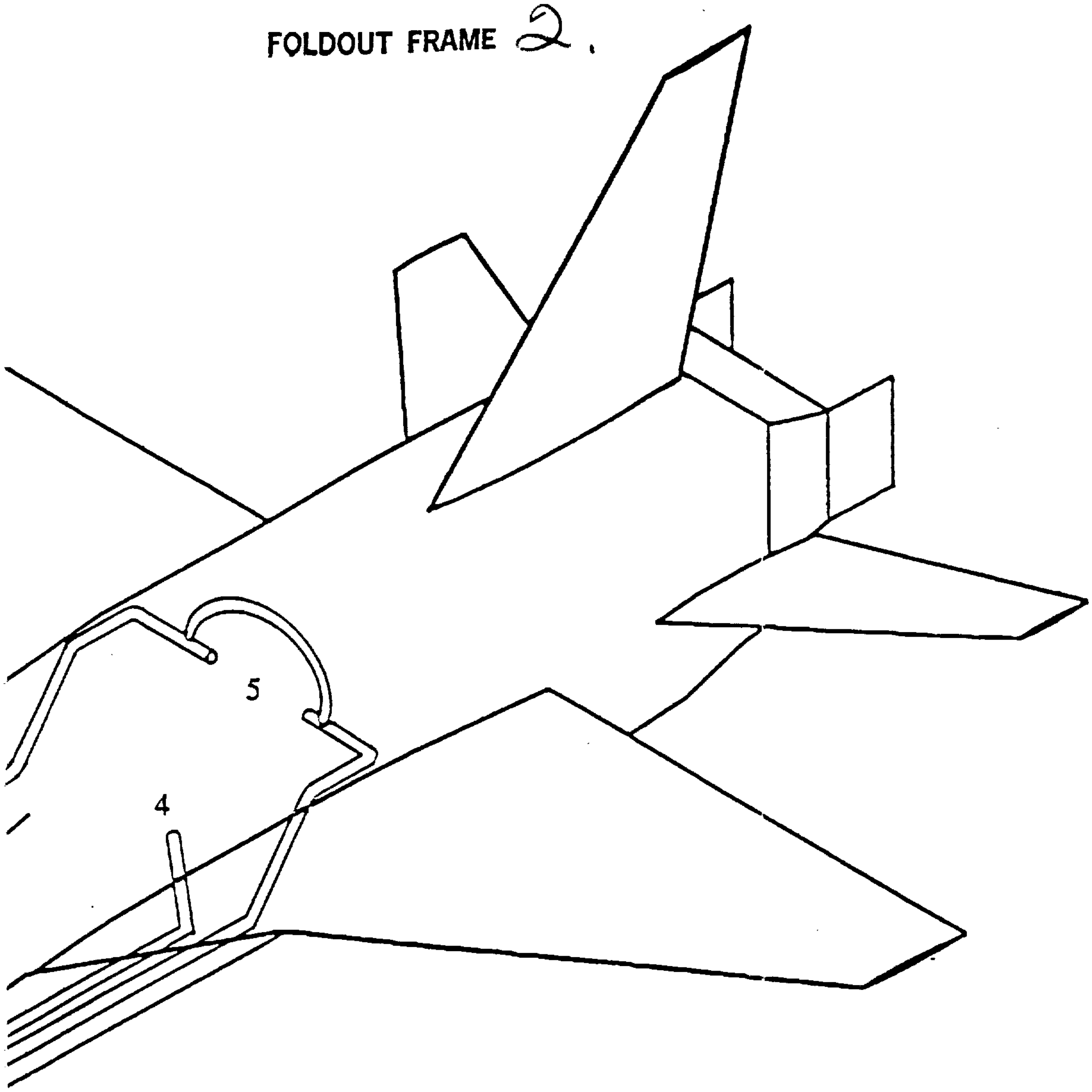


ystem Layout of the Monarch Aircraft

An on board oxygen generating system will be used to provide the pilot with the necessary amount of oxygen. The bulk of these two systems is located between the forward fuselage fuel tank and the dedicated lift engine, and above the internal weapons bay.

11.6 HYDRAULIC SYSTEM

The Monarch has a dual hydraulic system, both fed from the same reservoir. The hydraulic system layout is shown in Figure 11.9. The reservoir and pumps are located between the fuselage fuel tanks and above the ammunition drum. The pumps are electrically powered by the engine driven generators. Hydraulic power is used for:

nose gear retraction,
main gear retraction,
weapons bay door actuation,
and gun firing.

Each of the systems has a pressure of 4,000 psi and flow rates of 40 - 50 US gallons per minute. This higher pressure will allow for smaller actuator sizes.

Hydraulic accumulators are located at the landing gear mechanisms to allow for emergency use with the hydraulic pumps disabled.

11.7 AVIONICS SELECTION

An examination of current fighter aircraft was made to determine what type of equipment is required in the modern combat environment. The effectiveness of a combat aircraft is closely related to the effectiveness of its radar. Since the Monarch is required to complete both air superiority and battlefield air interdiction missions, a multi-mission radar was deemed necessary. It was also desired that the radar have day/night and all weather capability.

A system of multi-function displays and a HUD will provide the pilot with pertinent information. These will be designed to lessen the pilot's workload and allow him to concentrate on the task at hand. An IFF transponder will be used for identification purposes in combat. An air data computer and flight control computer will be used by the flight control system. A weapons control system is required to provide the pilot with efficient methods of deploying weapons. Communication is achieved with the use of UHF/VHF transceivers and navigation is provided by an inertial navigation system as well as TACAN and ILS systems. The Avionics bays of the Monarch aircraft are shown in Figure 11.6.

11.8 ECM SELECTION

Susceptibility is Defined in Reference 11.4 as the inability of an aircraft to avoid being damaged in the pursuit of its mission, and its probability of being hit. The best way to decrease an aircraft's susceptibility is to make it invisible, or stealthy, to the enemy. However, when this is not feasible, electronic counter measure devices can be used to warn of an impending attack and to provide a means to counter that threat. Much of the following discussion of the ECM devices selected for the Monarch were taken from

FOLDOUT FRAME /.

1. Nose Landing Gear Retraction Mechanism
2. Hydraulic Pump
3. Hydraulic Reservoir
4. Main Landing Gear Retraction Mechanism

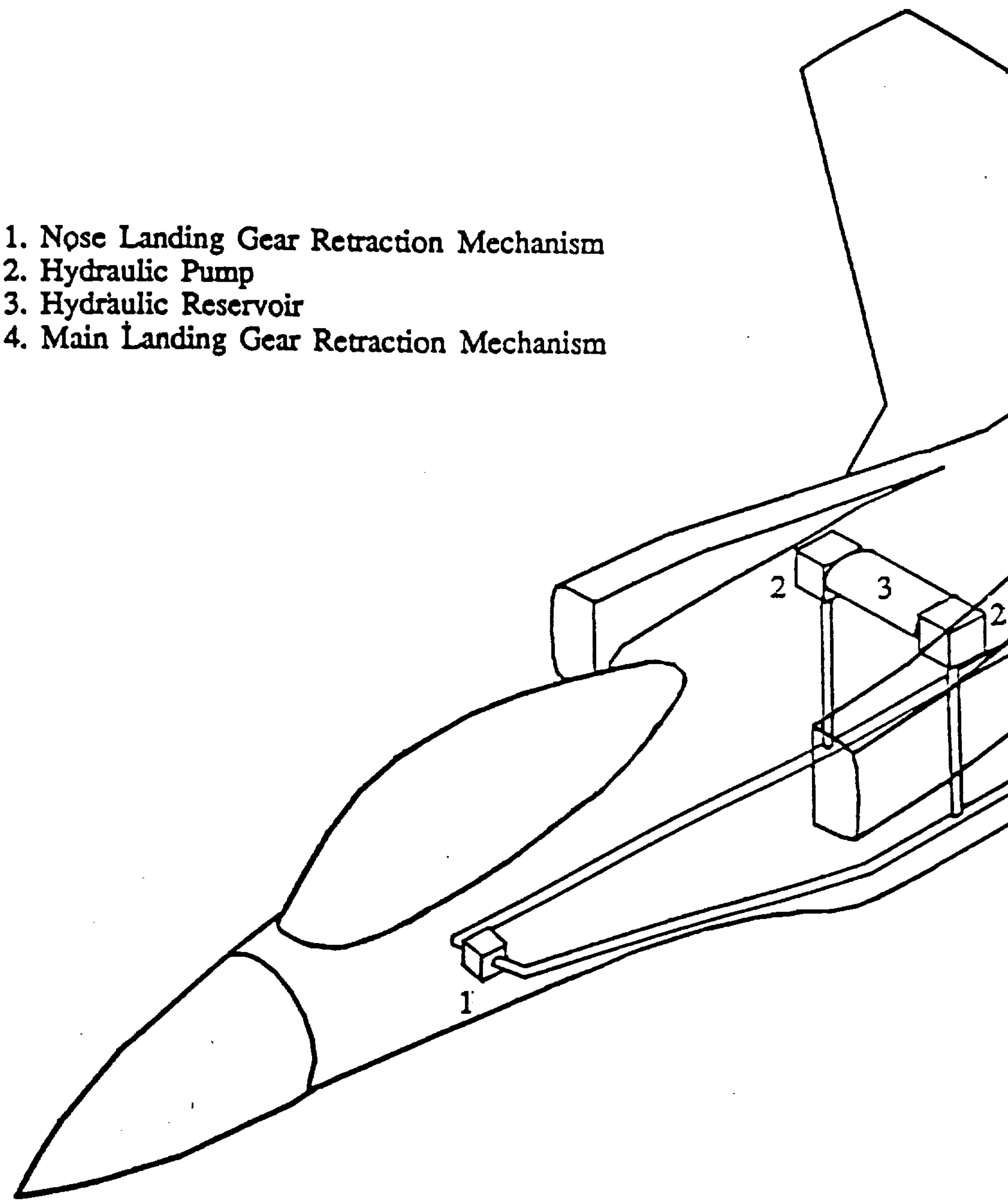
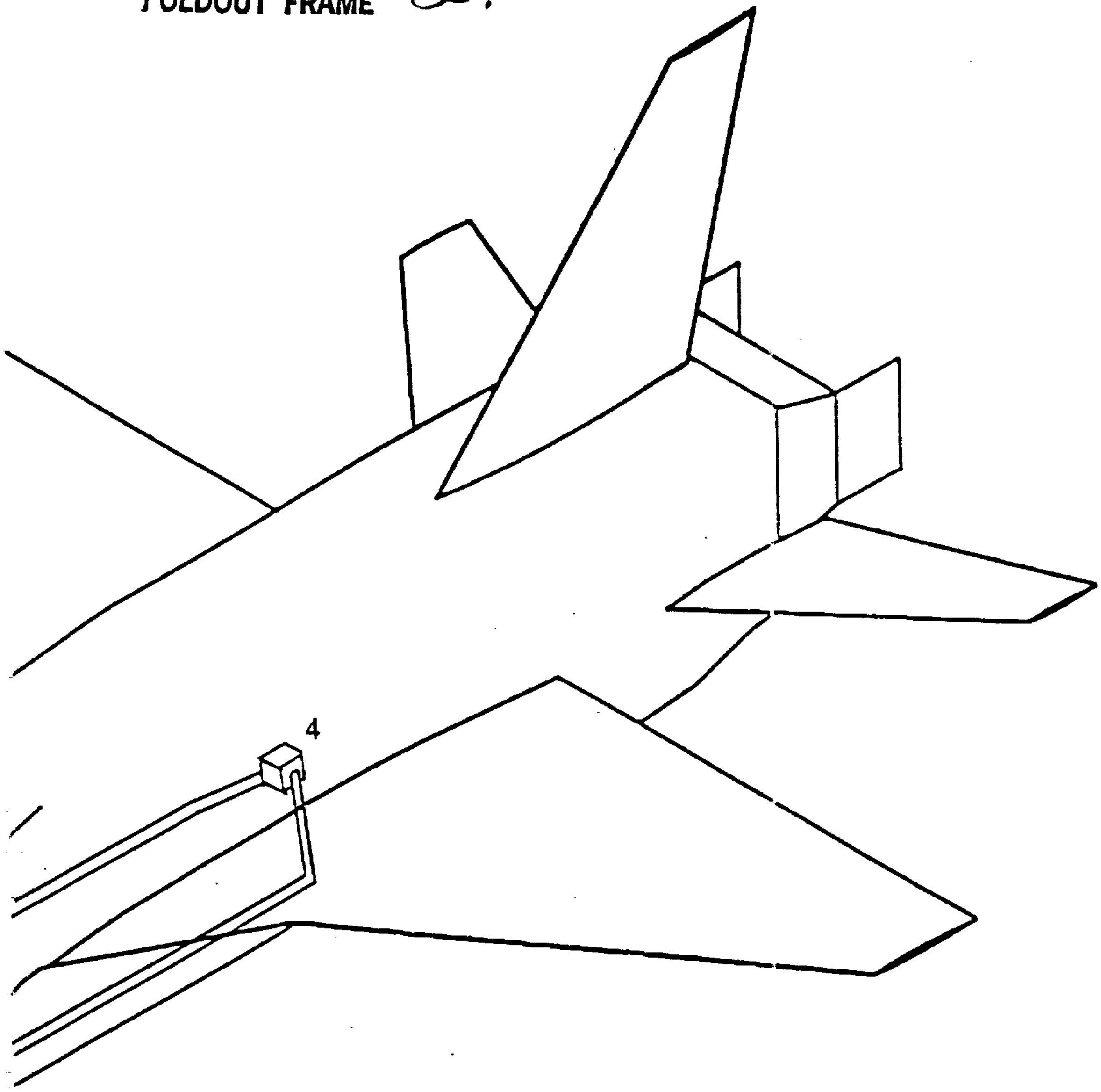


Figure 11.9 Hydraulic Sy

FOLDOUT FRAME 2.



tem Layout of the Monarch Aircraft

Reference 11.4.

The Monarch aircraft is equipped with a radar warning receiver (RWR). This allows for the detection of radiating threat elements and the accurate location and status of the weapon delivery systems intent on destroying the aircraft. The selection of an RWR is heavily influenced by the aircraft mission requirements. Each mission places certain requirements upon the RWR system. An RWR that is capable of meeting all of the Monarch's mission requirements, air-superiority as well as battlefield air interdiction, should be selected

The Monarch will use a jamming device that generates and directs transmission of a noise-like signal that has the characteristics of radar receiver noise. Jammers are often used to mask or obscure the target echo. Another ECM feature that will be used on the Monarch are expendables. These are materials or devices designed to be ejected from an aircraft for the purpose of denying or deceiving threat tracking systems for a limited period of time. The Monarch will use chaff and flare dispensers at the base of the vertical tail for this purpose.

Since the Monarch aircraft has no rudder, there should be sufficient room to store this ECM equipment in the vertical tail. The ECM pods are shown in Figure 11.6. This equipment will decrease the Monarch's susceptibility to attack and the resulting damage of that attack.

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11.3 Bonds, Ray, The Great Book of Modern Warplanes, Salamander Books, Ltd., 1987.

11.4 Ball, Robert E., The Fundamentals of Aircraft Combat Survivability Analysis and Design, American Institute of Aeronautics and Astronautics, Inc., 1985.

11.5 Roskam, Jan, Airplane Flight Dynamics and Automatic Controls: Part II, Roskam Aviation and Engineering Corporation, Ottawa, KS, 1979.

12. WEAPONS SYSTEMS INTEGRATION

The purpose of this chapter is to document design work and decisions made regarding the weapons systems integration. The Monarch is designed for three different missions.

Counter Air Mission
* M61 Vulcan 20mm cannon with 400 rds
* 2 Short Range Air-to-Air Missiles
* 2 Medium Range Air-to-Air Missiles

Battlefield Air Interdiction Mission #1
* M61 Vulcan 20mm cannon with 400 rds
* 2 AGM-88 HARMs
* 4 Mk-82 Bombs

Battlefield Air Interdiction Mission #2
* M61 Vulcan 20mm cannon with 400 rds
* 4 AGM-65 Mavericks
* 2 Mk-82 Bombs

Section 12.1 will detail the integration of the M61 Vulcan and ammunition drum. Section 12.2 will discuss the Counter Air (CA) mission weaponry while Section 12.3 will discuss the Battlefield Air Interdiction (BAI) mission weaponry. Section 12.4 will present a description of a constructed scale model of the internal weapons bay.

12.1 INTEGRATION OF THE M61 VULCAN CANNON

This section will address the integration of the M61 Vulcan cannon and the required ammunition drum and ammunition feed system. The material in this section is organized as follows:

12.1.1 M61 Vulcan Cannon Placement
12.1.2 Ammunition Drum Placement
12.1.3 Structural and System Requirements

12.1.1 M61 Vulcan Cannon Placement

Table 12.1 presents information available on the M61 Vulcan cannon:

Table 12.1 M61 Cannon Specifications (Ref 12.1)

Uninstalled Weight:	264	lbs
Maximum Rate of Fire:	6,000	RPM
Average Recoil Force:	3,980	lbs at 6,000 RPM
Ammunition (400 rounds):	M50	series, 20mm
Unit Ammo Weight:	0.55	lbs

Table 12.1 continued M61 Cannon Specifications (Ref 12.1)

Overall Length:	74	in
Barrel Length:	53	in
Maximum Diameter:	9.90	in
Muzzle Diameter:	4.90	in

Factors considered in placement of the cannon in the airframe include:
* avoiding locations where muzzle flashes may degrade night vision of the pilot,
* avoiding locations where the engine inlet may possibly ingest muzzle exhaust gases,
* locating the cannon with ample fuselage volume for maintenance,
* and avoiding locations near vibration sensitive sensors.

For the Monarch to be an effective all-weather, day-or-night fighter, the placement of the cannon muzzle must be so that the muzzle flashes do not enter the direct or peripheral vision of the pilot. Thus, the muzzle must be located either on the bottom surface of the fuselage where the fuselage shields the muzzle from the pilot or sufficiently aft of the pilot.

Gun exhaust gasses are highly corrosive to the fan, compressor, and turbine blades. Every effort should be made to assure that these gases do not enter the inlets of the engine. This may be accomplished by placing the muzzle:
* behind the inlets as on the Dassault Mystere,
* outboard of the inlets as on the McDonnell Douglas F-15 Eagle,
* so that the slipstream around the fuselage carries the exhaust gases away from the inlet capture area as on the Northrop F-20 Tigershark and McDonnell Douglas F/A-18 Hornet,
* or where the wing or fuselage shields the inlets from the exhaust gases as on the Mig-29 Fulcrum and General Dynamics F-16 Fighting Falcon.

To satisfy the constraints mentioned, the cannon may be placed:
* in the blended area between the fuselage and the upper surface of the inlet and behind the inlet face,
* or below and behind the inlet face.

The second option is chosen for the Monarch because:
* the shape of the fuselage (expanding from the narrow nose/cockpit section to the wide aft fuselage section allows for relatively easy exposure of the muzzle, reducing the need for special fairings around the gun or muzzle;
* the muzzle is hidden from the view of the pilot since it is relatively far aft of the pilot and beneath the inlet,
* and sufficient volume is nearby for the ammunition drum and feed system.

Figure 12.1 presents this location of the cannon in the Monarch airframe. Two problems arise from this cannon location, and need to be addressed:

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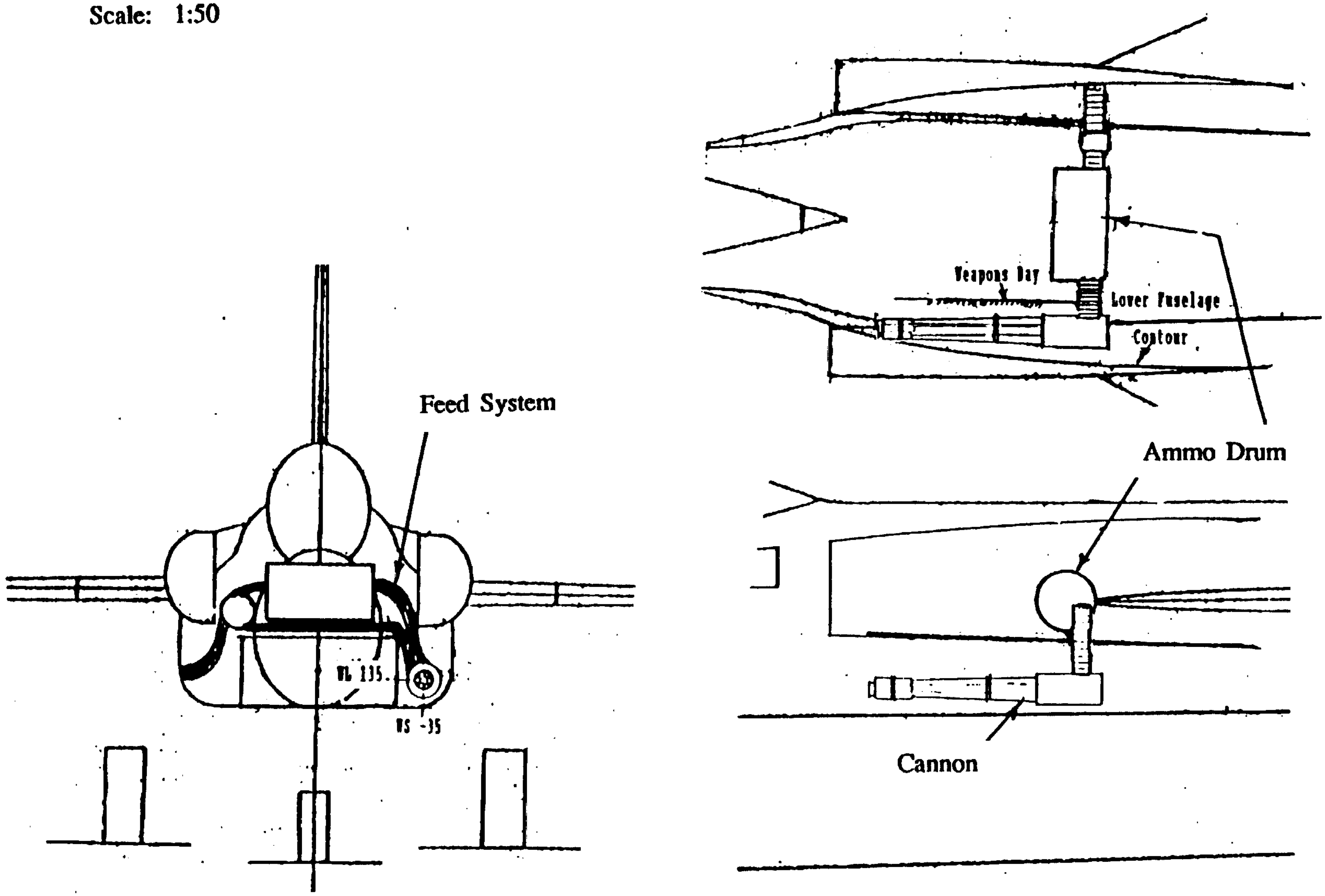

Figure 12.1 Vulcan Cannon Location in Monarch Airframe

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* Since the cannon is not on the centerline of the aircraft, adverse yawing moments will be created. If this proves to be a problem, rudder feedback may be required upon firing of the cannon.

  At 6,000 rounds/min, the maximum yawing moment created by the recoil force is 11,600 ft-lbs. However, 400 rounds of ammunition last only 4.1 seconds (allowing one-third second for the cannon to reach its maximum rate of fire) (Ref 12.2). This large of a moment is not a likely scenario since the pilot is more likely to fire several small bursts to conserve ammo. A larger magnitude problem in the F-15, where the cannon placement is 67" outboard and the ammo drum contains 940 rounds, required no rudder feedback (Ref 12.2).

* Nearby accelerometers and other sensors may have to be insulated from the vibrations created by the firing cannon. This proved to be the cause of two F-16 accidents in the summer of 1979. Cannon vibrations were sending false readings to the flight control computer (Ref 12.2).

12.1.2 Ammunition Drum Placement

The mission specifications for the Monarch stipulate 400 20mm rounds. This amount of ammo requires 5.2 cubic feet of volume for the drum. The dimensions for the drum are:

Diameter:	18"
Length:	35"

Given the gun placement, the ammunition drum may be located either:
* behind the cannon as on the Grumman F-14 Tomcat (see Figure 12.2) and Fairchild A-10 Thunderbolt, below the inlet, and between the internal weapons bay and outer fuselage,
* or above the internal weapons bay and between the bifurcated inlets similar to the General Dynamics F-16 Fighting Falcon, see Figure 12.3).

The first option is desirable because of the ease of access to the drum for maintenance and reloading. It also does not sacrifice any fuel volume in the center of the aircraft. However, there is insufficient volume available for clearances (less than 2" of clearance laterally), the weight of the ammo is displaced from the aircraft centerline, and complicated twists are required in the feed system.

The second option is chosen for the Monarch primarily for simplicity. The Monarch ammunition drum and feed system is shown in Figure 12.4 on the previous page. The system is much like that used on the F-16, see Figure 12.2. The drum is located above the weapons bay and below the upper fuselage surface. Access to the drum for maintenance reasons is through access panels on the upper fuselage.

The feed system uses linkless ammunition and fits through the 6" clearance between the corner of the weapons bay and the inlet. New rounds of ammunition from the drum feed to the cannon. The used casings then are fed above the weapons bay and underneath the ammo drum to the starboard side where they are fed back into the ammo drum. During reloading procedures, the expended cartridges are dispensed simultaneously while

Scale: None

Ref. 12.3

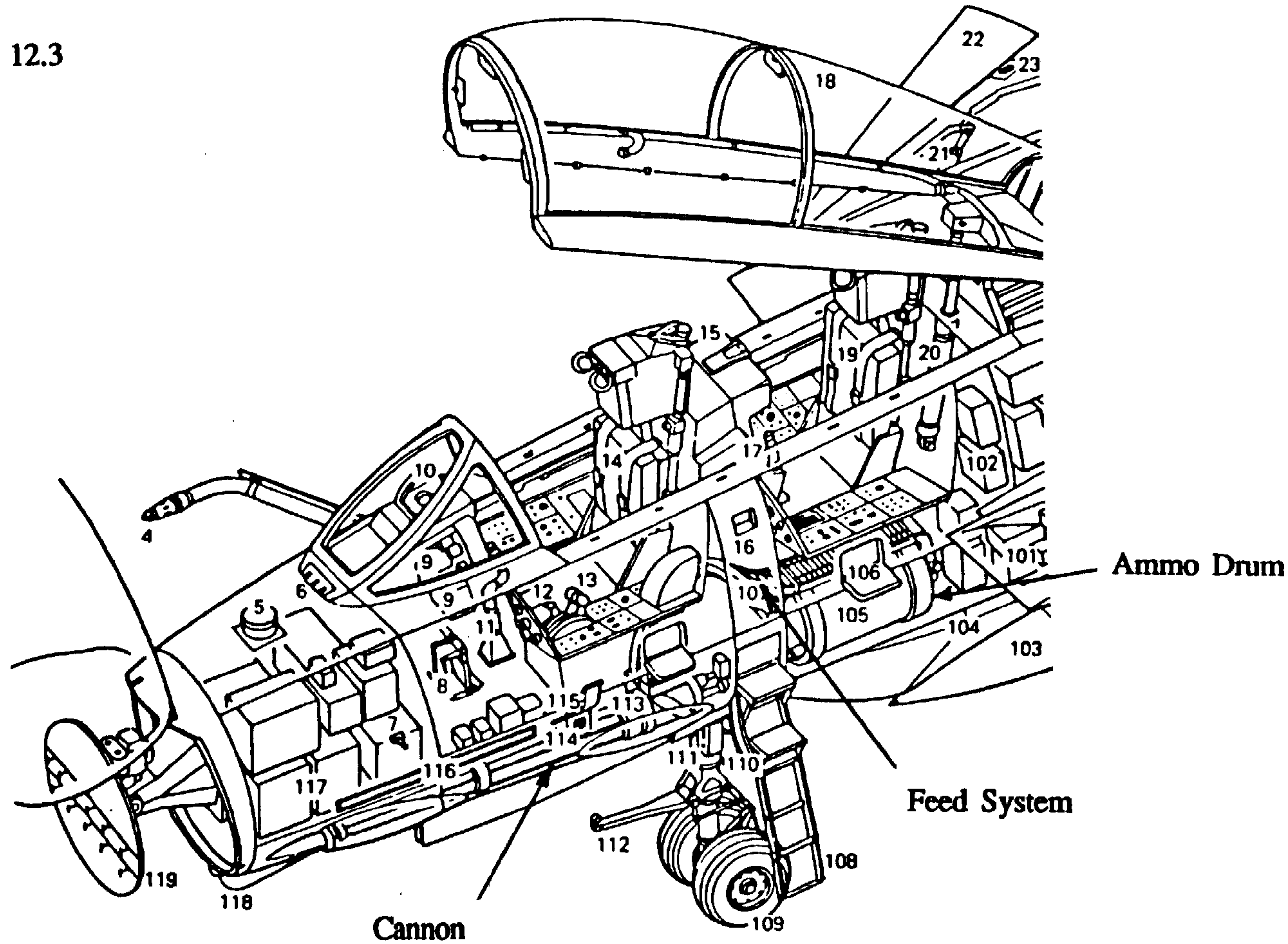

Figure 12.2 Grumman F-14 Tomcat Cannon Installation

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Scale: None

Ref. 12.3

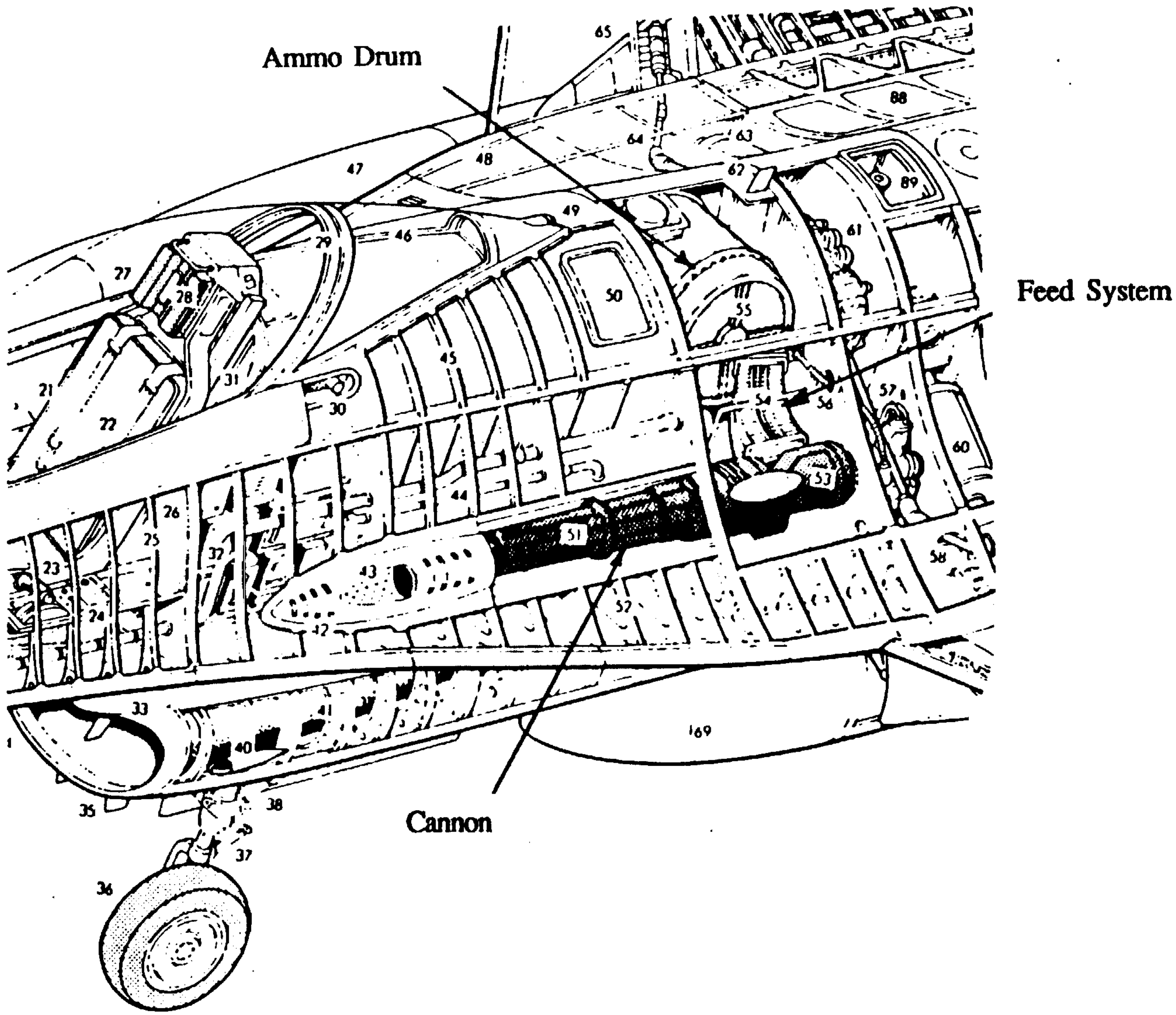

Figure 12.3 General Dynamics F-16 Falcon Cannon Installation

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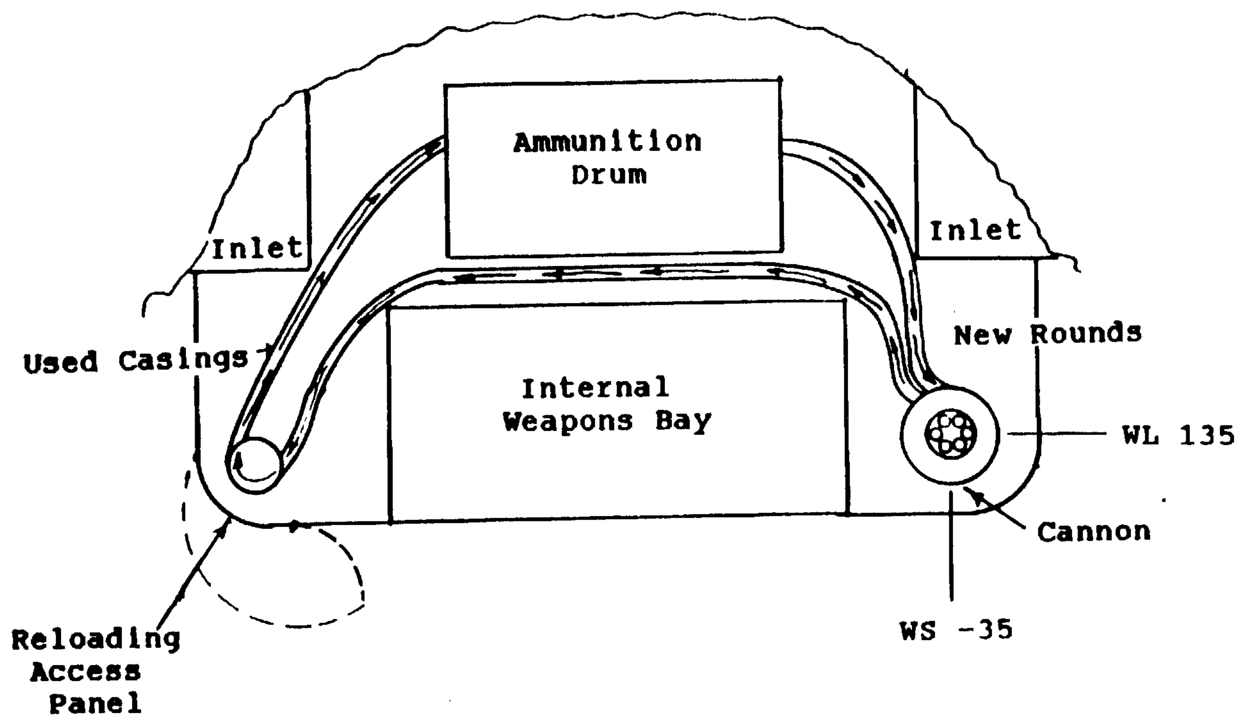

Cross-Section as FS 340

Figure 12.4 Monarch Ammunition Drum and Feed System the new rounds are loaded through the starboard loading access panel.

Reloading of new cartridges and unloading of used casings is accomplished through an access door on the starboard side of the fuselage underneath the inlet that is approximately 1 foot square. This low location will allow for easy reloading by armorers since they do not need to climb on the aircraft or use ladders. However, simultaneous loading of ammunition and Sparrow missiles into the internal missile bay will be hampered. This is rather unavoidable due the location of the bay. The problem could be avoided if the cannon was nose mounted. The problem may be reduced by simultaneously reloading ammo and the port side ordnance before loading the starboard ordnance.

12.1.3 Structural and System Requirements

Structural Requirements

The Vulcan cannon will mount on a frame at FS 350. This frame is currently designed for several other uses (see Chapter 7). Other gun mounts where the barrel will need support are at FS 283 and FS 313. The ammunition drum will also mount to the bulkhead at FS 350 above the internal weapons bay. Cut outs will have to be provided for the muzzle at FS 276 (with appropriate cooling vents), for the reloading access panel, and for the ammo drum and cannon access panels.

System Requirements

The Vulcan and feed system may be powered electrically, hydraulically, or both. The cannon is triggered electrically. by a signal from the cockpit (Ref 12.2).

12.2 COUNTER AIR MISSION WEAPONS INTEGRATION

This section will present the detail design and decision made regarding the weapons integration for the Counter Air mission. The material in this chapter is arranged as follows:

12.2.1 Short Range Missile Integration
12.2.2 Medium Range Missile Integration
12.2.3 Structural and System Requirements

The Counter Air mission uses the following weapons:
* 2 Short Range Air-to-Air Missiles
* 2 Medium Range Air-to-Air Missiles

The AIM-9 Sidewinder and AIM-7 Sparrow are currently in the US inventory as short range and medium range air-to-air missiles, respectively. The ASRAAM and AMRAAM projects, replacements for the Sidewinder and Sparrow, are experiencing difficulties. Since the AIM-9 and AIM-7 missiles are larger than the ASRAAM and AMRAAM, the weapons integration will be designed for the larger missiles, assuming the smaller missiles can be integrated easily in the future. Table 12.2 presents available data for the Counter Air mission weapons:

Table 12.2 Counter Air Weaponry Data (Ref 12.1)

	AMRAAM	AIM-7F	AIM-9J	AIM-9L	ASRAAM
Guidance Method	Radar	Semi-Active Radar	IR	IR	IIR
Range (miles)	>100	24	6	10	11.9
Launch Method	Eject	Eject or Rail	Rail	Rail	Rail
Weight (lbs)	327	514	170	191	150
Overall Length (in)	144.0	146.0	113.0	115.0	98.4
Body Diameter (in)	6.8	8.0	5.0	5.0	5.9
Fin Span (in)	25.0	40.0	25.0	24.0	17.7

12.2.1 SHORT RANGE MISSILE INTEGRATION

It was originally intended to carry the Short Range Missiles internally. However, after further consulting (Ref 12.3), it was decided to mount the AIM-9Ls on wingtip launchers because:

* pre-launch target acquisition is required.
* the wingtip launchers provide the AIM-9L with a larger field of view, allowing better acquisition of targets. The proximity of the fuselage severely limited the field of view if internal storage was employed.
* wingtip launchers increase the effective aspect ratio of the wing, thus increasing the wing efficiency and reducing induced drag (Ref 12.4).
* reliability and simplicity of the system increase the effectiveness of the weapon and reduce the cost of both the weapon and aircraft.

The disadvantages of mounting the Sidewinder missiles on the wingtips are:

* a decrease in stealth,
* an increase in parasite drag,
* and an increase in rolling moment of inertia.

Figure 12.5 illustrates the wingtip launcher and Figure 12.6 illustrates a schematic diagram of the missile-restraint device within the wingtip launch rail. The rail must guide the missile during launch, yet it must hold the missile in place during +9/-3 g maneuvers and retain the missile in the case of an accidental rocket motor ignition. With the locking mechanism in place, the missile is not allowed to move until the missile is selected and the locking mechanism is raised. The blocking mechanism spring is designed to retain the missile in a low acceleration environment so that upon motor ignition the missile lug will

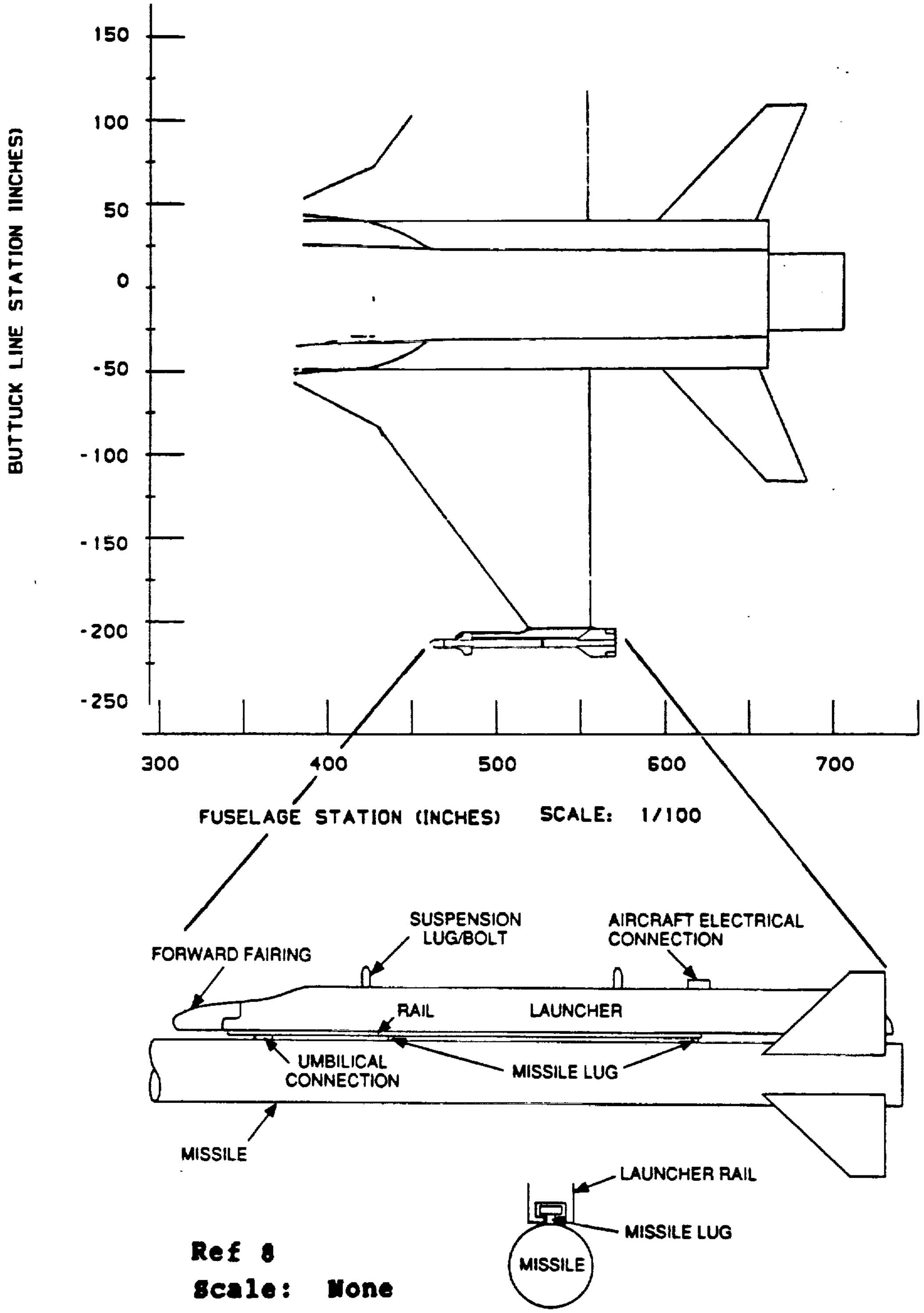

Figure 12.5 Monarch Wingtip Rail Launcher

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Ref. 12.5

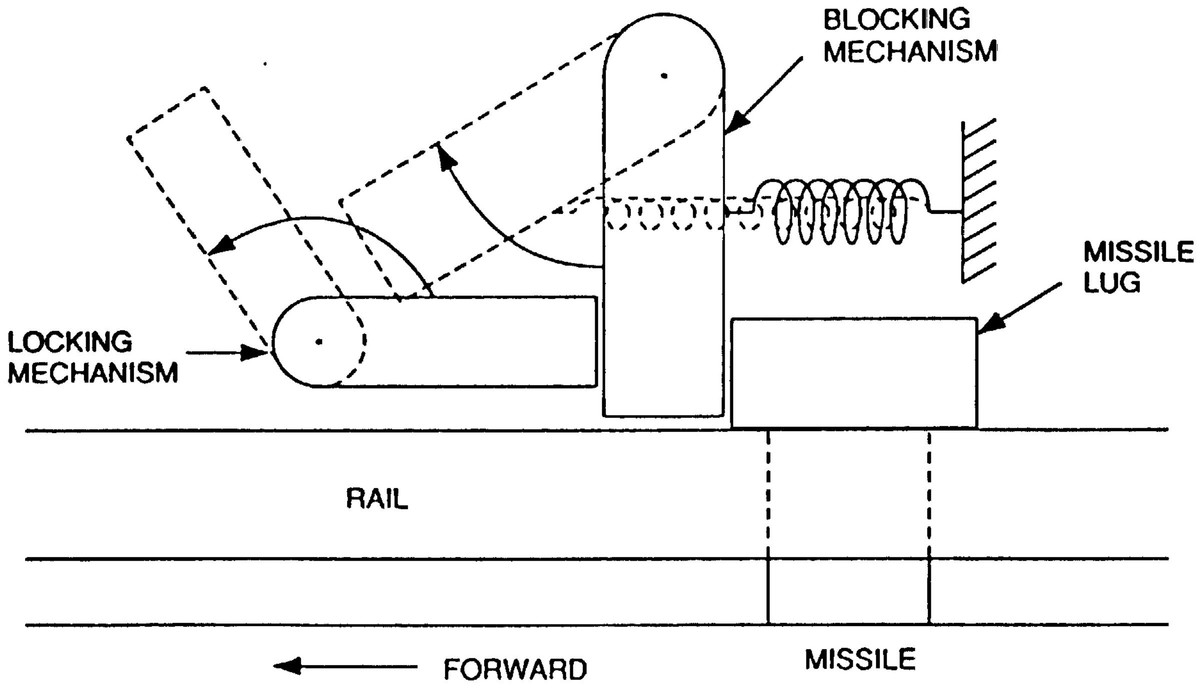

Figure 12.6 Schematic Rail Launcher Holdback Restraint

rotate the blocking mechanism forward. The missile umbilical will separate when the missile moves forward along the rail. The connector is designed to separate when subjected to shear loads.

12.2.2 MEDIUM RANGE MISSILE INTEGRATION

Internal storage of the medium range missiles is specified in the mission requirements. The Sparrow may either be ejected or rail launched, it is decided to eject the missile since the ejection mechanism requires less volume than a retractable rail launcher.

Due to the large span of the AIM-7F (40 inches), the fins are folded. Figure 12.7 presents the fin folding scheme of the AIM-7F. The fin fold layout was determined considering:

* all four fins should be the same and have only one hingeline to reduce the cost.
* the minimum volume is created by making the missile "square" when stored.
* the fins may not strike each other when they deploy.

Launch Sequence

The medium range missile may incorporate systems similar to those used on the AIM-7M in the future. The AIM-7M uses the AIM-7F airframe but incorporates a new digital guidance section. It is designed for improved capability in look-down and ECM environments. The AIM-7M also incorporates a LTE/BIT. During the Launch-To-Eject (LTE) cycle a Built In Test (BIT) is conducted. If the LTE/BIT detects a missile failure, the missile will not be launched. The LTE cycle typically requires 1 to 2 seconds depending primarily on the time required for gyroscope run-up which typically takes 0.75 to 1.5 seconds (Ref 12.5). Figure 12.8 presents a typical LTE cycle. However, in the case of the internal weapons bay, the weapons bay doors open when the trigger is pulled. Since the pilot currently experiences a 2 second delay between trigger and launch, it is believed that the door actuators may be sized to open at more than 45 deg/sec. The BIT will have an additional "doors open" test before ejection. Upon jettison of the missile, separation of the umbilical, and predetermined linear acceleration, the fins deploy and the rocket motor ignites. The missile then undergoes a preprogrammed maneuver until it clears the aircraft.

Ejector Design

For safe, reliable and effective use of the AIM-7F, the missile must be ejected clear of the aircraft. To achieve clean separation, major concerns that need to be tested and simulated are (Ref 12.5):

* that the missile will not strike the launch aircraft during powered (motor-fire) or jettison (no motor-fire),
* that the missile will not strike the aircraft as a result of a failure during separation,
* and that the missile rocket blast will not adversely affect the launch aircraft.

Key parameters for design of the required ejector are the linear velocity and angular velocity imparted to the missile at the end of the ejection stroke. Typical linear velocities are 18-20 ft/sec and typical nose-down angular rates are from 0-30 deg/sec depending on

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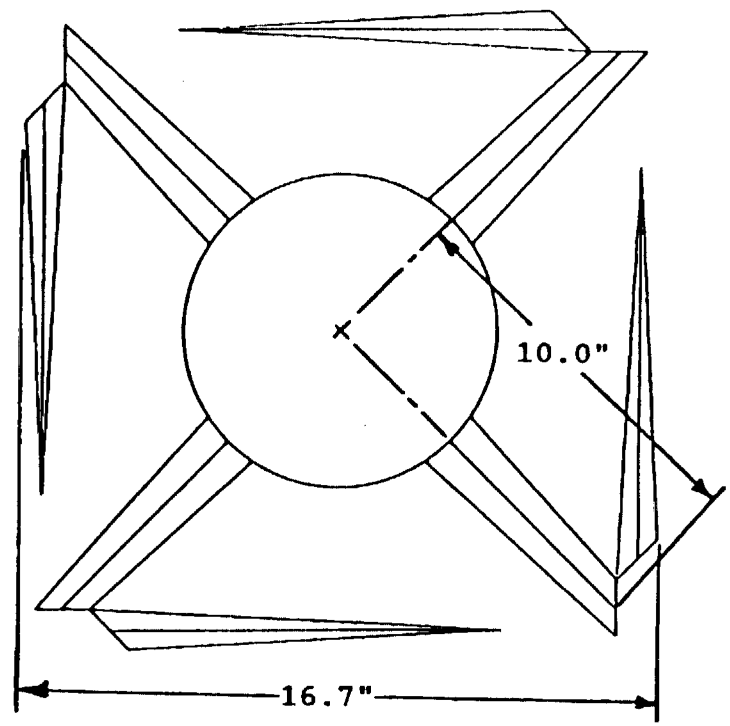

Figure 12.7 AIM-7F Sparrow Fin Folding Layout

Ref. 12.5

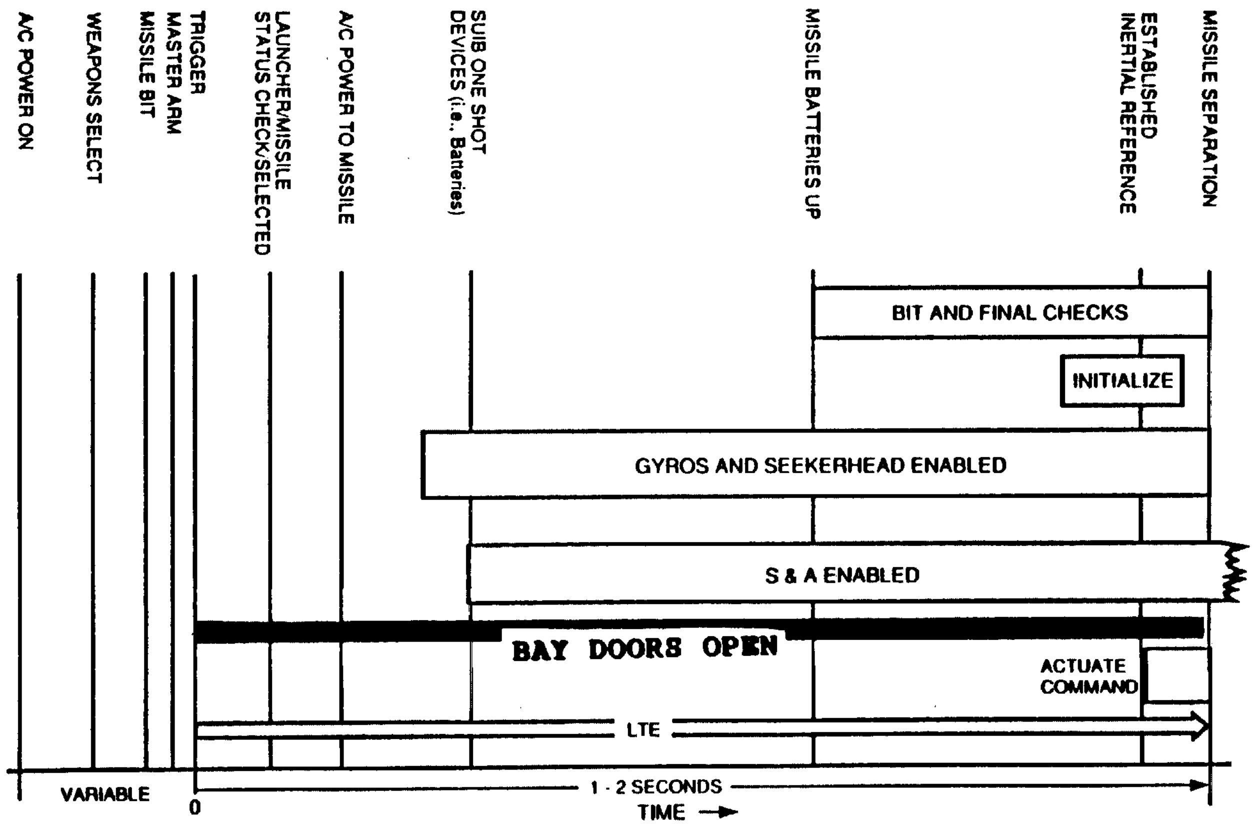

Figure 12.8 Monarch Sparrow Launch-to-Eject Cycle

the weight, moment of inertia, and installation of the missile (Ref 12.5).

The limiting factor on linear velocity is the structural limit of the missile. To achieve 20 ft/sec linear velocity, average accelerations of 10-20 g's and peak accelerations of 30g's are experienced by the missile. The angular rate is limited by the capabilities of the seekerhead stabilization loop (Ref 12.5). Because of the ejector location (inside the fuselage as opposed to on the fuselage surface, as is the case on both the F-15 and F/A-18), it is assumed that the ejectors will have to be powerful and deliver high linear and angular velocities. To determine the correct velocities, extensive simulation, tunnel modeling, and full-scale flight testing are required (Ref 12.5).

A typical ejector is illustrated in Figure 12.9. The missile is mechanically attached to the ejector. Separate ejector feet are used to eject the missile. Upon ignition of the pyrotechnic cartridges in the breech (if one fails to ignite, one will ignite the other), the exhaust gases release the mechanical links between the missile and the ejector as the required pressure builds within the ejector cylinders. At the end of the ejector stroke, the gases are vented out of the pistons and simple springs retract the piston into the cylinders. The required angular rate and linear velocity are controlled by a gas flow control valve between the breech and each ejector cylinder (Ref 12.5). The pyrotechnic cartridges may be replaced in the ejector rack from the side without removing the missile.

Missile Bay Design

Given the allotted volume and required missile and ejector rack volumes, the minimum volume for the internal missile bay is:

Length:	156"
Width:	48"
Depth:	22"

The design of the internal missile bay and ejector are shown in Figure 12.10. Each door operates independently and consists of two panels. The inboard panel is hinged near the aircraft centerline and two rotary actuators rotates each door about the hinge. The inboard edge of the outboard panel is hinged to the outer edge of the inboard panel. The outer edge of the outboard panel follows two lateral tracks on the forward and aft ends of the missile bay. When opening, the inboard panel will rotate down 90 degrees and the outer panel will rotate -90 degrees while traversing inboard. This door design was chosen because:

* the doors are opened only briefly and the degradation in handling qualities and increase in drag is not as severe as if the doors were opened for a longer period of time. The increase in airplane drag coefficient from the opened weapons bay and doors at Mach 1.6, 30,000 ft is estimated to be 5 drag counts, an increase of 1.2% in zero lift drag. See Appendix 5 for the calculations.
* the door design is relatively simple, only four moving parts: two door panels and two actuators,
* the doors open to the center so that armorers have unobstructed access to the weapons bay,

Scale: None Ref. 12.5

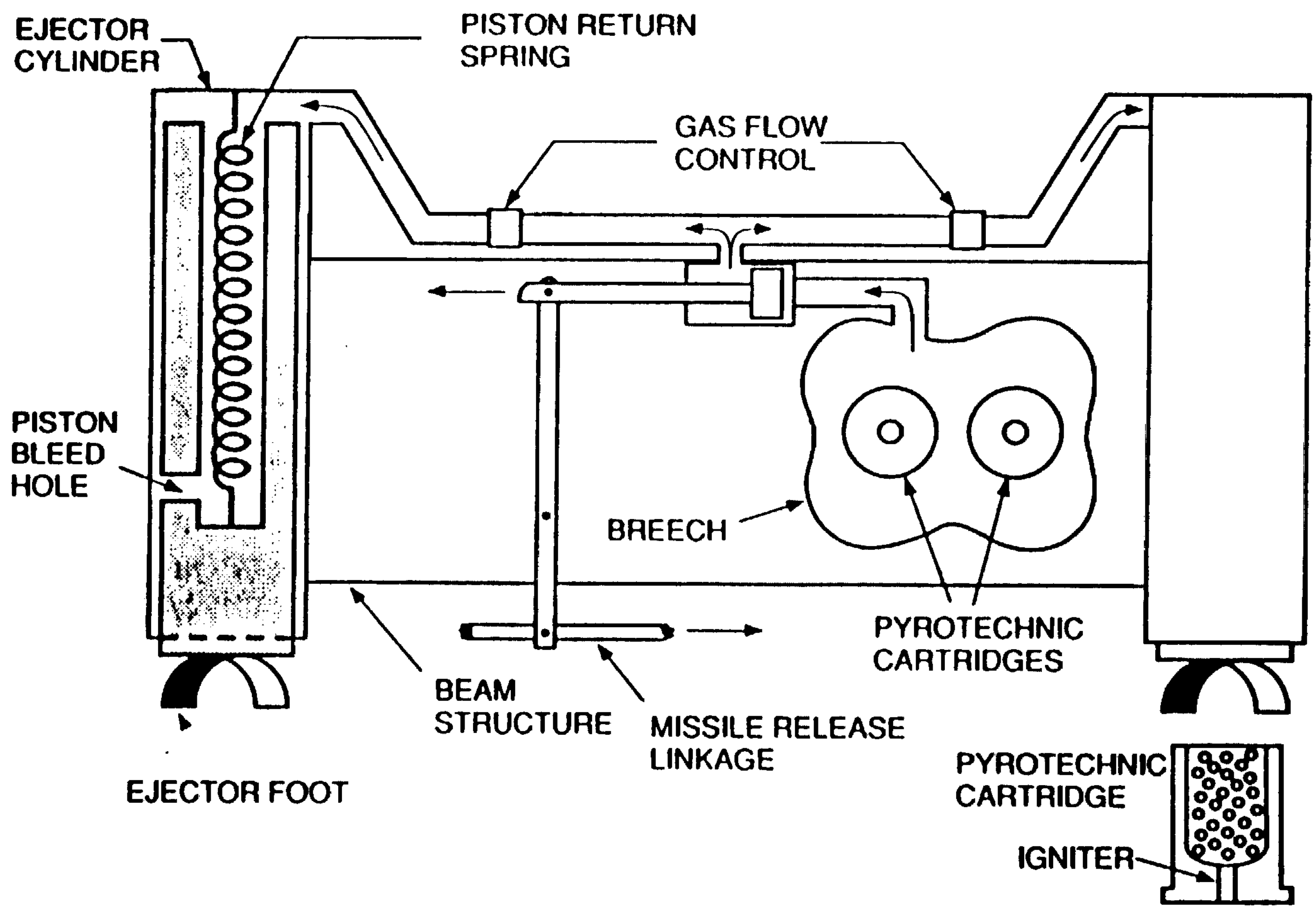

Figure 12.9 Typical Ejector Launcher

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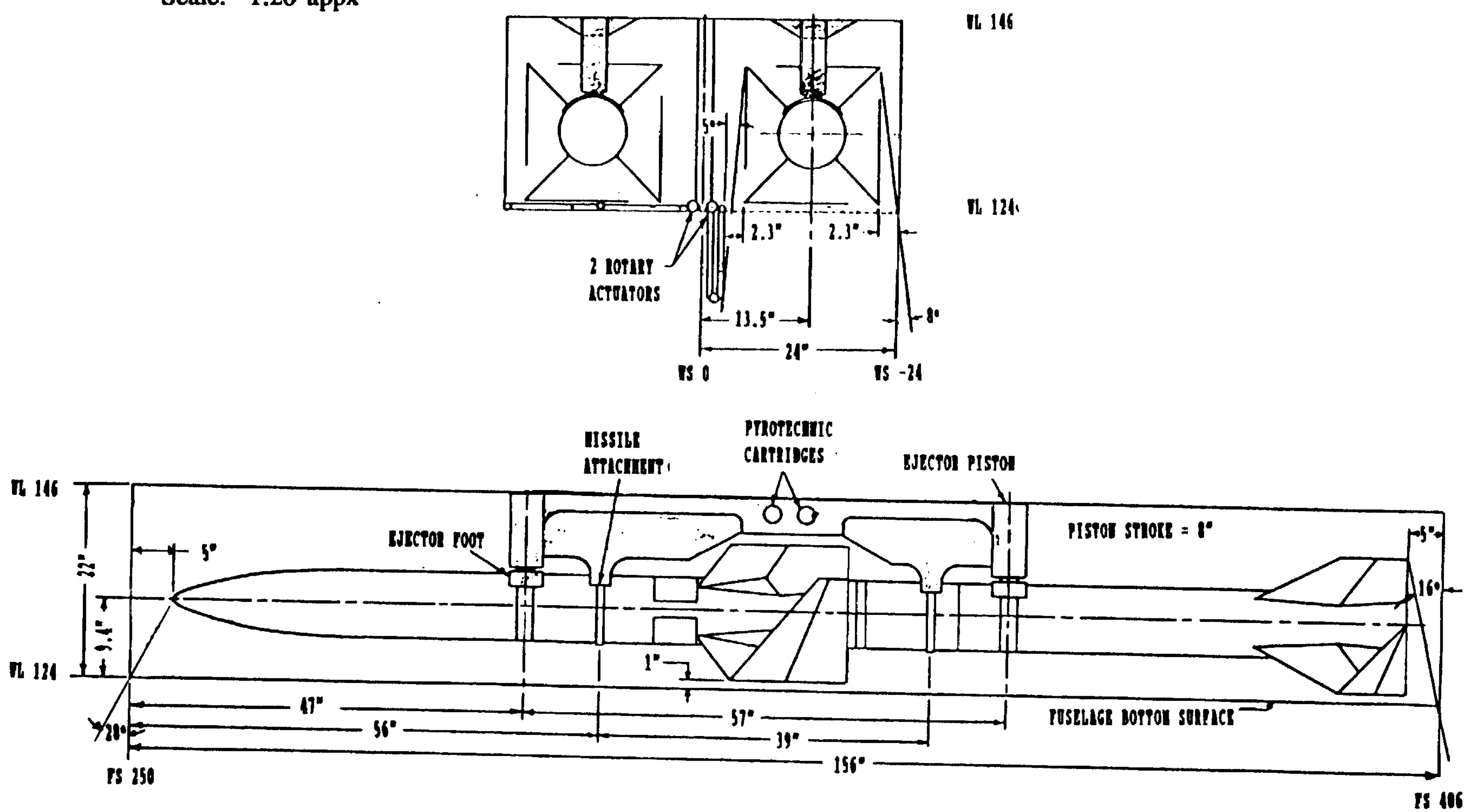


Figure 12.10 Monarch Internal Weapons Bay Arrangement

* and the division of the weapons bay along the centerline allows for the addition of a keel beam along the fuselage and should aid in carrying the loads around the smaller cutouts in the fuselage.

The clearance angles between the missile bay and the missile are:

Forward:	28 deg
Aft:	16 deg
Inboard:	5 deg
Outboard:	8 deg

Reference 12.4 suggests 10 degrees of clearance both laterally and longitudinally. Since the missile is relatively guided though the bay rather than dropped, the laterally clearances are assumed to be acceptable. The large longitudinal clearances allow for door tracks and other structure.

12.2.3 Structural and System Requirements

Structural Requirements

The following structural requirements are needed for the integration of the Counter Air mission weaponry:

Attachments are required on the wingtips for the launch rails. The internal weapons bay outer walls are located at:

WS	+/-24
FS	250
FS	356
WL	124
WL	146

The forward and aft walls of the weapons bay must provide for door track attachments. The beam along the centerline must provide for actuator attachment and may be no more than two inches wide at the bottom.

The top of the weapons bay needs extra structure near the ejector pistons at FS 293 and FS 350 at WS +/-13.5.

System Requirements

All that is required for the wingtip launchers is electric power and trigger signal to the launch rail. The internal weapons bay requires four rotary actuators, either electric or hydraulic. The ejector rack and missile require electric power and signaling.

12.3 BATTLEFIELD AIR INTERDICTION MISSION WEAPONS INTEGRATION

This section discusses the integration of the weapons required for the two Battlefield Air Interdiction missions. The two mission loadings are as follows:

* BAI #1: 2 AGM-88 HARMs and 4 Mk-82's
* BAI #2: 4 AGM-65 Mavericks and 2 Mk-82's

The material in this chapter is arranged as follows:

12.3.1 Battlefield Air Interdiction Mission Weapons Arrangement
12.3.2 Structural and Systems Requirements

12.3.1 Battlefield Air Interdiction Mission Weapons Arrangement

Two primary factors are considered in determining the placement of the BAI weapons. Due to their large size and the lack of a supercruise requirement, it is decided to carry BAI weapons externally. This arrangement also allows simultaneous carriage of the counter air weaponry (2 AIM-9Ls and 2 AIM-7Fs). If required, two auxiliary fuel tanks may be fitted into the internal weapons bay which add approximately 35 cubic feet or 1,700 lbs of additional fuel (see Appendix 5 for the calculations).

Table 12.3 presents available data on the BAI ordnance:

Table 12.3 BAI Ordnance Specifications (Ref 12.1)

	AGM-88A HARM	AGM-65 Maverick	Mk-82 Slick	Mk-82 Snakeye
Guidance Method	Radar	TV IIR Laser	None	None
Range	13 miles	N/A	N/A	N/A
Launch Method	Ejection	Rail	Drop	Drop
Weight (lbs)	807	463	521	560
Overall Length (in)	164.0	98.0	87.0	88.5
Body Diameter (in)	10.0	11.8	10.8	10.8
Fin Span (in)	44.0	28.3	16.0	16.0

Due to the large weight of the ordnance, it is desirable to keep the weapons as far inboard as possible in order to reduce asymmetric loads during hover and the aircraft rolling moment of inertia. The considerations that limit the placement of the weapons are:

* conformal fuselage mounting is rejected because of the limited space available between the weapons bay doors, lift engine, and landing gear doors. Also, during hover, the fountain core that is created impinges on the fuselage at approximately FS 400 would heat up the stores.
* the HARMs and Mavericks are considered to be "high value" stores when compared to the gravity bombs. It is undesirable to have to require the jettison of "high value" stores in order to reduce an asymmetric load for balance in hover. Thus the HARMs and Mavericks are carried inboard of the Mk-82's.

For BAI mission #1, the HARMs are carried on an inboard wing pylon. The tail of the missile determines both the lateral and longitudinal placement of the missile in that:

* the tail must be forward of the trailing edge devices,
* and the tail must be able to clear the main gear doors and the deflected gear upon landing.

The vertical placement of the HARM is limited primarily by clearance of the leading edge devices.

This places the inboard pylon at WS 85 and the center of gravity of the HARM at:

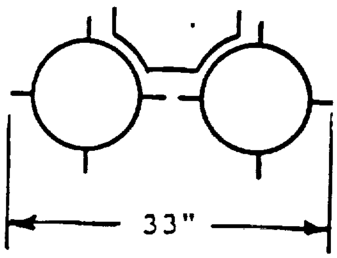
FS	400
WS	85
WL	130

The Mk-82's are carried on an outboard wing pylon. They are placed outboard sufficient for clearance of the HARMs. The Mk-82's are mounted on a twin stores ejector. Figure 12.11 presents approximate lateral clearances of the twin stores ejector. Figure 12.12 presents the current layout of the pylons for BAI mission #1.

For BAI mission #2, two Mavericks are carried on the inboard wing pylon. To allow for adequate growth, lateral space is allowed for a Maverick triple rail launcher. The clearances are illustrated in Figure 12.11. Another constraint applicable to the Maverick carriage is that the seeker head of the Maverick has a 5 degree half-angle cone of vision (Ref 12.2). These angles are demonstrated in Figure 12.13 on the layout of the weapons for BAI mission #2.

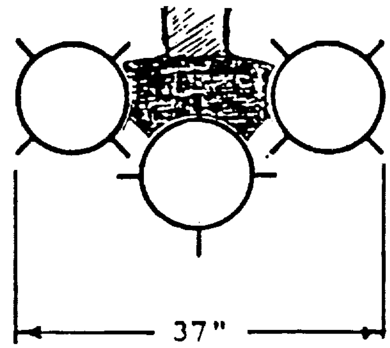
The Mk-82's during BAI #2 are mounted on single ejector racks on the outboard pylon. The outboard pylon is placed such that allowable clearances are made for simultaneous carriage of 3 Mavericks and 3 Mk-82's on each wing. The approximate lateral clearance of the Mk-82's on a triple ejector rack is illustrated in Figure 12.11. The outboard pylon is located at WS 135. Figure 12.14 presents the fully loaded Monarch with 6 Mavericks and 6 Mk-82's. Of particular interest are the lateral tip over clearance angles. As seen in Figure 12.14, all stores are satisfactorily inside the line 5 inches above a 5 degree angle line from the main gear contact point (Ref 12.4).

Scale: 1:20



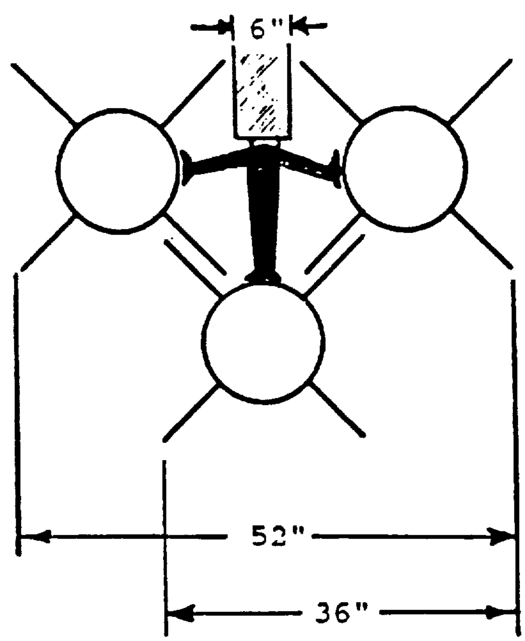
Twin Stores Carriage

(Mk-82s)



Triple Ejector Rack

(Mk-82s)



Triple Rail Launcher with Mavericks

Figure 12.11 External Stores Carriage Racks

Scale: 1:50

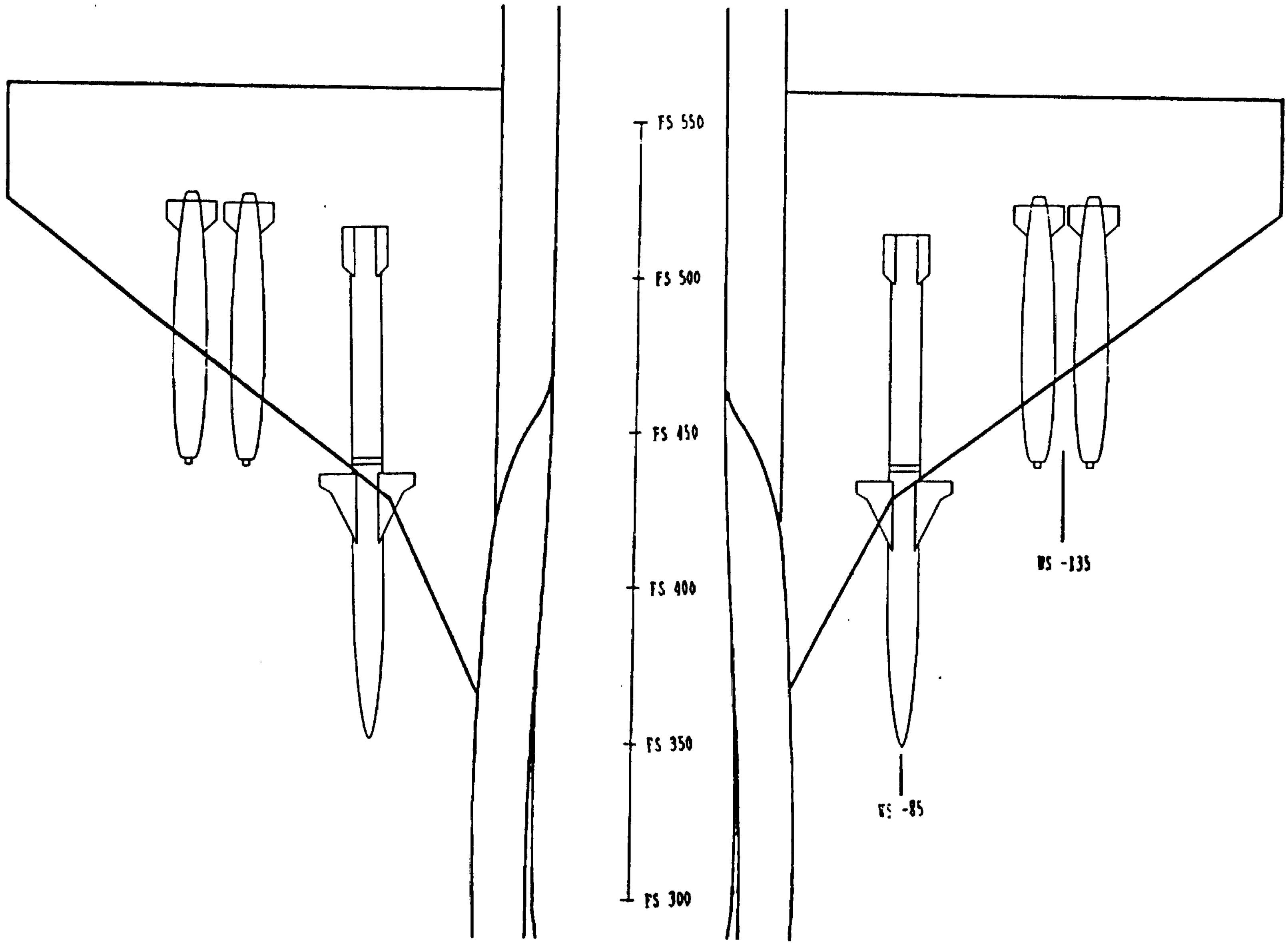
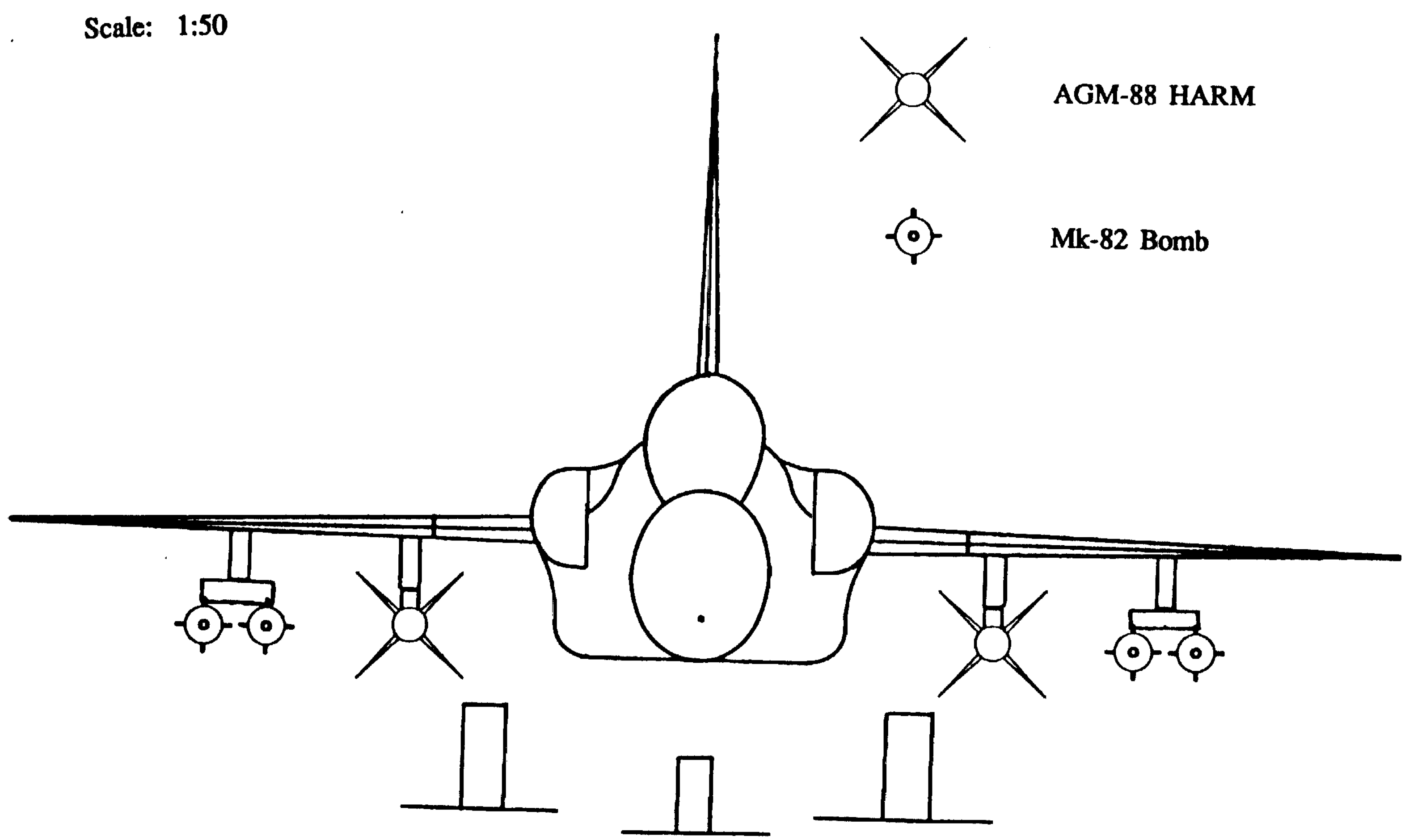


Figure 12.12a Battlefield Air Interdiction Mission #1 Loading (Top View)



<u>Figure 12.12b Battlefield Air Interdiction Mission #1 Loading (Front View)</u>

Scale: 1:50

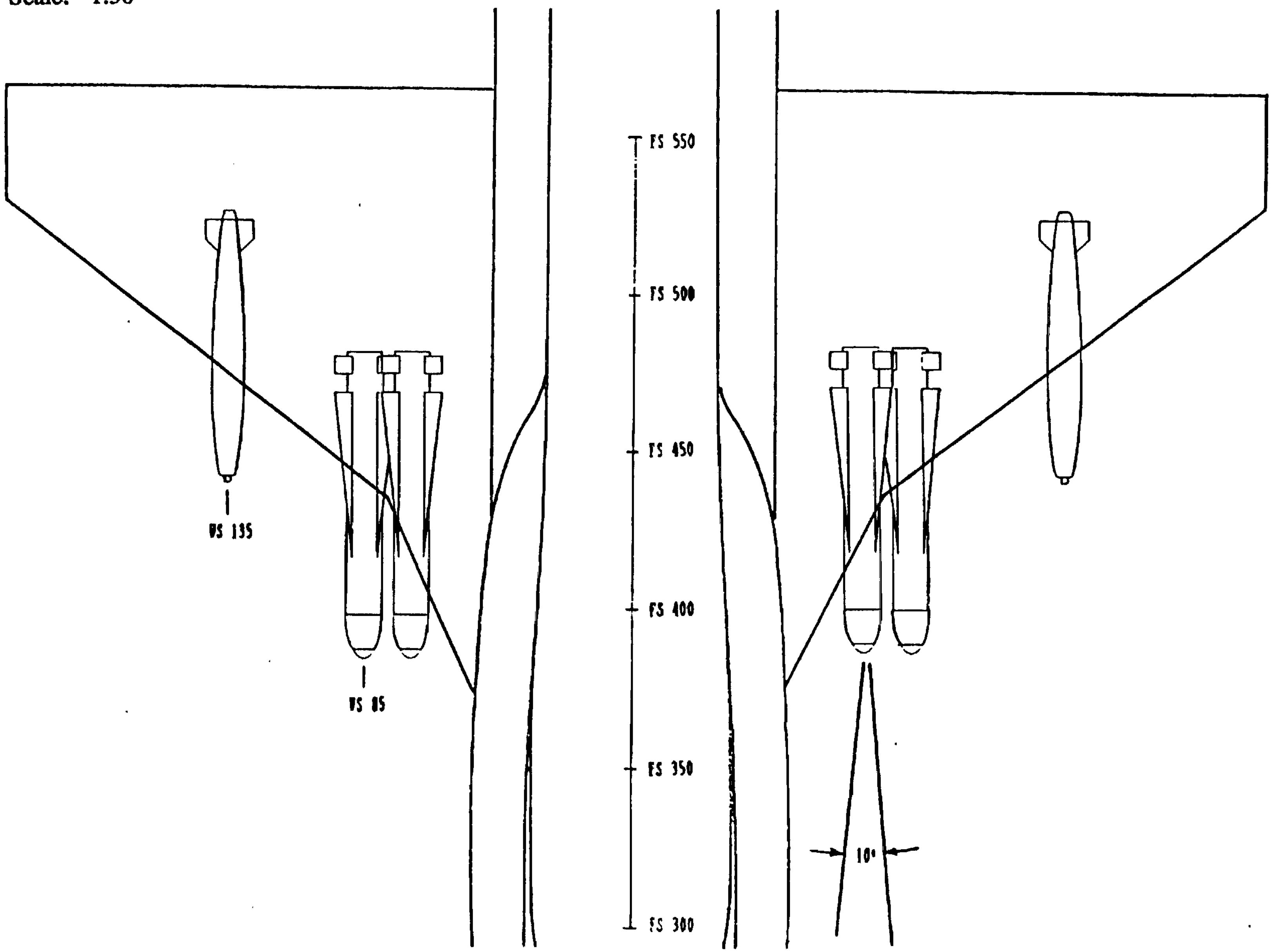


Figure 12.13a Battlefield Air Interdiction Mission #2 Loading (Top View)

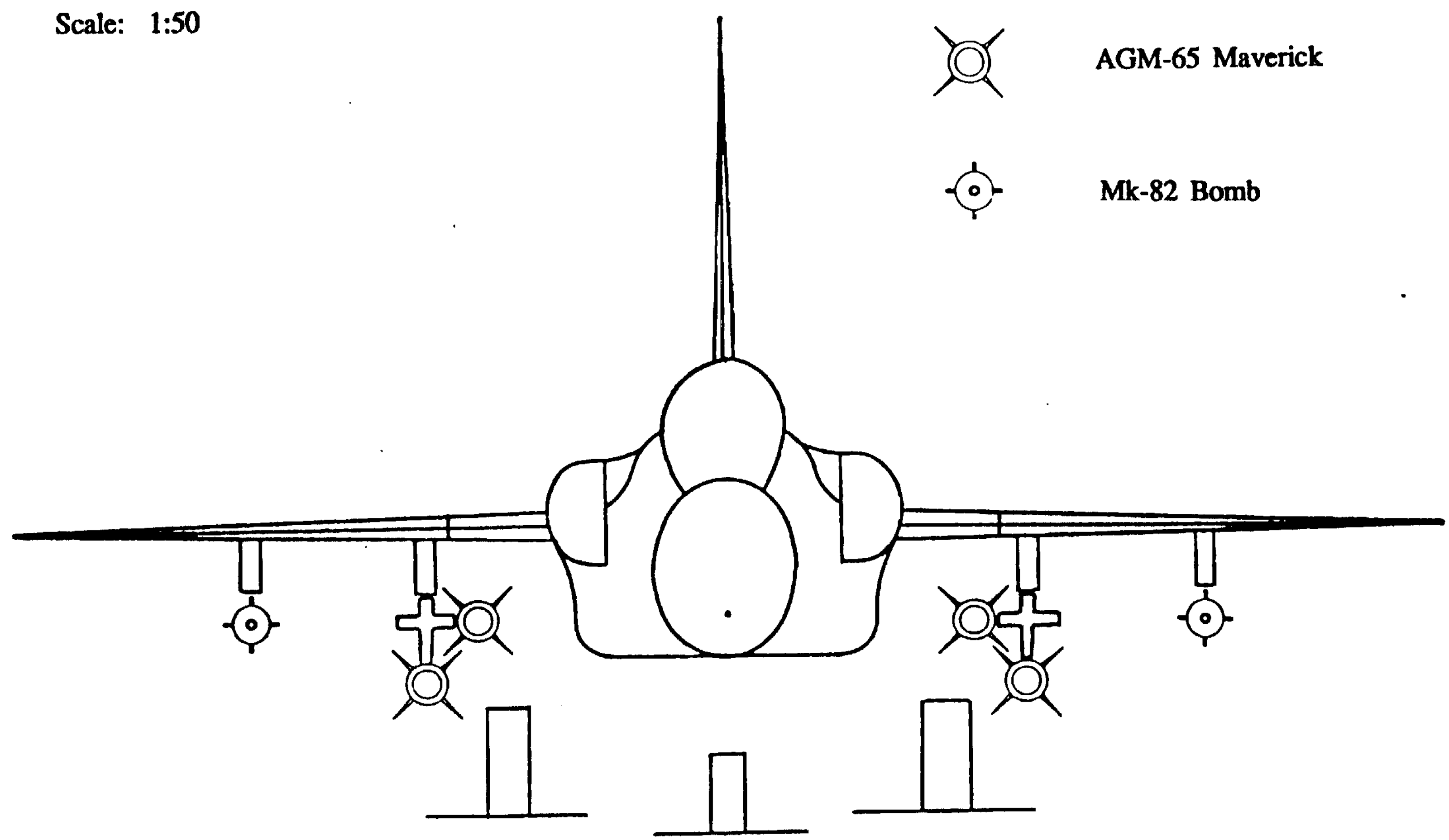


Figure 12.13b Battlefield Air Interdiction Mission #2 Loading (Front View)

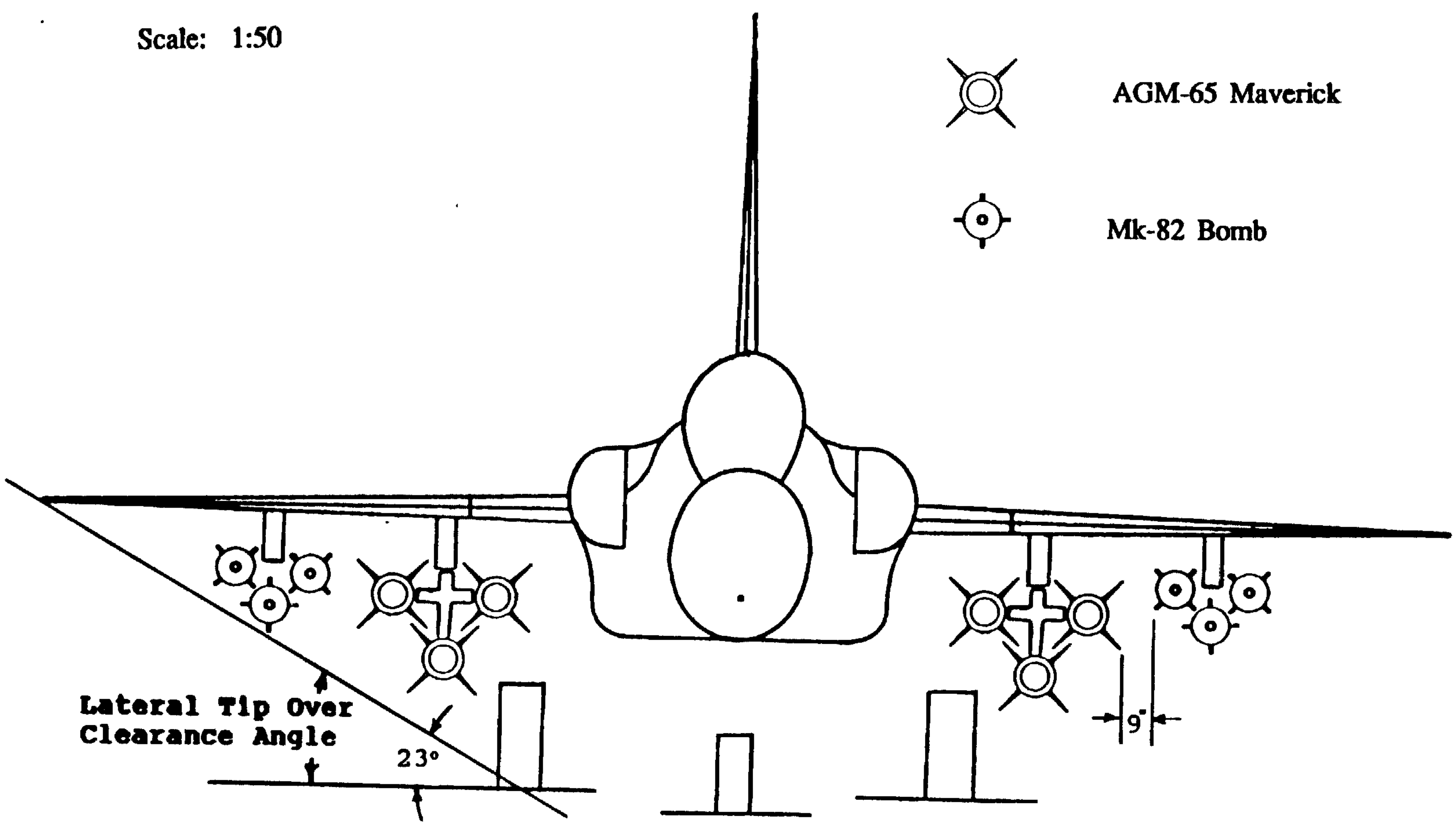


Figure 12.14 Maximum External Stores Carriage Clearances

During hover, assuming that the Mk-82's are jettisoned and assuming that the Mavericks are launched symmetrically (i.e. they were launched from alternate wing locations), the maximum asymmetric rolling moment created is 3,000 ft-lbs. For the asymmetric HARM loading, the moment created is 4,200 ft-lbs.

12.3.2 Structural and Systems Requirements

Structural Requirements

The longitudinal location of the hard points is determined by those in the structures group. They will be slightly aft of the front spar. An additional small attachment will need to be provided near the rear spar for attachment of the pylon.

System Requirements

The only systems required by the Battlefield Air Interdiction mission ordinance pylons are electric power and signalling to each of the wing pylons. Other possible requirements which have not yet been discussed are the addition of fuel lines to the internal weapons bay and the wing pylons to allow the carriage of auxiliary fuel tanks.

12.4 INTERNAL WEAPONS BAY MODEL DESCRIPTION

A model demonstrating the feasibility of the internal weapons bay has been constructed. This model will be transported to the USRA conference in June 1990.

The scale of the model is one-fifth scale. One-tenth scale is too small for intricate pieces and one-half scale is too large for the allotted conference display area.

With the close proximity of the cannon and ammunition drum to the weapons bay the model also includes the M61 Vulcan cannon and ammunition drum installation.

The model is constructed of:

* bass and balsa woods in missile fins, and the cannon,
* plywood in the structural frame and doors,
* metal in the moving parts of the ejector, door tracks, and hinges,
* cardboard for the flexible surfaces such as fuselage, and ammo feed system
* PVC pipe for the ejector pistons and ammunition drum.

The approximate model size in 1/5 scale is:

Length: 32"
Width: 18"
Depth: 13"

Due to the proximity of the inlets and auxiliary inlets, they are incorporated into the model.

Figure 12.15 shows photographs of the completed model.

Photgraph pending completion of model

Figure 12.15 Internal Weapons Bay Model Photograph

REFERENCES FOR CHAPTER 12

12.1 Roskam, Jan, Airplane Design Part IV: Layout Design of Landing Gear and Systems, Roskam Aviation and Engineering Corporation, Ottawa, Kansas, 1989.

12.2 Fitzsimons, Bernard; de Ste. Croiz, Philip; Bonds, Ray; and Hall, Tony, eds., The Great Book of Modern Warplanes, Portland House, New York, 1987.

12.3 Hahn, Andy, NASA Ames Powered Lift Engineer, personal conversation, February, 8, 1990.

12.4 Raymer, Daniel P., Aircraft Design: A Conceptual Approach, American Institute of Aeronautics and Astronautics, Washington D.C., 1989.

12.5 Eichblatt, Emil J. Jr., ed., Test and Evaluation of the Tactical Missile, Volume 119 of Progress in Astronautics and Aeronautics, American Institute of Aeronautics and Astronautics, Washington D.C., 1989.

13. LIFE CYCLE COST ANALYSIS

The purpose of this chapter is to present the results of the cost analysis of the Monarch fighter program.

The methodology of Chapters 3, 4, 6 and 7 of Reference 13.1 were used to determine the following costs for the Monarch fighter program:

13.1 Research, Development, Test and Evaluation Cost

13.2 Acquisition Cost

13.3 Operating Cost

13.4 Life Cycle Cost

The life cycle cost (LCC) is made up of the research, development, test and evaluation (RDTE) cost, the program acquisition cost, the program operating cost and the disposal cost. These four components of the life cycle cost are incurred during the six phases of the aircraft life cycle as shown in Figure 13.1. This figure also illustrates the percentage of the life cycle cost that is locked in during each phase of the aircraft life. The Monarch is currently through Phase 1 and Phase 2 of the aircraft life cycle. This indicates that 85% of the life cycle cost of the Monarch fighter program is locked in. By being aware of the implications of the data presented in Figure 13.1, the design team has insight into the influence that decisions made early in the design process have on the life cycle cost of the aircraft.

It is assumed that the Monarch fighter will be in operation for 25 years beginning in the year 2005. A baseline production run of 500 airplanes and an annual utilization of 325 flight hours per airplane per year are assumed. The costs are based on anticipated 2005 rates and US dollar value.

13.1 RESEARCH, DEVELOPMENT, TEST AND EVALUATION COST

The following values were required for the determination of the RDTE cost of the Monarch fighter program:

Takeoff weight = 31,336 lbs
Maximum Velocity = 794 keas
Number of Airplanes Built for the RDTE Phase = 10
Difficulty Factor for the Monarch fighter program = 2.0
(2.0 assumes an aggressive use of new technology)
CAD Experience Factor = 0.8
(0.8 assumes CAD experience)
Engineering Manhour Rate = $105.00
(reflects a 50% increase over non-security rate)
Cost Escalation Factor = 3.1
(estimated for 2005)

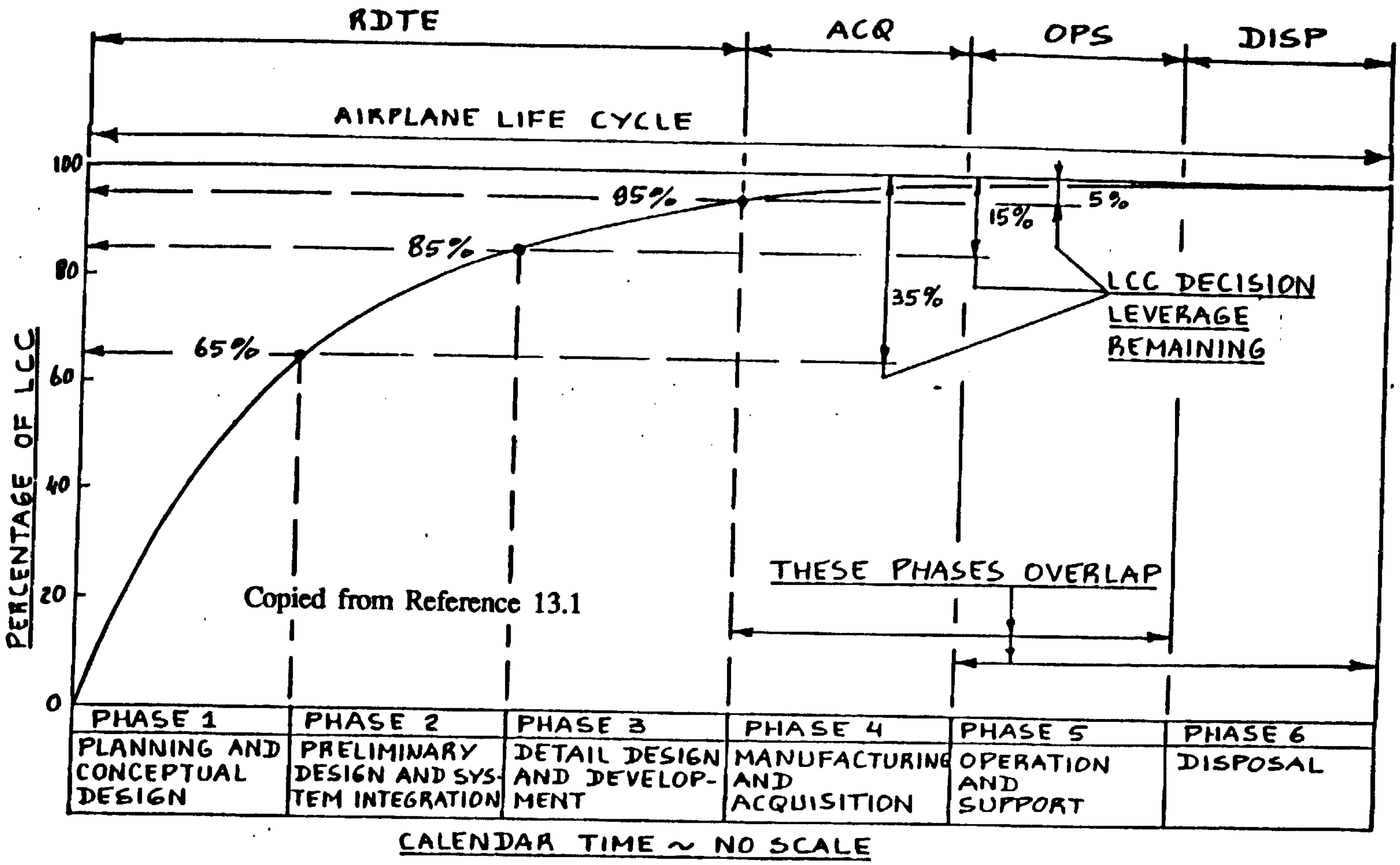


Figure 13.1 Impact of Airplane Program Phases on Life Cycle Cost

Number of Conventional Engines Per Airplane = 1
Number of Static Test Airplanes = 2
Main Engine Takeoff Thrust = 35,573 lbs
STOVL Equipment:
    Lift Engine = \$1.027 million
    Lift Engine Nozzle = \$154,080
    1-D Vectoring Ventral Nozzles = \$222,400
    2-D Vectoring Main Nozzle = \$444,800
Material Correction Factor = 2.5
    (2.5 assumes construction with conventional composite materials)
RDTE Production Rate = .35 airplanes/month
Manufacturing Manhour Rate = \$68.00
    (reflects a 50% increase over non-security rate)
Tooling Manhour Rate = \$83.00
    (reflects a 50% increase over non-security rate)
Stealth Factor = 1.0
    (1.0 assumes no designed stealth features)
Test Facilities Cost Adjustment Factor = 0.2
    (0.2 assumes extensive test facilities are required)
Percentage of Profit on the RDTE Phase = 10%
Financing Cost Factor = 0.13
    (0.13 assumes a 13% interest rate on the financing)

The value for the STOVL lift engine was determined by increasing the value of a conventional engine of equal thrust (12,105 lbs) by 20%. This increase was due to the advanced technology required to produce this engine. The lift engine is estimated to weigh 480 pounds, thus resulting in an installed thrust-to-weight ratio of 25. The lift engine nozzle was estimated to be 15% of the lift engine value. The ventral nozzles were estimated to be 10% of the main engine value. The 2-D main nozzle was estimated to be 20% of the main engine value. These estimated values were obtained from Reference 53.

The total RDTE cost for the Monarch fighter program was determined to be 3.716 billion dollars.

13.2 ACQUISITION COST

The acquisition cost is the cost that the government or taxpayers pay for the total number of airplanes in the program. The acquisition cost is the sum of the manufacturing cost and the profit made by the manufacturer. The following values were used as input in determining the acquisition cost of the Monarch fighter program:

Number of Airplanes Built to Production = 500
Manufacturing Rate of Production Airplanes = 10/month
Airplane Operating Cost Per Flight Hour = \$10,146/hr
    (from Section 13.3)
Test Flight Hours before Delivery = 20
Overhead Factor = 4.0
Manufacturing Finance Factor = 0.13
Manufacturing Profit = 0.1

The acquisition cost for the Monarch fighter program is determined to be 12.206 billion dollars. Figure 13.2 illustrates the effect of the number of airplanes produced on the total program acquisition cost.

The average estimated price (AEP) per fighter is determined by summing the RDTE and the acquisition costs and dividing by the number of airplanes produced. With 500 airplanes produced, the average estimated cost of the Monarch fighter is 32.6 million dollars. Figure 13.3 illustrates the AEP of the Monarch fighter as a function of the number of airplanes produced. Note that for production runs greater than about 600 airplanes, no significant decrease in AEP is experienced for a reasonable increase in the number of airplanes produced.

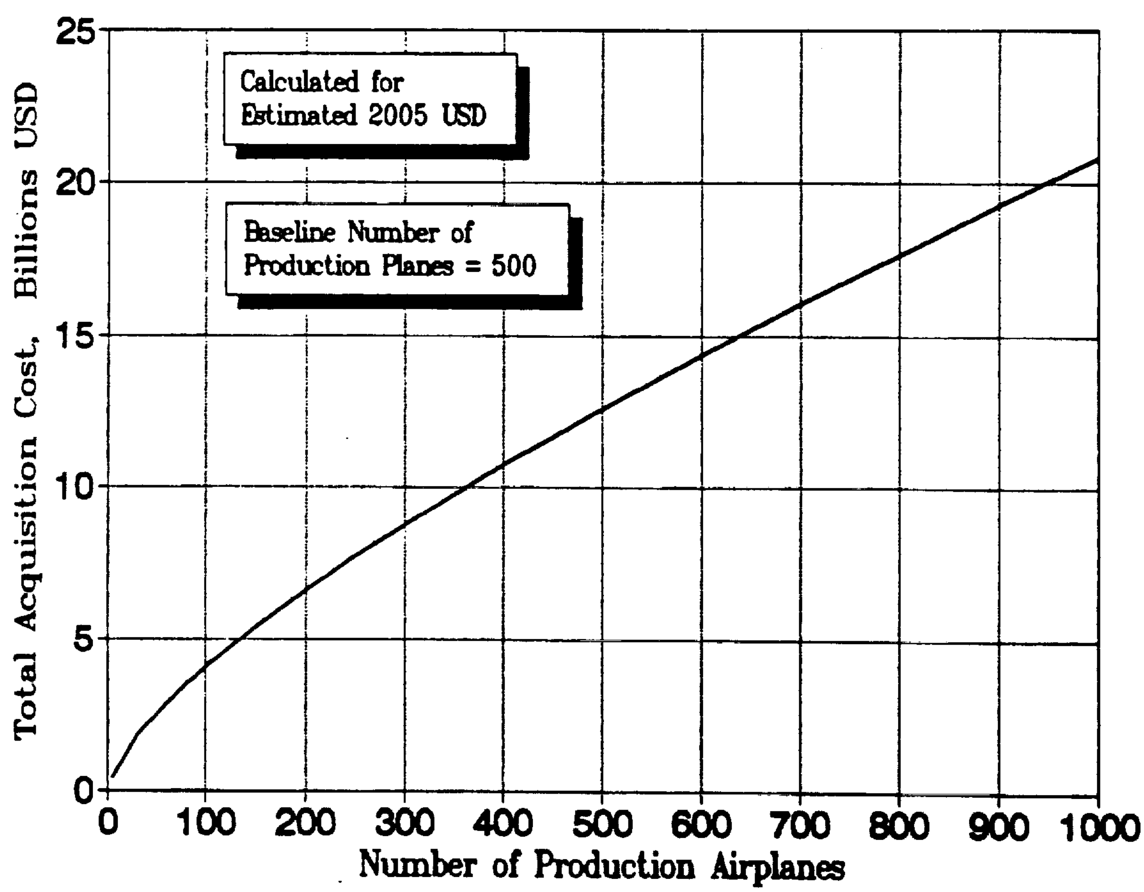


Figure 13.2 Effect of Number of Airplanes Produced on the Acquisition Cost for the Monarch Fighter Program

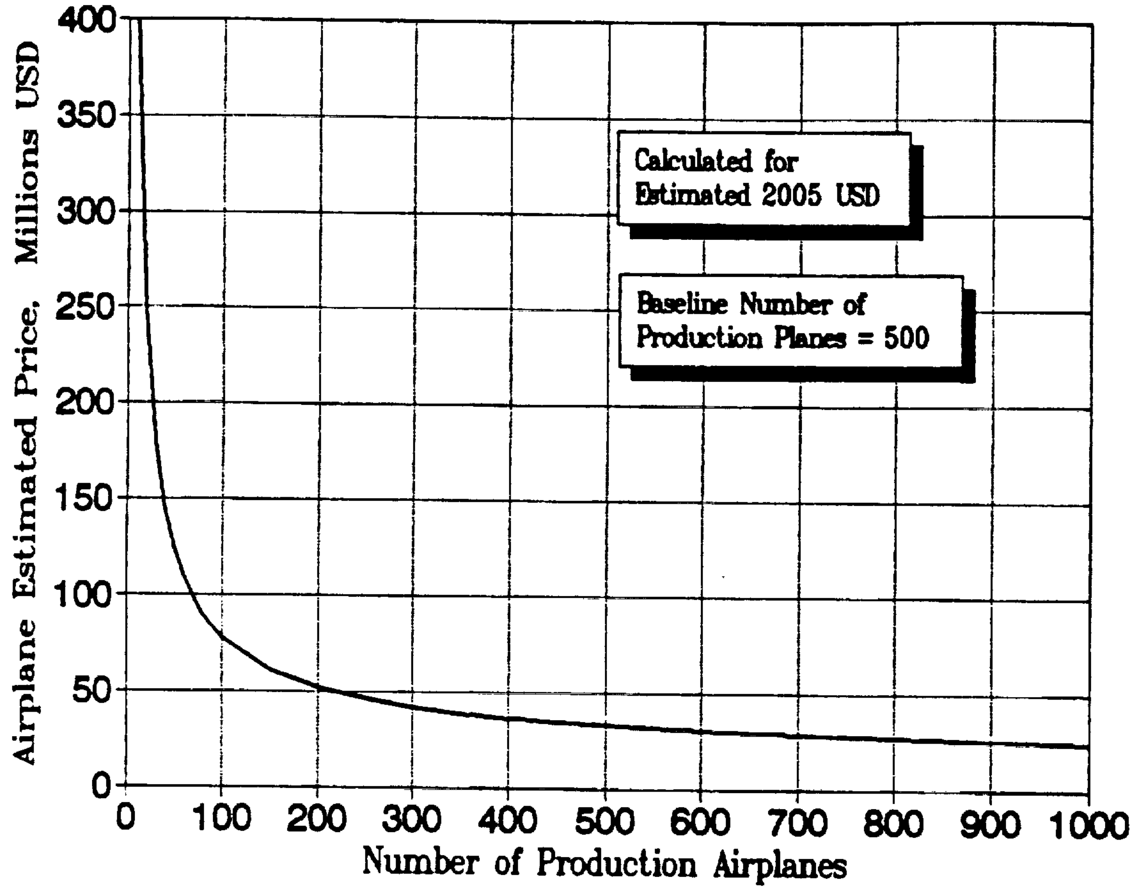


Figure 13.3 Effect of Number of Airplanes Produced on the Average Estimated Price of the Monarch Fighter

13.3 OPERATING COST

The program operating cost is the total amount of money that will be required to operate a specified number of airplanes, flying a certain number of hours per year for a set number of years.

The following values were used to determine the total operating cost of the Monarch fighter program:

Mission Fuel Weight = 8642 lbs
Fuel Price = $1.95/gal (estimated 2005 price)
Fuel Density = 6.55 lbs/gal (JP-4)
Annual Utilization = 325 flight hours
Average Mission Time = .80 hrs (air-to-air mission)
Number of Airplanes Built to Production = 500
Annual Loss Rate Per 100,000 Flight Hours = 7
Number of Years in Active Duty = 25
Number of Crew Members = 1
Crew Ratio Per Airplane = 1.1

Aircrew Basic Pay = \$34,000
Aircrew Incentive Pay = \$500/month
Aircrew Re-up Bonus = \$14,000
Maintenance Man Hours Per Flight Hour = 15
Cost Escalation Factor = 3.1
Airplanes Used by the Reserves Factor = 0.10
Indirect Personnel Cost Factor = 0.2
Spare Part Cost Factor = 0.18
Depot Cost Factor = 0.16

The total operating cost for the Monarch fighter program is determined to be 28.88 billion dollars.

The operating cost per flight hour is determined by dividing the total program operating cost by the number of airplanes in service, the number of years the airplane is in active service and the number of hours each airplane is flown annually.

Therefore the operating cost per flight hour for the Monarch fighter is determined to be \$10,146. Figure 13.4 illustrates the effect of the number of airplanes produced on the operating cost per flight hour for the Monarch. Note that when conducting this trade study, the cost of program indirect personnel, consumable materials, depot maintenance and miscellaneous accruals are held at the baseline production number values. For the baseline production of 500 airplanes these costs have the following values:

Indirect Personnel = 5.776 billion dollars
Consumable Materials = 286 million dollars
Depot Maintenance = 4.621 billion dollars
Miscellaneous Accruals = 1.144 billion dollars

13.4 LIFE CYCLE COST

The life cycle cost (LCC) represents the total amount of money spent on the airplane program. The life cycle cost is broken down into the following components:

1. Research, development, test and evaluation cost
2. Acquisition cost
3. Program operating cost
4. Disposal cost of the airplanes

Values for items 1 through 3 have been computed.

No accurate method exists for determining the cost of disposal. Reference 13.1 suggests that the disposal cost is 1% of the program life cycle cost. Figure 13.5 shows the breakdown of life cycle cost for the Monarch fighter program.

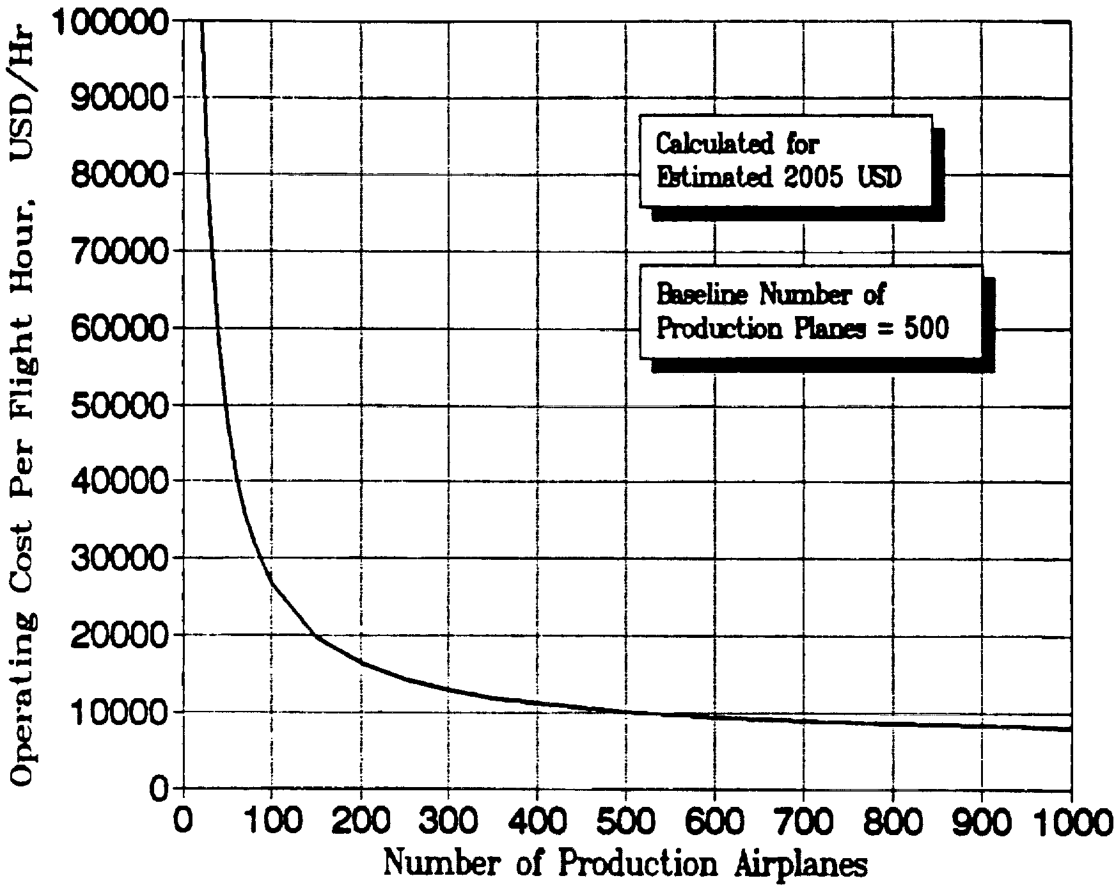


Figure 13.4 Effect of Number of Airplanes Produced on the Operating Cost Per Flight Hour for the Monarch Fighter

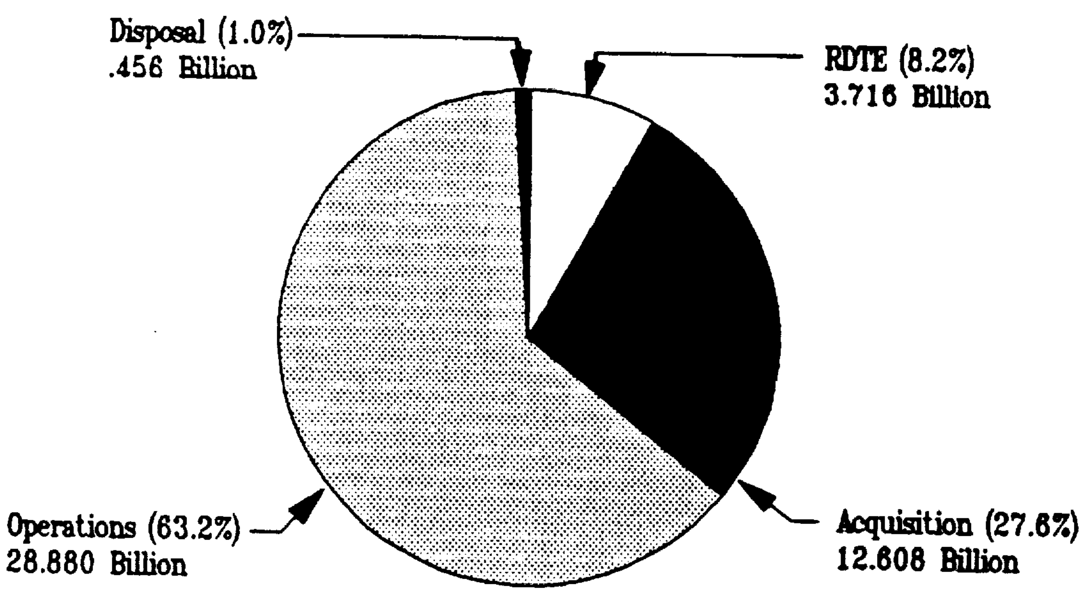


Figure 13.5 Life Cycle Cost Breakdown of the Monarch Fighter Program

REFERENCES FOR CHAPTER 13

13.1 Roskam, Jan, Airplane Design: Part VIII, Airplane Cost Estimation: Design, Development, Manufacturing and Operating, Roskam Aviation and Engineering Corporation, Ottawa, KS, 1989.

13.2 Personal Consultation, Dr. S. Farokhi, University of Kansas, May 1, 1990.

14. CONCLUSIONS AND RECOMMENDATIONS

14.1 CONCLUSIONS

1) The results of the Phase I study of three supersonic STOVL concepts indicate that the configuration with the lift+lift/cruise engine cycle suffered the least penalties for the short takeoff and vertical landing capability and was the concept with the most readily available technology for a Technology Availability Date of 1995.

2) Based on the Phase I study results, the lift+lift/cruise configuration was selected for Phase II design work. This work consisted of more detailed configuration design and concentrated on some of the STOVL aspects of the aircraft.

3) The aircraft has a wide, flat fuselage sections due to the requirements for:
   * the lift engine,
   * the internal weapons bay,
   * the large landing gear tire sizes,
   * the ventral nozzles for the cruise engine, and
   * shaping considerations for a favorable area rule distribution.

4) The aircraft weights were estimated using empirical equations based on statistical data and actual weights taken from operational fighters. The aircraft is balanced in hover and has acceptable inflight center of gravity travel.

5) The aircraft achieves powered lift with a lift engine and two ventral nozzles on the main engine. Three posts allowed the in ground effect suckdown to be reduced to 10 percent (this represents a 15 percent reduction when compared to the Phase I lift+lift/cruise configuration which had a two posts).

6) The thrust vectoring capabilities of the cruise engine nozzle allow for:
   * enhanced maneuvering at high angles of attack,
   * removal of the rudder,
   * and reduction in the vertical tail size.

   The reduced vertical tail size aided the area rule distribution of the aircraft to very favorably match that of the Sears-Haack shape.

7) The aircraft, without rotating for hover, can lift off in 238 ft for the design counter air mission and in less than 800 ft with an overload mission (the overload mission consists of 5,000 lbs of more ordnance than the counter air mission).

8) The aircraft has a high level of performance throughout its flight envelope and compares favorably to the operational aircraft of the United States and Soviet Union. The aircraft has growth potential in that it can perform typical NATO missions with acceptable range and ordnance carrying capability. Being a STOVL configured, the aircraft also has the ability to perform unconventional two stage missions, possibly allowing for increased sorties per day.

9) The aircraft can be trimmed at all investigated flight conditions. For three flight conditions, a digital stability augmentation system was employed for the longitudinal, lateral, and directional axis to meet Level 1 handling qualities. The augmented aircraft is not prone to inertia coupling at the three flight conditions investigated.

10) An aircraft structural layout was completed with structural synergism as a key priority. The materials for the aircraft were selected considering their resulting weight in an aircraft application, their cost and durability, and their ease of repair in a battlefield scenario. The manufacturing process and breakdown of the aircraft was preliminarily investigated along with its the accessibility and maintainability considerations.

11) The system layout for the aircraft was complete and all system conflicts were eliminated by using a combination of three view and ghost view system layouts. The avionics and electronic counter measures were selected for the aircraft.

12) The medium range missiles were successfully mounted inside the aircraft. The integration and launch mechanism of the internal missiles was verified by building a $1/5^{th}$ scaled model of the layout. The short range missiles are wing tip mounted and the battlefield air interdiction mission ordnance are carried on wing pylons.

4.2 RECOMMENDATIONS

1) The aircraft cannot meet the Level 1 specification for roll performance in all flight conditions with only aileron deflection. Two approaches should be investigated to achieve the required roll performance. First, the effect on the roll performance of the deflection of the leading edge flap should be calculated. Leading edge deflection may also aid in reducing unfavorable aeroelastic effects. Second, the effect on the roll performance of the all moving stabilator should be calculated.

2) Hot Gas Reingestion (HGR) may be a problem during hover close to the ground. Since HGR is very configuration dependent and is difficult to predict, wind tunnel tests should be performed. Nevertheless, possible solutions were investigated to alleviate the HGR that can be implemented without major design changes to the aircraft. The possible ground erosion due to the jet exhaust of the lift and main engine should also be investigated with experimental techniques.

3) The effect of the cannon firing creating adverse yawing moments needs to be studied further to determine if rudder feedback is required. Also, the effect of cannon vibrations in the airframe should be studied to determine if aircraft sensors need to be isolated.

4) The bending and torsion of the internal weapons bay doors while deflected at a high dynamic pressure need to be investigated. The actuators for the doors also need to be sized to effectively open the door at high dynamic pressures.

5) The aircraft will require a ride quality augmentation system to allow the aircraft to successfully perform low level ground attack missions.

C-4

APPENDIX 1

The purpose of this appendix is to document the weight and balance calculations presented in Chapter 5. The weight and balance was done using a spreadsheet. Appendix 1.1 defines the symbols and summarizes the equations used for the spreadsheet. Appendix 1.2 shows the spreadsheet used to calculate the weight and balance.

Appendix 1: Table of Contents

CLASS II WEIGHT SPREADSHEET SYMBOLS

Symbol	Parameter	Unit
S_w	Wing area	ft^2
A_w	Wing aspect ratio	----
LM_LE_w	Wing sweep angle	deg
Lm_w	Wing taper ratio	----
(t/c)_m_w	Wing thickness ratio max	----
cbar_w	Wing mgc	ft
l_f	Length of fuselage	ft
h_f	Height of fuselage	ft
S_h	Horizontal tail area	ft^2
b_h	Horizontal tail span	ft
cbar_h	Horizontal mgc	ft
l_h	Distance from wing c/4 to tail $c_h/4$	ft
t_r_h	Horizontal tail max root thickness	ft
z_h	Distance from the vertical tail root to where the horizontal tail is mounted on the fuselage. fuselage mounted horizontal tails, z_h = 0.	ft
A_v	Vertical tail aspect ratio	----
b_v	Vertical tail span	ft
S_v	Vertical tail area	ft^2
l_v	Distance from wing c/4 vert. tail $c_v/4$	ft
Lm_v	Vertical tail taper ratio	----
LM_c/4_v	Vertical tail sweep angle c/4	deg
S_r	Rudder area	ft^2
S_c	Canard area	ft^2
b_c	Canard span	ft
t_r_c	Canard max root thickness	ft

cbar_c	Canard mgc	ft
l_c	Distance from wing c/4 to canard $c_c/4$	ft
W_TO	Takeoff Weight	lbs
W_E	Empty Weight	lbs
W_F	Fuel Weight	lbs
W_e1	Engine 1 weight	lbs
W_e2	Engine_2 weight	lbs
W_crew	Crew weight	lbs
GW	Gross design weight	lbs
W_wtr	Weight of water for injection	lbs
W_iae	Weight of instruments and avionics	lbs
W_glw	Weight of gun and launcher	lbs
W_ets	Miscellaneous weight	lbs
K_fcf	Fixed Equipment - flight control sys	----
K_ec	Power Plant - engine controls	----
K_g_r	Structural - landing gear - main wing	----
K_w	Structural - wing	----
K_inl	Structural - fuselage	----
K_d	Power Plant - air induction system	----
K_m	Power Plant - air induction system	----
K_fsp	Power Plant - fuel sys - self sealing	lbs/gal
N_inl	Power Plant - air induction system	----
N_e	Power Plant - engine controls	----
N_cr	Number of crew	----
n_ult	Ultimate load in g's	----
M_H	Maximum Mach speed at sea level	----
qbar_D	Design dive dynamic pressure	lb/ft^2
L_d	Power Plant - air induction system	ft
A_inl	Power Plant - air induction system	ft^2
P_2	Power Plant - air induction system	psi
W_mg/W_g	Weight of main gear to nose gear	----
M_ff	Mission fuel fraction	----
GW/T_TO	Ratio of GW to takeoff weight	----
Weapons	Put weight of weapons in spreadsheet	lbs

A_ng
B_ng
C_ng
D_ng

PART V CONSTANTS FOR LANDING GEAR PAGE 82
NO UNITS

A_mg
B_mg
C_mg
D_mg

WEIGHT SPREADSHEET GUIDE

This guide has equations developed into a spreadsheet for determining class II weights for supersonic fighter attack aircraft.

Procedure for spreadsheet weight analysis:

1.) Follow equations in this guide and enter parameters for particular airplanes at the TOP of the spreadsheet ONLY!

2.) Page down in spreadsheet to see how weights have changed.

3.) Split Screen and iterate takeoff weight

4.) Enter F.S., B.L., W.L., in lower portion of spreadsheet. weights will have automatically transferred.

NOTES: Enter all areas in ft.
Enter all angles in degrees.
Enter all weights in lbs.

Enter GW/W_{TO} fraction in TOP of spreadsheet

Enter W_{mg}/W_g at TOP of spreadsheet

Enter M_H at TOP of spreadsheet

Enter W_{PL} at TOP of spreadsheet

Enter landing gear option (1 or 2) at TOP of spreadsheet

Option 1: General gear equation

Option 2: Main gear on wing

STRUCTURAL WEIGHT (lbs)

$W_{struct} = W_w + W_{emp} + W_f + W_g$

(5.9) <u>WING</u> (W_w)

$$W_w = 3.08[\{(K_w n_{ult} GW)/(t/c)_m\}\{(\tan\Lambda_{LE} - 2(1-\lambda)/A(1+\lambda))^2 +$$
$$+ 1.0\}X10^{-6}]^{.593}\{A(1+\lambda)\}^{0.89}(S)^{0.741}$$

<u>Notes</u>:

K_w = 1.00 for fixed wing

$_{LE}$ = leading edge sweep angle of the wing

GW = Gross Design Weight (lbs)

$(t/c)_m$ = maximum thickness ratio

n_{ult} = ultimate load factor in g's

<u>EMPENNAGE</u> ($W_{emp} = W_h + W_v + W_c$) (lbs)

(5.17) <u>HORIZONTAL TAIL</u> (W_h) (lbs)

$$W_h = .0034\{(W_{TO}n_{ult})^{0.813}(S_h)^{0.584}(b_h/t_{r\_h})^{0.033}(c/l_h)^{0.28}\}^{0.915}$$

(5.18) <u>VERTICAL TAIL</u> (W_v) (lbs)

$$W_v = 0.19\{(1 + z_h/b_v)^{0.5}(W_{TO}n_{ult})^{0.363}(S_v)^{1.089}(M_H)^{.601} \ x$$
$$x \ (l_v)^{-0.726}(1 + S_r/S_v)^{0.217}(A_v)^{0.337}(1+\lambda_v)^{.363}(\cos\Lambda_{c/4\_v})^{-0.484}\}^{1.014})$$

<u>Notes</u>:

z_h = distance from the vertical tail root to where the horizontal tail is mounted on the vertical tail, in ft. <u>Warning</u>: for fuselage mounted horizontal tails, z_h = 0 in ft

l_v = distance from c/4 to vert. tail $c_v/4$ in ft.

S_r = rudder area ft^2

$_v$ = vertical tail taper ratio

(5.17) <u>CANARD</u> (W_c) (lbs)

$$W_c = .0034\{(W_{TO}n_{ult})^{0.813}(S_c)^{0.584}(b_c/t_{r\_c})^{0.033}(c/l_c)^{0.28}\}^{0.915}$$

(5.26) FUSELAGE (W_f) (lbs)

$$W_f = 10.43(K_{inl})^{1.42}(q_D/100)^{0.283}(W_{TO}/1000)^{0.95}(l_f/h_f)^{0.71}$$

Notes:

K_{inl} = 1.25 for airplanes with inlets in or on the fuselage for a buried engine installation.

K_{inl} = 1.0 for inlets located elsewhere

q_D = design dive dynamic pressure in lbs/ft^2

l_f = length of fuselage in ft.

h_f = height of fuselage in ft.

(5.41) LANDING GEAR GENERAL (W_g) (lbs)

$$W_g = 62.61(W_{TO}/1000)^{0.84}$$

(5.42) LANDING GEAR - MAIN ON WING AND NOSE ON FUSELAGE (W_g)

$$W_g = K_{g\_r}(A_g + B_g(W_{TO})^{3/4} + C_g W_{TO} + D_g(W_{TO})^{3/2}) \text{ (lbs)}$$

Notes:

K_{g_r} = 1.0 for low wing

K_{g_r} = 1.08 for high wing

Constants A through D are in spreadsheet

POWER PLANT (lbs)

$W_{pwr} = W_e + W_{ai} + W_{fs} + W_{fd} + W_p$ (lbs)

ENGINE (W_e) (lbs)

Actual weight of specific engines

Notes:

This includes: engine, exhaust, cooling, lubrication.

(6.9) AIR INDUCTION SYSTEM (W_{ai}) (lbs)

$W_{ai} = 0.32(N_{inl})(L_d)(A_{inl})^{0.65}(P_2)^{0.6} +$

(duct support structure)

$+ 1.735((L_d)(N_{inl})(A_{inl})^{0.5}(P_2)(K_d)(K_m))^{0.7331}$

(subsonic part of duct)

Notes:

K_d = 1.33 for ducts with flat cross sections

K_d = 1.0 for ducts with curved cross sections

K_m = 1.0 for M_D below 1.4

K_m = 1.5 for M_D above 1.4

L_d = duct length in ft.

N_{inl} = number of inlets

A_{inl} = capture area per inlet in ft^2

P_2 = maximum static pressure at engine compressor face in psi. Typical values: 15 to 50 psi.

(6.20) FUEL SYSTEM - SELF SEALING BLADDER CELLS (W_{fs}) (lbs)

$$W_{fs} = 41.6((W_F/K_{fsp})/100)^{0.818} + W_{supp}$$

$$W_{supp} = 7.91((W_F/K_{fsp})/100)^{0.854}$$

Notes:

K_{fsp} = 6.55 lbs/gal for JP-4

(6.26) FUEL DUMPING (W_{fd}) (lbs)

$$W_{fd} = 7.38((W_F/K_{fsp})/100)^{0.458}$$

PROPULSION SYSTEM ($W_p = W_{ec} + W_{ess} + W_{wi}$) (lbs)

(6.23) Engine Controls (W_{ec}) (lbs)

fuselage/wing-root mounted jet engines

$$W_{ec} = K_{ec}(l_f N_e)^{0.792}$$

Notes:

K_{ec} = 0.686 for non-afterburning

K_{ec} = 1.080 for afterburning

N_e = number of engines

l_f = fuselage length in ft.

b = wing span in ft

(6.29) Electric Starting System (W_{ess}) (lbs)

$$W_{ess} = 38.93(W_e/1000)^{0.918}$$

Notes:

W_e = total weight of all engines in lbs

(6.37) Water Injection (W_{wi}) (lbs)

$$W_{wi} = 8.586W_{wtr}/8.35$$

Notes:

W_{wtr} = weight of water carried in lbs

FIXED EQUIPMENT WEIGHT (lbs)

$W_{feq} = W_{fc} + W_{els} + W_{iae} + W_{api} + W_{ox} + W_{apu} + W_{fur} + W_{arm} + W_{glw} +$
$+ W_{aux} + W_{pt} + W_{etc}$ (lbs)

(7.9) FLIGHT CONTROL SYSTEMS (W_{fc}) (lbs)

$$W_{fc} = K_{fcf}(W_{TO}/1000)^{0.581}$$

Notes:

K_{fcf} = 106 for airplanes with elevon control and no horizontal tail

K_{fcf} = 138 for airplanes with a horizontal tail

K_{fcf} = 168 for airplanes with a variable sweep wing

(7.19) ELECTRICAL SYSTEMS (W_{els}) (lbs)

$$W_{els} = 426\{(W_{fs} + W_{iae})/1000\}^{0.51}$$

AVIONICS/INSTRUMENTATION/ELECTRONICS (W_{iae}) (lbs)

Actual Data or Appendix A.

(7.33) AIR/ICE/DE-ICE (W_{api}) (lbs)

$$W_{api} = 202\{(W_{iae} + 200N_{cr})/1000\}^{0.735}$$

(7.39) OXYGEN SYSTEM (W_{ox}) (lbs)

$$W_{ox} = 16.9(N_{cr})^{1.494}$$

(7.40) APU (W_{apu}) (lbs)

$$W_{apu} = (0.004 \text{ to } 0.013)W_{TO}$$

(7.47) FURNISHINGS (W_{fur}) (lbs)

$$W_{fur} = 22.9(N_{cr}q_D/100)^{0.743} + 107(N_{cr}W_{TO}/100{,}000)^{0.585}$$

(ejection seats) (misc. emergency equip.)

ARMAMENT (W_{arm}) (lbs)

ACTUAL DATA - APPENDIX A

GUNS, LAUNCHERS, WEAPONS, (W_{glw}) (lbs)

ACTUAL DATA - APPENDIX A

(7.50) AUXILARILY GEAR (W_{aux}) (lbs)

$W_{aux} = 0.01W_E$

(7.51) PAINT ESTIMATE (W_{pt}) (lbs)

$W_{pt} = 0.0045_{TO}$

MISCELLANEOUS WEIGHT (W_{etc}) (lbs)

W_{etc} = actual weight data

AE 622 LIFT STOVL DESIGN

LAST REVISED: Sunday March 10 1990
REVISED BY: Brian Cox

CLASS II COMPONENT WEIGHTS

-----AIRPLANE GEOMETRY-----				---WEIGHTS---		---------CONSTANTS---------			
S_w	322	A_v	1.61	W_TO	31336	K_fcf	138	L_d	17
A_w	3.5	b_v	10.9	W_E	21415	K_ec	1.08	A_inl	3.4
LM_LE_w	37.8	S_v	43	W_F	8642	K_g_r	1	P_2(psi)	30
Lm_w	0.19	l_v	17	W_e1	3557	K_w	1		
(t/c)_m_w	0.045	Lm_v	0.35	W_e2	480	K_inl	1.25		
cbar_w	11.06	LM_c/4_v	38	W_Crew	225	K_d	1	W_TO.old	
b_w	33.57	S_r	0	GW	21935	K_m	1.5	31336	
l_f	56			W_wtr	0	K_fsp	6.55		
h_f	6	S_h	40	W_iae	1517			W_E	
		b_h	11.4	M_ff=	0.738	N_inl	2	21117	
W_mnzl	420	t_r_h	0.2	W_glw	630	N_e	2		
W_vntv	300	cbar_h	3.75			N_cr	1		
W_tpipe	300	l_h	16	GW/W_TO	0.7	n_ult	13.5		
W_lmas	40			W_mg/W_g	0.85	M_H	1.2		
W_lmam	262			W_payload	1196	qbar_D	2133		

W_E = W_struct + W_pwr + W_feq (2.1)

STRUCTURAL WEIGHT

W_struct = W_w + W_emp + W_f + W_g + W_vntv

Wing Weight (5.9)

W_w = 2490

Empennage Weight

Vertical Tail (5.18)

W_v = 256

Horizontal Tail (5.17)

W_c = 295

W_emp= 551

Fuselage (5.26)

W_f = 4385

Landing Gear

W_g = 1131

Launch Mechanism

W_lmas = 40
W_lmam = 262

Ventral Nozzle W_vntv= 300

Therefore,

W_struct= 9159
=================

POWER PLANT WEIGHT

W_pwr = W_e + W_ai + W_fs + W_fd + W_p + W_mnzl

Engine Weight

W_e = 4037

Air Induction System (6.9)

W_ai = 773

Fuel System Bladder (6.20)

W_fs = 415

Fuel Dumping (6.26)

W_fd = 24

Propulsion System

W_p = W_ec + W_ess + W_wi

Engine Controls (6.23)

W_ec = 45.3

Engine Start-Up (6.29)

W_ess= 125

Water Injection (6.37)

W_wi = 0

Therefore,

W_p = 170

Main Nozzle Weight W_mnzl = 420

Main Engine Tailpipe section = 300

Therefore,

W_pwr = 6139
=================

FIXED EQUIPMENT WEIGHT

W_feq = W_fc + W_iae + W_els + W_api + W_ox + W_apu + W_fur
+ W_glw + W_aux + W_pt + W_rcsd + W_rcsn + W_rcsc

Flight Control Sys (7.9)

W_fc = 1021

Avionics (Actual Data)

W_iae= 1517

Electrical Systems (7.19)

W_els= 596

Air cond./press./anti- and De-Ice (7.33)

W_api= 301

Oxygen System (7.39)

W_ox = 17

APU (7.40)

W_apu= 298

Furnishings (7.47)

W_fur= 277

Gun and Launch Provisions (Actual Data)

W_glw= 630

AUX Gear (7.50)

W_aux= 214

Paint Est. (7.51)

W_pt= 204

RCS Duct Weight(WRDC) RCS Nozzle Weight(WRDC)

W_rcsd= 287 W_rcsn= 83

RCS Controls Weight(WRDC)

W_rcsc=   35

Therefore,

W_feq =   5480
=================

====================
¦ W_E =   20777   ¦
====================

CLASS II WEIGHT AND BALANCE

COMPONENT	FACTOR	WEIGHT	x	Wx	y	Wy	z	Wz
Fuselage	1	4385	420	1841719	0	0	160	701607
Wing	1	2490	490	1220281	0	0	160	398459
Vert Tail	1	256	590	151273	0	0	235	60253
Hort Tail	1	295	640	188758	0	0	160	47190
Main Gear	1.3	1249	505	630919	0	0	138	172410
Nose Gear	1.3	220	165	36378	10	2205	135	29764
Launch Mech								
A ASRAAM		40	530	21200	0	0	160	6400
A AMRAAM		261.6	330	86328	0	0	135	35316
Vent Nozzles	1	300	578	173400	0	0	138	41400
Struct Wt	1	9498	x_cg=	458	y_cg=	0	z_cg=	157
Engine #1	1	3557	530	1885210	0	0	165	586905
Engine #2	1	480	230	110400	0	0	142	68160
#1 Tailpipe	1	300	578	173400	0	0	165	49500
#1 Nozzle	1	420	660	277200	0	0	165	69300
Air Induct	1	773	380	293642	0	0	165	127502
Fuel Bladder	1	415	390	161782	0	0	160	66372
Fuel Dumping	1	24	550	13230	0	0	140	3368
Eng Controls	1	45	525	23799	10	453	150	6800
Eng Start-Up	1	125	550	68634	0	0	160	19966
Water Inject	1	0	550	0	0	0	0	0
Prpl. Wt	1	6139	x_cg=	490	y_cg=	0	z_cg=	163

Flght Cntrl	1	1021	470	479930	0	0	160	163381
Avionics	1	1517	440	667480	0	0	140	212380
Elect Sys	1	596	400	238404	0	0	160	95362
Air/de-ice	1	301	460	138250	0	0	160	48087
Oxygen Sys	1	17	155	2620	0	0	150	2535
APU	1	298	600	178615	25	7442	150	44654
Furnishings	1	277	125	34594	0	0	160	44280
Gun, Prov.	1	630	330	207900	-35	-22050	140	88200
AUX Gear	1	214	200	42830	0	0	155	33193
Paint	1	204	420	85547	0	0	160	32589
RCS Duct	1	287	475	136493	0	0	155	44540
RCS Nozzle	1	83	475	39545	0	0	155	12904
RCS Controls	1	35	475	16708	0	0	155	5452
Fix Equip Wt	1	5480	x_cg=	414	y_cg=	-3	z_cg=	151
Empty Weight		21117	x_cg=	456	y_cg=	-1	z_cg=	157
Pilot		225	190	42750	0	0	170	38250
W_tfo		157	450	70506	0	0	160	25069
Oper Empty Weight		21498	x_cg=	453	y_cg=	-1	z_cg=	157
		1						
Hover Fuel (20%)		1728	410	708655	0	0	160	276548
Fuel		8642	390	3370434	0	0	165	1425953
W_owe + W_fuel =		30140	x_cg=	435	y_cg=	-0	z_cg=	160
COUNTER AIR MISSION								
ASRAAM #1		161	530	85330	205	33005	160	25760
ASRAAM #2		161	530	85330	-205	-33005	160	25760
AMRAAM #1		327	330	107910	-15	-4905	135	44145
AMRAAM #2		327	330	107910	15	4905	135	44145
AMMO - 400RDS		220	350	77000	0	0	160	35200

BATTLEFIELD AIR INTERDICTION

BAI #1

Mk-82 #1	560	470	263200	101.5	56840	130	72800
Mk-82 #2	560	470	263200	122.5	68600	130	72800
Mk-82 #3	560	470	263200	-101.5	-56840	130	72800
Mk-82 #4	560	470	263200	-122.5	-68600	130	72800
HARM #1	807	390	314730	62	50034	130	104910
HARM #2	807	390	314730	-62	-50034	130	104910
AMMO - 400RDS	220	350	77000	0	0	160	35200
EJECTOR RACKS	186	430	79980	0	0	130	24180

BAI #2

Mk-82 #1	560	470	263200	112	62720	130	72800
Mk-82 #2	560	470	263200	-112	-62720	130	72800
Maverick #1	494	390	192660	62	30628	130	64220
Maverick #2	494	390	192660	77	38038	130	64220
Maverick #3	494	390	192660	-62	-30628	130	64220
Maverick #4	494	390	192660	-77	-38038	130	64220
AMMO - 400RDS	220	350	77000	0	0	160	35200
EJECTOR RACKS	186	430	79980	0	0	130	24180

==

CA W_TO	31336	x_cg= 433	y_cg=	-0	z_cg=	159
BAI #1 W_TO	34400	x_cg= 435	y_cg=	-0	z_cg=	156
BAI #2 W_TO	33642	x_cg= 433	y_cg=	-0	z_cg=	157
CA W_HOV	24423	x_cg= 447	y_cg=	-0	z_cg=	157
BAI #1 W_HOV	24382	x_cg= 446	y_cg=	-0	z_cg=	155
BAI #2 W_HOV	24744	x_cg= 445	y_cg=	-0	z_cg=	155

APPENDIX 2

The purpose of this appendix is to show the spreadsheets used to calculate the mission performance data and mission capability as discussed in Chapter 8.

Appendix 2: Table of Contents

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Brian Cox
Last Revised: 12 March 1990

POINT PERFORMANCES:

LANDING DISTANCE, SL = 4300 FT

TIME TO CLIMB:

ALTITUDE	40000	FT
TIME	1.75	MINUTES
ABSOLUTE CEILING	80000	FT

SPECIFIC EXCESS ENERGY:

H (FT)	a (FT/S)	RHO	MACH	PS (FT/S)
30000	994.70	0.000889	0.90	505
10000	1077.40	0.001755	0.90	920

MANEUVERING:

SUSTAINED TURN RATE

H (FT)	a (FT/S)	RHO	MACH	TURN RATE (DEG/S)
15000	1057.30	0.001496	0.80	15.00
30000	994.70	0.000889	0.90	10.00
30000	994.70	0.000889	1.20	9.90

OTHER

H (FT)	a (FT/S)	RHO	MACH	LOAD FACTOR (G'S)
30000	994.70	0.000889	1.60	7.75
15000	1057.30	0.001496	0.90	8.70

ACCELERATION:

H (FT)	MACH START	MACH END	TIME (SEC)
30000	0.90	1.60	47.35
30000	0.50	1.40	62.14
10000	0.30	0.90	18.44

DRAG POLARS:

H	M	CDobase	CDo	+ K*CL^2	CDwave	CDmissile	CDtrim
0	0.15	0.02277	0.02313	0.11430	0.0000	0.00036	0.00000
0	0.25	0.02198	0.02234	0.10910	0.0000	0.00036	0.00000
15000	0.56	0.02192	0.02407	0.10190	0.0000	0.00035	0.00180
15000	0.90	0.02230	0.02533	0.10030	0.0009	0.00033	0.00180
30000	0.85	0.02288	0.02829	0.11030	0.0004	0.00034	0.00462
30000	1.60	0.02034	0.04069	0.10030	0.0196	0.00031	0.00044
0	0.85	0.02069	0.02173	0.10140	0.0004	0.00032	0.00027

POINT PERFORMANCE WEIGHT

```
Takeoff Weight =                31366 lbs
Operating Empty Weight =        21498 lbs
     50% fuel weight =           4321 lbs
Two short range missiles =        362 lbs
200 rounds ammo =                 110 lbs

Performance Weight =            26291 lbs

Wing Area =                    347.90 ft^2

Performance Wing Loading =      75.57 lb/ft^2
Takeoff Wing Loading =          90.16 lb/ft^2
Takeoff Maximum Thrust =        35573 lb
Takeoff Thrust to Weight =       1.13


PERFORMANCE MATCHING CALCULATIONS:

##########################################################################
LANDING DISTANCE:
                  W/S (L) = 5.50*.002378*CL(MAX)*SL
                  W/S (TO) = W/S (L) / {W(LANDING)/W(TAKEOFF)}
                  NOTE: LANDING AT SEALEVEL
                  W(L)/W(TO) =           0.80

                  V(APPROACH) =        154.56 KNOTS

         CL(MAX)            W/S (L)            W/S (TO)
           1.300              73.11              91.39


##########################################################################
TIME TO CLIMB:
         RC(SEALEVEL) = H(ABS)/T(CL) * LN{ [1- H/H(ABS)]^-1 }
         RC(SEALEVEL) =        528.11   FPS

         (L/D) (MAX) = .5 * SQRT{ PI*A*e/CDo }
         (L/D) (MAX) =          12.0

         V = SQRT{ [2 * (W/S)] / RHO * SQRT(CDo*PI*A*e) }

         P(DL) = (L/D)^2 / [ 1 + (L/D)^2 ]
         P(DL) =                0.99

         (T/W) = RC(SEALEVEL)/V / [ P(DL) - SQRT{ P(DL)^2 - P(DL) + [1 +

           (W/S)            V (FT/S)           (T/W) (MAN)
            90.16            395.08               1.36

*********THRUST REQUIRED =    35635 lbs          H =     0.00 ft
                                                 M =     0.00


##########################################################################
```

```
SPECIFIC EXCESS ENERGY:
        (T/W) (MAN)= PS/V + D/W

        W = W(TAKEOFF) * W(POINT PERFORMANCE)/W(TAKEOFF)
        V = MACH * SPEED OF SOUND
        D = CDo + K*CL^2
        CL = [2 * (W/S)] / [RHO * V^2]

        REQUIREMENT:
                PS =            505    FT/S
                MACH =         0.90
                H =           30000    FT
                V =          895.23    FT/S

  (W/S)             CL                 CD                 (T/W) (MAN)
  75.57           0.2121             0.0298                   0.70

*********THRUST REQUIRED =    18530 lbs           H =     30000 ft
                                                  M =      0.90


        REQUIREMENT:
                PS =            920    FT/S
                MACH =         0.90
                H =           10000    FT
                V =          969.66    FT/S
                                                                  10000.00
      (W/S)             CL                 CD                     (T/W) (MA
      75.57           0.0916             0.0244                       1.21

*********THRUST REQUIRED =    31940 lbs           H =     10000 ft
                                                  M =      0.90


#########################################################################
MANEUVERING:
        (T/W) (MAN) = QBAR * CDo / (W/S)  +  (W/S) * n^2 / [PI*A*e*QBAR
        QBAR = .5 * RHO * V^2
        V = MACH * SPEED OF SOUND
        n = SQRT{ [V * TURN RATE / G]^2 + 1 }

        SUSTAINED TURN RATE:
                H =                   15000    FT
                V =                  845.84    FT/S
                CDo =                0.0253
                TURN RATE =           15.00    DEG/S
                n =                    6.95    G'S
                QBAR =               535.15    PSF


      (W/S)           (T/W) (MAN)                                     CL
      75.57               0.86                                        0.98

*********THRUST REQUIRED =    22699 lbs           H =     15000 ft
                                                  M =      0.80

                H =                   30000    FT
```

```
                    V =                  895.23   FT/S
                    CDo =                0.0283
                    TURN RATE =           10.00   DEG/S
                    n =                    4.95   G'S
                    QBAR =               356.36   PSF

                             30000.00 FT
             (W/S)           (T/W) (MAN)                               CL
              75.57              0.71                                  1.05

*********THRUST REQUIRED =     18601 lbs          H =      30000 ft
                                                  M =       0.90

                    H =                   30000   FT
                    V =                 1193.64   FT/S
                    CDo =                0.0413
                    TURN RATE =            9.90   DEG/S
                    n =                    6.48   G'S
                    QBAR =               633.53   PSF

                             30000.00 FT
             (W/S)           (T/W) (MAN)                               CL
              75.57              0.85                                  0.77

*********THRUST REQUIRED =     22320 lbs          H =      30000 ft
                                                  M =       1.20

                    H =                   30000   FT
                    V =                 1591.52   FT/S
                    CDo =                0.0413
                    n =                    7.75   G'S
                    QBAR =              1126.27   PSF

                             30000.00 FT
             (W/S)           (T/W) (MAN)                               CL
              75.57              1.02                                  0.52

*********THRUST REQUIRED =     26810 lbs          H =      30000 ft
                                                  M =       1.60

                    H =                   15000   FT
                    V =                  951.57   FT/S
                    CDo =                0.0253
                    n =                    8.70   G'S
                    QBAR =               677.12   PSF

                             15000.00 FT
             (W/S)           (T/W) (MAN)                               CL
              75.57              1.07                                  0.97

*********THRUST REQUIRED =     28242 lbs          H =      15000 ft
                                                  M =       0.90


##############################################################################
```

ACCELERATION:

ACCEL = (Tstart - Tend)/Mavg with Mavg = (Wstart + Wend)/2

TIME = (Vend - Vstart)/ACCEL

REQUIREMENT: M = 0.90 to 1.60 at 30,000 ft

H =	30000 FT	Vstart =	895.23 ft/sec
RHO =	0.000889 SLUG/FT^3	Vend =	1591.52 ft/sec
V SOUND =	994.70 FT/SEC		

START CONDITIONS		END CONDITIONS	
W =	26291 lbs	W =	26056 lbs
CL =	0.2121	Q =	1126.27 lb/ft^2
Q =	356.36 lb/ft^2	CL =	0.0665
CD =	0.0332	CD =	0.0411
Thrust =	4122 lbs	Thrust =	16085 lbs
FFlow =	5827.00 (lbf/hr)	FFlow =	61971.60 (lbf/hr)

Acceleration = 14.71 ft/sec^2
Time = 47.3 sec

REQUIREMENT: M = 0.50 to 1.40 at 30,000 ft

H =	30000 FT	Vstart =	497.35 ft/sec
RHO =	0.000889 SLUG/FT^3	Vend =	1392.58 ft/sec
V SOUND =	994.70 FT/SEC		

START CONDITIONS		END CONDITIONS	
W =	26291 lbs	W =	26016 lbs
CL =	0.6871	Q =	862.30 lb/ft^2
Q =	109.99 lb/ft^2	CL =	0.0867
CD =	0.0804	CD =	0.0414
Thrust =	3075 lbs	Thrust =	14785 lbs
FFlow =	2861.00 (lbf/hr)	FFlow =	55390.00 (lbf/hr)

Acceleration = 14.41 ft/sec^2
Time = 62.1 sec

REQUIREMENT: M = 0.30 to 0.90 at 10,000 ft

H =	10000 FT	Vstart =	323.22 ft/sec
RHO =	0.001755 SLUG/FT^3	Vend =	969.66 ft/sec
V SOUND =	1077.40 FT/SEC		

START CONDITIONS		END CONDITIONS	
W =	26291 lbs	W =	26101 lbs
CL =	0.8243	Q =	825.06 lb/ft^2
Q =	91.67 lb/ft^2	CL =	0.0909
CD =	0.1032	CD =	0.0415
Thrust =	3293 lbs	Thrust =	31834 lbs
FFlow =	2861.00 (lbf/hr)	FFlow =	73249.00 (lbf/hr)

Acceleration = 35.05 ft/sec^2
Time = 18.4 sec

SUMMARY OF POINT PERFORMANCE THRUST

	H	M	REQUIREMENTS		(T/W) MAN	W MAN
1A	0.00	0.00	40000.00 80000.00	1.75 min	1.14	31366
2A	30000	0.90		505 ft/sec	0.70	26291
2B	10000	0.90		920 ft/sec	1.21	26291
3A	15000	0.80		15.00 deg/sec	0.86	26291
3B	30000	0.90		10.00 deg/sec	0.71	26291
3C	30000	1.20		9.90 deg/sec	0.85	26291
3D	30000	1.60		7.75 g's	1.02	26291
3E	15000	0.90		8.70 g's	1.07	26291
4A	30000	0.90	1.60	47.3 sec	0.61	26291
4B	30000	0.50	1.40	62.1 sec	0.56	26291
4C	10000	0.30	0.90	18.4 sec	1.21	26291

MISSION ANALYSIS FOR AE 622
Brian Cox
Last Revised: 10 April 1990

The following mission legs burn fuel and need to be accounted for:

Counter Air Superiority	Battlefield Air Int.
1. Engine Start/Warm Up	1. Engine Start/Warm Up
2. Taxi	2. Taxi
3. Short Take-off	3. Short Take-off
4. Accelerate to Climb Speed	4. Accel. to Climb Speed
5. Climb	5. Climb
6. Subsonic Cruise	6. Subsonic Cruise
7. Accelerate to Supersonic Cruise	7. Sea-Level Dash-in
8. Supersonic Cruise	8. Strafe Run
9. Combat	9. Sea-level Dash Out
10. Supersonic Cruise	10. Climb
11. Subsonic Cruise	11. Subsonic Cruise
12. Hover - Half Minute	12. Hover - Half Minute
13. Landing	13. Landing

*************************MISSION ANALYSIS******************************

For the following segments, statistical fuel burns fractions are used due to lack of detailed analsis methods (at this time):

* Engine Start/Warm-up
* Taxi
* Short Takeoff
* Landing

Also, the following is assumed as part of the mission:

CA mission combat fuel burn:	20 % of total fuel
BAI mission strafe run fuel burn:	10 % of total fuel

Equations and methods specific to the aircraft flight phase under consideration will be used for the remaining mission legs.

------------Aircraft Parameters

CA Mission Takeoff Weight =	31366 lbs
BAI Mission Takeoff Weight =	34400 lbs
Fuel Weight =	8642 lbs
Wing Area, S =	347.9 ft^2

CA Mission, WOE =	21498 lbs	BAI Mission, WOE =	21778 lbs
20% fuel =	1728.4 lbs	10% fuel =	864.2 lbs
Hover Weapons =	1196 lbs	Hover Weapons =	2328 lbs
Hover Weight =	24422 lbs	Hover Weight =	24970 lbs
(T/W) Required =	1.27	(T/W) Required =	1.27
Hover Thrust =	31016 lbs	Hover Thrust =	31712 lbs

Drag Polars

H (ft)	Mach	CA Mission CDo +	K*CL^2	BAI #1 CDo +	K*CL^2
0	0.15	0.02277	0.1143	0.02277	0.1143
0	0.25	0.02198	0.1091	0.02198	0.1091
15000	0.56	0.02372	0.1019	0.02372	0.1019
15000	0.90	0.02500	0.1003	not in mission	
30000	0.85	0.02790	0.1103	0.02790	0.1103
30000	1.60	0.04038	0.1003	not in mission	
0	0.85	0.02136	0.1014	0.02136	0.1014

================COUNTER AIR MISSION

1. Engine Start/Warm Up

W1/WTO = 0.99 Fuel Burn = 314 lbs

W1 = 31052 lbs
WF = 8328 lbs

2. Taxi

W2/WTO = 0.99 Fuel Burn = 279 lbs

W2 = 30773 lbs
WF = 8049 lbs

3. Short Takeoff

Using 0.5 minutes for takeoff thrust setting: 0.008333 hrs

LIFT Engine:		CRUISE Engine:	
SFC =	0.8086 (lbf/hr)/lbt	SFC =	1.311 (lbf/hr)/lbt
T(TO) =	12500 lbs	T(TO) =	25250 lbs
WFDOT =	10107.5 lbf/hr	WFDOT =	33102.75 lbf/hr
FuelBurn=	84 lbs	FuelBurn=	276 lbs

Fuel Burn = 360 lbs

W3 = 30413 lbs
WF = 7689 lbs

4. Accelerate to Climb Speed (Out)

Accelerate from M= 0.20 to M = 0.80 at sea level, so

Vstart = 223 ft/sec q(3/4 V)= 533 lb/ft^2
Vend = 893 ft/sec
Acceleration = 25 ft/sec^2
t(acc) = 26.8 sec

Thrust Required = Acceleration Force + Drag

Using H=0, M=0.25 drag polar, the begin weight, and velocity at 75% Vend throughout the acceleration:
Drag increment for two short range missiles: 0.00018

CL = 0.1639 CD = 0.0251

Acceleration Force, F = 23631 lbs
Drag, D = 4657 lbs
Thrust Required, T = 28288 lbs

From Engine Deck, SFC = 1.485 (lbf/hr)/lbt

Fuel Flow, WFDOT = 42008 lbf/hr

****
Fuel Burn = 313 lbs
****

W4 = 30100 lbs
WF = 7376 lbs

5. Climb (Out)

Average Rate of Climb = 15000 ft/min
Time to Climb to 30000 ft = 0.033 hrs

Use climb variables at 2/3 final altitude:

M = 0.80
V = 829 ft/sec
qbar = 436 lb/ft^2

The aircraft travels horizontally 99533 ft
while vertically 30,000 ft so theta is 16.77 degrees.

So, L = 28820 lbs
D = 8686 lbs
CL = 0.1902

Using drag polar for H=15000ft, M=0.56 for H=20000ft, M = 0.8

Drag increment for two short range missiles: 0.000174

CD = 0.0276
D = 4180 lbs

Thrust Required, T = 12866 lbs

From Engine Deck, SFC = 1.132 (lbf/hr)/lbt

Fuel Flow, WFDOT = 14564 lbf/hr

****

Fuel Burn = 485 lbs

****

W5 = 29615 lbs
WF = 6891 lbs

6. Subsonic Cruise (Out)

Range = 100 - 16 = 84 nm
(climb range credit)

Cruise Mach Number = 0.80
qbar = 282 lb/ft^2
CL = 0.3023
Drag increment for two short range missiles: 0.000168
CD = 0.0381

Drag = Thrust Required, T = 3737 lbs

From Engine Deck, SFC = 0.801 (lbf/hr)/lbt
Fuel Flow, WFDOT = 2993 lbf/hr
Cruise Time = 0.177 hrs

****

Fuel Burn = 531 lbs

****

W6 = 29084 lbs
WF = 6360 lbs

7. Accelerate to Supersonic Cruise (Out)

Accelerate from M= 0.80 to M = 1.60 at sea level, so

Vstart = 774 ft/sec q(3/4 V)= 600 lb/ft^2
Vend = 1549 ft/sec
Acceleration = 17 ft/sec^2
t(acc) = 45.6 sec

Thrust Required = Acceleration Force + Drag

Using H=30 k ft, M=1.6 drag polar, the begin weight, and velocity at 75% Vend throughout the acceleration:

Drag increment for two short range missiles: 0.000155

CL = 0.1393 CD = 0.0425

Acceleration Force, F = 15367 lbs
Drag, D = 8869 lbs

Thrust Required, T = 24236 lbs

From Engine Deck, SFC = 2.023 (lbf/hr)/lbt

Fuel Flow, WFDOT = 49030 lbf/hr

****
Fuel Burn = 620 lbs
****

W4 = 28463 lbs
WF = 5739 lbs

8. Supersonic Cruise (Out)

Range = 50 nm

Cruise Mach Number = 1.60
qbar = 1126 lb/ft^2
CL = 0.0726
Drag increment for two short range missiles: 0.000155
CD = 0.0411

Drag = Thrust Required, T = 16090 lbs

From Engine Deck, SFC = 1.563 (lbf/hr)/lbt
Fuel Flow, WFDOT = 25149 lbf/hr
Cruise Time = 0.053 hrs

****
Fuel Burn = 1334 lbs
****

W6 = 27130 lbs
WF = 4406 lbs

9. Combat

Using 20% total fuel for combat:

****
Fuel Burn = 1728 lbs
****

W9 = 25401 lbs
WF = 2677 lbs

Dropping two ASRAAMS and half ammo:

W9 = 24969 lbs

```
                    WF =       2677 lbs

 10. Supersonic Cruise (In)

          Range =        50 nm

          Cruise Mach Number =          1.60
                        qbar =          1126 lb/ft^2
                          CL =        0.0637
                          CD =        0.0408

  Drag = Thrust Required, T =          15982 lbs

       From Engine Deck, SFC =         1.563 (lbf/hr)/lbt
            Fuel Flow, WFDOT =         24979 lbf/hr
                 Cruise Time =         0.053 hrs

                                  **************************
                                        Fuel Burn =    1325 lbs
                                  **************************

                    W11 =      23645 lbs
                    WF =        1353 lbs

 11. Subsonic Cruise (In)

          Range =        100 nm

          Cruise Mach Number =          0.80
                        qbar =           282 lb/ft^2
                          CL =        0.2414
                          CD =        0.0343

  Drag = Thrust Required, T =           3363 lbs

       From Engine Deck, SFC =         0.801 (lbf/hr)/lbt
            Fuel Flow, WFDOT =          2693 lbf/hr
                 Cruise Time =         0.212 hrs

                                  **************************
                                        Fuel Burn =     571 lbs
                                  **************************

                    W12 =      23074 lbs
                    WF =         782 lbs

 12. Hover

          Half Minute =      0.0083 hrs

       Hover Thrust:
```

LIFT Engine = 12105 lbs
MAIN Engine = 18911 lbs

From Engine Deck, SFC:
LIFT Engine = 0.8094 (lbf/hr)/lbt
MAIN Engine = 0.924 (lbf/hr)/lbt

Fuel Flow, WFDOT:
LIFT Engine = 9798 lb/hr
MAIN Engine = 17474 lb/hr

Fuel Burn, Wburn:
LIFT Engine = 82 lb
MAIN Engine = 146 lb

****
Fuel Burn = 227 lbs
****

W13 = 22846 lbs
WF = 554 lbs

13. Landing

****
W14/W13 = 0.995 Fuel Burn = 114 lbs
****

W14 = 22732 lbs
WF = 440 lbs

--------------------CA Mission Fuel Burn Summary-----------------------

Phase	Fuel Burn
1. Engine Start/Warm Up	314 lbs
2. Taxi	279 lbs
3. Short Take-off	360 lbs
4. Accel. to Climb Speed	313 lbs
5. Climb	485 lbs
6. Subsonic Cruise	531 lbs
7. Accel. to Supersonic Cruise	620 lbs
8. Supersonic Cruise	1334 lbs
9. Combat	1728 lbs
10. Supersonic Cruise	1325 lbs
11. Subsonic Cruise	571 lbs
12. Hover - Half Minute	227 lbs
13. Landing	114 lbs

CA Mission Fuel = 8202 lbs

============BATTLEFIELD AIR INTERDICTION MISSION

1. Engine Start/Warm Up

```
                            **************************
        W1/WTO =     0.99         Fuel Burn =     327 lbs
                            **************************

            W1 =    34073 lbs
            WF =     8315 lbs
```

2. Taxi

```
                            **************************
        W2/WTO =     0.99         Fuel Burn =     307 lbs
                            **************************

            W2 =    33767 lbs
            WF =     8009 lbs
```

3. Short Takeoff

Using 0.5 minutes for takeoff thrust setting: 0.008333 hrs

```
LIFT Engine:                         CRUISE Engine:
    SFC =   0.773 (lbf/hr)/lbt            SFC =   1.311 (lbf/hr)/lbt
  T(TO) =   12500 lbs                   T(TO) =   27000 lbs
  WFDOT = 9668.75 lbf/hr                WFDOT =   35397 lbf/hr
FuelBurn=      81 lbs                FuelBurn=      295 lbs

                            **************************
                                  Fuel Burn =     376 lbs
                            **************************

          W3 =      33391 lbs
          WF =       7633 lbs
```

4. Accelerate to Climb Speed (Out)

Accelerate from M= 0.20 to M = 0.80 at sea level, so

```
          Vstart =      223 ft/sec              q(3/4 V)=     533 lb/ft^2
            Vend =      893 ft/sec
    Acceleration =       24 ft/sec^2
          t(acc) =     27.9 sec
```

Thrust Required = Acceleration Force + Drag

Using H=0, M=0.25 drag polar, the begin weight, and velocity at 75% Vend throughout the acceleration:

Drag increment for BAI #1 mission: 0.00331

CL = 0.1799 CD = 0.0288

Acceleration Force, F = 24908 lbs
Drag, D = 5349 lbs

Thrust Required, T = 30257 lbs

From Engine Deck, SFC = 1.311 (lbf/hr)/lbt

Fuel Flow, WFDOT = 39667 lbf/hr

****
Fuel Burn = 308 lbs
****

W4 = 33083 lbs
WF = 7325 lbs

5. Climb (Out)

Average Rate of Climb = 15000 ft/min
Time to Climb to 30000 ft = 0.033 hrs

Use climb variables at 2/3 final altitude:

M = 0.80
V = 829 ft/sec
qbar = 436 lb/ft^2

The aircraft travels horizontally 99533 ft
while vertically 30,000 ft so theta is 16.77 degrees.

So, L = 31676 lbs
D = 9547 lbs
CL = 0.2090

Using drag polar for H=15000ft, M=0.56 for H=20000ft, M = 0.8

Drag increment for BAI #1 mission: 0.00339
CD = 0.0316
D = 4783 lbs

Thrust Required, T = 14331 lbs

From Engine Deck, SFC = 1.126 (lbf/hr)/lbt

Fuel Flow, WFDOT = 16136 lbf/hr

****
Fuel Burn = 538 lbs
****

W5 = 32546 lbs
WF = 6788 lbs

6. Subsonic Cruise (Out)

Range = 200 - 16 = 184 nm
(climb range credit)

Cruise Mach Number = 0.80
qbar = 282 lb/ft^2
CL = 0.3322
Drag increment for BAI #1 mission: 0.0032
CD = 0.0433

Drag = Thrust Required, T = 4239 lbs

From Engine Deck, SFC = 0.806 (lbf/hr)/lbt
Fuel Flow, WFDOT = 3417 lbf/hr
Cruise Time = 0.389 hrs

****
Fuel Burn = 1331 lbs
****

W6 = 31215 lbs
WF = 5457 lbs

7. Sea Level Dash (Out)

Range = 80 nm

Cruise Mach Number = 0.85
qbar = 1071 lb/ft^2
CL = 0.0838
Drag increment for BAI #1 mission: 0.00309
CD = 0.0252

Drag = Thrust Required, T = 9373 lbs

From Engine Deck, SFC = 0.903 (lbf/hr)/lbt
Fuel Flow, WFDOT = 8463 lbf/hr
Cruise Time = 0.142 hrs

****
Fuel Burn = 1204 lbs
****

W7 = 30011 lbs
WF = 4253 lbs

8. Strafe Run

Using 10% total fuel for strafe run:

****

Fuel Burn = 864 lbs
****

W8 = 29146 lbs
WF = 3388 lbs

Dropping two Mark 82 Bombs and two AGM 65's:

W8 = 27038 lbs
WF = 3388 lbs

9. Sea Level Dash (In)

Range = 80 nm

Cruise Mach Number =	0.85
qbar =	1071 lb/ft^2
CL =	0.0726
Drag increment for two AGM 65:	0.0013
CD =	0.0232
Drag = Thrust Required, T =	8640 lbs
From Engine Deck, SFC =	0.903 (lbf/hr)/lbt
Fuel Flow, WFDOT =	7802 lbf/hr
Cruise Time =	0.142 hrs

****
Fuel Burn = 1110 lbs
****

W9 = 25928 lbs
WF = 2278 lbs

10. Climb (In)

Average Rate of Climb = 15000 ft/min
Time to Climb to 30000 ft = 0.033 hrs

Use climb variables at 2/3 final altitude:

M = 0.80
V = 829 ft/sec
qbar = 436 lb/ft^2

The aircraft travels horizontally 99533 ft
while vertically 30,000 ft so theta is 16.77 degrees.

So, L = 24825 lbs
D = 7483 lbs
CL = 0.1638

Using drag polar for H=15000ft, M=0.56 for H=20000ft, M = 0.8

Drag increment for two AGM 65: 0.0014
CD = 0.0279
D = 4221 lbs

Thrust Required, T = 11704 lbs

From Engine Deck, SFC = 0.836 (lbf/hr)/lbt

Fuel Flow, WFDOT = 9784 lbf/hr

****
Fuel Burn = 326 lbs
****

W10 = 25602 lbs
WF = 1952 lbs

11. Subsonic Cruise (In)

Range = 200 - 16 = 184 nm
(climb range credit)

Cruise Mach Number = 0.80
qbar = 282 lb/ft^2
CL = 0.2614
Drag increment for two AGM 65: 0.00134
CD = 0.0368

Drag = Thrust Required, T = 3602 lbs

From Engine Deck, SFC = 0.801 (lbf/hr)/lbt
Fuel Flow, WFDOT = 2885 lbf/hr
Cruise Time = 0.389 hrs

****
Fuel Burn = 1124 lbs
****

W11 = 24479 lbs
WF = 829 lbs

12. Hover

Half Minute = 0.0083 hrs

Hover Thrust:
LIFT Engine = 12800 lbs
MAIN Engine = 18912 lbs

From Engine Deck, SFC:
LIFT Engine = 0.8086 (lbf/hr)/lbt
MAIN Engine = 1.01498 (lbf/hr)/lbt

```
Fuel Flow, WFDOT:
    LIFT Engine =        10350 lb/hr
    MAIN Engine =        19195 lb/hr

Fuel Burn, Wburn:
    LIFT Engine =           86 lb
    MAIN Engine =          160 lb

                          **************************
                                Fuel Burn =     246 lbs
                          **************************

                 W13 =     24232 lbs
                 WF =        582 lbs
```

13. Landing

```
                          **************************
        W13/W12 =   0.995       Fuel Burn =     121 lbs
                          **************************

            W13 =   24111 lbs
            WF =      461 lbs
```

--------------------BAI Mission Fuel Burn Summary----------------------

Phase	Fuel Burn
1. Engine Start/Warm Up	327 lbs
2. Taxi	307 lbs
3. Short Take-off	376 lbs
4. Accel. to Climb Speed	308 lbs
5. Climb	538 lbs
6. Subsonic Cruise	1331 lbs
7. Sea-Level Dash-in	1204 lbs
8. Strafe Run	864 lbs
9. Sea-level Dash Out	1110 lbs
10. Climb	326 lbs
11. Subsonic Cruise	1124 lbs
12. Hover - Half Minute	246 lbs
13. Landing	121 lbs

BAI Mission Fuel = 8181 lbs

PERFORMANCE FOR THE MONARCH 13 April 1990
Brian Cox

W = 26291.00 (W/S) = 75.57
S = 347.90
Trim Drag Factor = 1.04

Mach	Altitude	0.00	5000.00	10000.00	15000.00	20000.00	25000.00	30000.00	36089.00	40000.00	45000.00	50000.00
0.20	Delta =	1.00										
	a =	1116.40										
	qbar =	59.25										
	T =	34209.20										
	CDo =	0.02330										
	CL =	1.28										
	PI*A*e=	8.95										
	D =	4225.53										
	n(max)=	1.0200										
	maxturn=	1.66										
	Ps =	254.98										
0.40	Delta =	1.00	0.83	0.69	0.56	0.46	0.37	0.30				
	a =	1116.40	1097.10	1077.40	1057.30	1036.80	1016.00	994.70				
	qbar =	237.01	197.19	162.99	133.74	108.91	87.95	70.39				
	T =	34333.20	30691.20	26943.60	23317.20	19320.00	15890.40	12944.40				
	CDo =	0.02271	0.02330	0.02388	0.02447	0.02468	0.02490	0.02511				
	CL =	0.32	0.38	0.46	0.57	0.69	0.86	1.07				
	PI*A*e=	9.33	9.36	9.40	9.43	9.15	8.86	8.58				
	D =	2770.97	2674.15	2651.26	2713.39	2929.48	3310.63	3905.58				
	n(max)=	3.7900	3.1600	2.6100	2.1400	1.7400	1.4100	1.1300			1.0000	
	maxturn=	15.09	12.59	10.31	8.25	6.33	4.51	2.44				
	Ps =	536.09	467.65	398.20	331.44	258.55	194.46	136.79				
0.60	Delta =	1.00	0.83	0.69	0.56	0.46	0.37	0.30	0.22	0.19	0.15	
	a =	1116.40	1097.10	1077.40	1057.30	1036.80	1016.00	994.70	968.10	968.10	968.10	
	qbar =	533.27	443.68	366.73	300.92	245.04	197.90	158.38	119.13	98.71	77.59	
	T =	36145.20	32484.00	28812.00	25094.40	21542.40	17736.00	14481.60	11158.80	9229.20	7226.40	
	CDo =	0.02212	0.02269	0.02326	0.02384	0.02404	0.02425	0.02446	0.02471	0.02495	0.02520	
	CL =	0.14	0.17	0.21	0.25	0.31	0.39	0.48	0.63	0.77	0.97	
	PI*A*e=	9.71	9.74	9.78	9.81	9.52	9.22	8.92	8.83	8.75	8.66	
	D =	4487.00	3962.43	3522.31	3168.26	2901.76	2758.67	2753.68	2911.87	3150.46	3637.06	
	n(max)=	7.9000	6.5000	5.4400	4.4600	3.6300	2.9300	2.3500	1.7700	1.4600	1.1500	
	maxturn=	21.57	18.21	15.25	12.63	10.34	8.33	6.57	4.63	3.38	1.80	
	Ps =	806.56	714.11	621.82	529.06	441.06	347.27	266.23	182.20	134.12	79.28	

0.80											
Delta =	1.00	0.83	0.69	0.56	0.46	0.37	0.30	0.22	0.19	0.15	0.11
a =	1116.40	1097.10	1077.40	1057.30	1036.80	1016.00	994.70	968.10	968.10	968.10	968.10
qbar =	948.03	788.76	651.96	534.97	435.62	351.81	281.57	211.79	175.48	137.94	108.55
T =	37284.00	33986.40	31040.40	27339.60	23893.20	20551.20	16916.40	13070.40	10822.80	8485.20	6633.50
CDo =	0.02152	0.02208	0.02263	0.02319	0.02339	0.02359	0.02380	0.02403	0.02427	0.02452	0.02476
CL =	0.08	0.10	0.12	0.14	0.17	0.21	0.27	0.36	0.43	0.55	0.70
PI*A*e=	9.86	9.90	9.93	9.97	9.67	9.37	9.07	8.98	8.89	8.80	8.71
D =	7309.49	6312.33	5440.55	4688.94	4017.00	3490.71	3109.23	2816.02	2756.11	2813.90	3036.02
n(max)=	9.0000	9.0000	8.8900	7.2900	5.9100	4.8000	3.8400	2.8900	2.3900	1.8800	1.4900
maxturn=	18.46	18.79	18.89	15.74	13.01	10.65	8.59	6.45	5.17	3.79	2.60
Ps =	1010.25	923.85	839.26	728.72	627.06	527.43	417.91	302.07	237.63	167.07	105.95

0.90											
Delta =	1.00	0.83	0.69	0.56	0.46	0.37	0.30	0.22	0.19	0.15	0.11
a =	1116.40	1097.10	1077.40	1057.30	1036.80	1016.00	994.70	968.10	968.10	968.10	968.10
qbar =	1199.85	998.28	825.14	677.08	551.33	445.27	356.36	268.05	222.09	174.58	137.38
T =	38274.00	35434.00	31833.60	28550.40	25252.80	21847.20	18566.40	14364.00	11892.00	9330.00	7308.00
CDo =	0.02155	0.02211	0.02266	0.02322	0.02342	0.02362	0.02383	0.02406	0.02430	0.02455	0.02479
CL =	0.06	0.08	0.09	0.112	0.14	0.17	0.212	0.28	0.34	0.43	0.55
PI*A*e=	9.86	9.90	9.93	9.97	9.67	9.37	9.07	8.98	8.89	8.80	8.71
D =	9162.53	7870.44	6748.42	5764.39	4865.46	4135.91	3568.74	3069.82	2884.63	2784.58	2845.54
n(max)=	9.0000	9.0000	9.0000	8.7800	7.1500	5.7700	4.6200	3.4800	2.8000	2.2600	1.7800
maxturn=	16.41	16.70	17.00	16.90	13.99	11.46	9.29	7.05	5.71	4.29	3.12
Ps =	1112.55	1034.91	925.19	824.71	723.59	616.00	510.68	370.29	298.51	216.92	147.89

1.00											
Delta =	1.00	0.83	0.69	0.56	0.46	0.37	0.30	0.22	0.19	0.15	0.11
a =	1116.40	1097.10	1077.40	1057.30	1036.80	1016.00	994.70	968.10	968.10	968.10	968.10
qbar =	1481.30	1232.44	1010.69	835.90	680.66	549.71	439.95	330.92	274.19	215.53	169.61
T =	39266.40	35068.00	32754.00	29529.60	26415.60	23185.20	19861.80	15823.20	13106.40	10290.00	8064.00
CDo =	0.03652	0.03708	0.03763	0.03819	0.03839	0.03859	0.03880	0.03903	0.03927	0.03952	0.03976
CL =	0.05	0.06	0.07	0.09	0.11	0.14	0.17	0.23	0.28	0.35	0.45
PI*A*e=	9.86	9.90	9.93	9.97	9.67	9.37	9.07	8.98	8.89	8.80	8.71
D =	18955.17	16059.72	13533.87	11344.97	9393.41	7766.75	6436.02	5162.74	4561.81	4010.96	3691.30
n(max)=	9.0000	9.0000	9.0000	8.7907	7.5752	6.4000	5.2876	4.0700	3.3700	2.6402	2.0518
maxturn=	14.77	15.03	15.30	15.23	13.35	11.47	9.62	7.51	6.13	4.66	3.43
Ps =	862.48	826.58	787.64	731.30	671.28	595.84	508.07	392.54	314.63	231.21	161.01

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1.20											
Delta =	1.00	0.83	0.69	0.56	0.46	0.37	0.30	0.22	0.19	0.15	0.11
a =	1116.40	1097.10	1077.40	1057.30	1036.80	1016.00	994.70	968.10	968.10	968.10	968.10
qbar =	2133.07	1774.72	1466.91	1203.69	980.15	791.58	633.52	476.53	394.83	310.36	244.24
T =	41539.20	37983.60	34921.20	31584.00	28552.80	25453.20	22460.40	18699.60	15420.00	12038.40	9362.40
CDo =	0.04102	0.04156	0.04209	0.04263	0.04283	0.04302	0.04321	0.04344	0.04368	0.04391	0.04415
CL =	0.04	0.04	0.05	0.06	0.08	0.10	0.12	0.16	0.19	0.24	0.31
PI*A*e=	10.19	10.23	10.26	10.30	9.99	9.68	9.37	9.27	9.18	9.09	9.00
D =	30532.39	25768.13	21814.86	18013.12	14806.41	12106.83	9859.42	7651.89	6547.42	5465.39	4655.46
n(max)=	9.0000	9.0000	9.0000	9.0000	8.2913	7.2433	6.2159	5.0569	4.1450	3.2188	2.4914
maxturn=	12.31	12.52	12.75	13.00	12.20	10.85	9.47	7.87	6.38	4.85	3.62
Ps =	560.86	611.69	654.36	654.91	650.52	618.92	572.10	488.16	392.05	291.33	208.00

1.40									
Delta =	0.69	0.56	0.46	0.37	0.30	0.22	0.19	0.15	0.11
a =	1077.40	1057.30	1036.80	1016.00	994.70	968.10	968.10	968.10	968.10
qbar =	1996.63	1638.36	1334.09	1077.43	862.29	640.61	537.41	422.44	332.55
T =	37819.20	34136.40	30834.00	27633.60	24640.80	20964.00	17230.80	13401.60	10399.20
CDo =	0.04046	0.04097	0.04116	0.04135	0.04153	0.04175	0.04198	0.04220	0.04243
CL =	0.04	0.05	0.06	0.07	0.09	0.12	0.14	0.18	0.23
PI*A*e=	10.59	10.63	10.31	9.99	9.67	9.57	9.48	9.38	9.29
D =	28195.02	23467.79	19247.07	15683.02	12698.22	9741.95	8238.36	6703.56	5552.03
n(max)=	9.0000	9.0000	9.0122	8.1081	7.1494	6.0056	4.9000	3.7895	2.9215
maxturn=	10.93	11.14	11.37	10.43	9.37	8.05	6.53	4.97	3.73
Ps =	552.11	600.66	639.67	646.55	632.57	578.51	463.57	345.29	249.88

1.60							
Delta =	0.46	0.37	0.30	0.22	0.19	0.15	0.11
a =	1036.80	1016.00	994.70	968.10	968.10	968.10	968.10
qbar =	1742.48	1407.26	1126.26	847.16	701.92	551.75	434.20
T =	34048.80	30285.60	26800.00	23000.40	18867.60	14630.80	11309.60
CDo =	0.04039	0.04057	0.04075	0.04097	0.04118	0.04139	0.04161
CL =	0.04	0.05	0.07	0.09	0.11	0.14	0.17
PI*A*e=	10.63	10.30	9.97	9.87	9.77	9.67	9.58
D =	24593.91	20001.16	16145.28	12311.16	10345.49	8318.15	6703.60
n(max)=	9.0000	8.7200	7.8270	6.7813	5.5154	4.2403	3.2555
maxturn=	9.94	9.82	8.99	7.98	6.46	4.90	3.69
Ps =	596.57	635.90	645.47	629.77	502.09	372.35	270.18

1.90

Delta =	0.30	Delta =	0.22	Delta =	0.19	Delta =	0.15	Delta =	0.11
a =	994.70	a =	968.10	a =	968.10	a =	968.10	a =	968.10
qbar =	1425.43	qbar =	1072.19	qbar =	888.37	qbar =	698.31	qbar =	549.53
T =	27640.80	T =	23500.00	T =	19363.20	T =	15022.80	T =	11647.20
CDo =	0.04033	CDo =	0.04054	CDo =	0.04075	CDo =	0.04096	CDo =	0.04117
CL =	0.05	CL =	0.07	CL =	0.09	CL =	0.11	CL =	0.14
PI*A*e=	10.17	PI*A*e=	10.07	PI*A*e=	9.97	PI*A*e=	9.87	PI*A*e=	9.77
D =	20137.19	D =	15305.26	D =	12817.90	D =	10230.99	D =	8241.52
n(max)=	7.4663	n(max)=	6.7792	n(max)=	5.4927	n(max)=	4.1941	n(max)=	3.1940
maxturn=	7.62	maxturn=	7.09	maxturn=	5.71	maxturn=	4.31	maxturn=	3.21
Ps =	511.01	Ps =	548.45	Ps =	433.83	Ps =	317.07	Ps =	225.73

2.00

Delta =	0.22	Delta =	0.19	Delta =	0.15	Delta =	0.11
a =	968.10	a =	968.10	a =	968.10	a =	968.10
qbar =	1072.19	qbar =	888.37	qbar =	698.31	qbar =	549.53
T =	23930.40	T =	19681.20	T =	15277.20	T =	11846.40
CDo =	0.04012	CDo =	0.04032	CDo =	0.04053	CDo =	0.04074
CL =	0.07	CL =	0.09	CL =	0.11	CL =	0.14
PI*A*e=	10.2691	PI*A*e=	10.1664	PI*A*e=	10.0647	PI*A*e=	9.9641
D =	15145.44	D =	12682.78	D =	10129.56	D =	8151.70
n(max)=	7.0487	n(max)=	5.7282	n(max)=	4.3829	n(max)=	3.3400
maxturn=	7.38	maxturn=	5.97	maxturn=	4.51	maxturn=	3.38
Ps =	582.27	Ps =	463.86	Ps =	341.19	Ps =	244.88

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PERFORMANCE FOR THE MONARCH - DOG HOUSE PLOT 14 April 1990

Brian Cox

W =	26291.00		(W/S) =	75.57
S =	347.90		a =	1057.3 ft/sec
H =	15000			
DELTA =	0.5643			

MACH	CDo	K	QBAR	CL	CD	Thrust	Thrust n(max)	CLmax	Lift n(max)	n(max)	T RATE
0.264	0.0241	9.053	58	1.3001	0.2108	22305.60	2.40	1.30	1.00	1.00	0.00
0.3	0.0238	9.243	75	1.0045	0.1330	22811.40	2.79	1.25	1.25	1.25	4.35
0.4	0.0235	9.434	134	0.5650	0.0574	23317.20	3.76	1.21	2.14	2.14	8.25
0.5	0.0232	9.624	209	0.3616	0.0368	24205.80	4.77	1.17	3.22	3.22	10.68
0.6	0.0229	9.814	301	0.2511	0.0293	25094.40	5.81	1.12	4.46	4.46	12.63
0.7	0.0226	9.892	410	0.1845	0.0261	26217.00	6.85	1.08	5.83	5.83	14.30
0.8	0.0223	9.970	535	0.1413	0.0243	27339.60	7.39	1.03	7.29	7.29	15.74
0.9	0.0232	9.970	677	0.1116	0.0244	28550.40	8.86	0.98	8.78	8.78	16.90
1.0	0.0373	9.970	836	0.0904	0.0381	29529.60	8.85	0.93	10.29	8.85	15.34
1.1	0.0395	10.136	1011	0.0747	0.0401	30556.80	9.27	0.89	11.94	9.00	14.18
1.2	0.0418	10.301	1204	0.0628	0.0422	31584.00	9.38	0.84	13.28	9.00	13.00
1.3	0.0410	10.467	1413	0.0535	0.0412	32860.20	9.73	0.73	13.74	9.00	12.00
1.4	0.0401	10.633	1638	0.0461	0.0403	34136.40	9.93	0.63	13.66	9.00	11.14
1.5	0.0398	10.798	1881	0.0402	0.0399	35502.00	9.84	0.59	14.59	9.00	10.40
1.6	0.0394	10.964	2140	0.0353	0.0395	36867.60	9.42	0.55	15.57	9.00	9.75

MISSION PERFORMANCE FOR AE 622 - FERRY RANGE CALCULATIONS
Brian Cox
Last Revised: 14 April 1990

------------Aircraft Parameters

CA Mission Takeoff Weight =	33076 lbs
BAI Mission Takeoff Weight =	34400 lbs
Fuel Weight =	10352 lbs
Wing Area, S =	347.9 ft^2

CA Mission, WOE =	21498 lbs
20% fuel =	2070.4 lbs
Hover Weapons =	1196 lbs
Hover Weight =	24764 lbs
(T/W) Required =	1.27
Hover Thrust =	31451 lbs

Drag Polars

		CA Mission		BAI #1	
H (ft)	Mach	CDo +	K*CL^2	CDo +	K*CL^2
0	0.15	0.02277	0.1143	0.02277	0.1143
0	0.25	0.02198	0.1091	0.02198	0.1091
15000	0.56	0.02372	0.1019	0.02372	0.1019
15000	0.90	0.02500	0.1003	not in mission	
30000	0.85	0.02790	0.1103	0.02790	0.1103
30000	1.60	0.04038	0.1003	not in mission	
0	0.85	0.02136	0.1014	0.02136	0.1014

===========BEST MACH AND ALTITUDE STUDY

1. START AIRCRAFT

****

W1/WTO = 0.99 Fuel Burn = 198 lbs

****

W1 = 32878 lbs
WF = 10154 lbs

2. TAXI

****

W2/WTO = 0.99 Fuel Burn = 164 lbs

****

W2 = 32713 lbs
WF = 9989 lbs

3. TAKEOFF

Using 0.35 minutes for takeoff thrust setting: 0.005833 hrs

LIFT Engine:		CRUISE Engine:	
SFC =	0.8086 (lbf/hr)/lbt	SFC =	1.311 (lbf/hr)/lbt
T(TO) =	12500 lbs	T(TO) =	25250 lbs
WFDOT =	10107.5 lbf/hr	WFDOT =	33102.75 lbf/hr
FuelBurn=	59 lbs	FuelBurn=	193 lbs

****

Fuel Burn = 252 lbs

****

W3 = 32461 lbs
WF = 9737 lbs

4. ACCELRATE TO CLIMB

Accelerate from M= 0.20 to M = 0.80 at sea level, so

Vstart = 223 ft/sec q(3/4 V)= 533 lb/ft^2
Vend = 893 ft/sec
Acceleration = 25 ft/sec^2
t(acc) = 26.8 sec

Thrust Required = Acceleration Force + Drag

Using H=0, M=0.25 drag polar, the begin weight, and velocity at 75% Vend
throughout the acceleration:
Drag increment for two short range missiles: 0.00018

CL = 0.1749 CD = 0.0255

Acceleration Force, F = 25223 lbs
Drag, D = 4732 lbs
Thrust Required, T = 29955 lbs

From Engine Deck, SFC = 1.485 (lbf/hr)/lbt

Fuel Flow, WFDOT = 44484 lbf/hr

****

Fuel Burn = 331 lbs

****

W4 = 32130 lbs
WF = 9406 lbs

5. CLIMB

Average Rate of Climb = 25000 ft/min
Time to Climb to 30000 ft = 0.020 hrs

Use climb variables at 2/3 final altitude:

M = 0.80
V = 829 ft/sec
qbar = 436 lb/ft^2

The aircraft travels horizontally 59720 ft
while vertically 30,000 ft so theta is 26.67 degrees.

So, L = 28711 lbs
D = 14423 lbs
CL = 0.1894

Using drag polar for H=15000ft, M=0.56 for H=20000ft, M = 0.8

Drag increment for two short range missiles: 0.000174

CD = 0.0276
D = 4175 lbs

Thrust Required, T = 18598 lbs

From Engine Deck, SFC = 1.132 (lbf/hr)/lbt

Fuel Flow, WFDOT = 21053 lbf/hr

****
Fuel Burn = 421 lbs
****

W5 = 31709 lbs
WF = 8985 lbs

6. SUBSONIC CRUISE I.E. FERRY RANGE

FERRY RANGE WING LOADING = 80.20

H = 30000 FT a = 995

MACH	QBAR	CL	CDo	PI*A*e	TREQ	FUEL FLOW	TIME	RANGE (NM
0.40	70	1.139	0.0241	8.784	4209	2976	10730	703
0.50	110	0.729	0.0238	8.854	3208	2610	12232	1001
0.60	158	0.506	0.0235	8.923	2878	2245	14224	1397
0.70	216	0.372	0.0232	8.995	2894	2510	12720	1458
0.80	282	0.285	0.0229	9.066	3120	2776	11505	1507
0.90	356	0.225	0.0238	9.066	3643	3506	9108	1342
1.00	440	0.182	0.0379	9.066	6362	6440	4958	812
1.10	532	0.151	0.0435	9.217	8512	9254	3450	621
1.20	634	0.127	0.0423	9.367	9700	12069	2646	520
1.30	744	0.108	0.0414	9.518	11025	15553	2053	437
1.40	862	0.093	0.0407	9.669	12478	19037	1677	384
1.50	990	0.081	0.0404	9.819	14143	22250	1435	352
1.60	1126	0.071	0.0399	9.970	15833	25462	1254	328

H = 36089 FT a = 968

MACH	QBAR	CL	CDo	PI*A*e	TREQ	FUEL FLOW	TIME	RANGE (NM
0.50	83	0.969	0.0241	4.417	6817	2723	11728	934
0.60	119	0.673	0.0238	8.834	3112	2368	13487	1289
0.70	162	0.495	0.0234	8.905	2870	2430	13140	1466
0.80	212	0.379	0.0231	8.975	2879	2493	12811	1633
0.90	268	0.299	0.0240	8.975	3168	2928	10906	1564
1.00	331	0.242	0.0381	8.975	5140	5095	6268	999
1.10	400	0.200	0.0437	9.125	6700	7084	4508	790
1.20	477	0.168	0.0426	9.274	7569	9073	3519	673
1.30	559	0.143	0.0416	9.423	8519	11529	2770	574
1.40	649	0.124	0.0409	9.572	9590	13984	2283	509
1.50	745	0.108	0.0406	9.721	10826	16263	1963	469
1.60	847	0.095	0.0401	9.870	12086	18541	1722	439

H = 40000 FT a = 968

MACH	QBAR	CL	CDo	PI*A*e	TREQ	FUEL FLOW	TIME	RANGE (NM
0.50	69	1.170	0.0243	4.373	8044	3008	10616	846
0.60	99	0.812	0.0240	8.745	3416	2616	12208	1167
0.70	134	0.597	0.0237	8.816	2995	2554	12503	1395
0.80	175	0.457	0.0233	8.886	2860	2492	12813	1633
0.90	222	0.361	0.0242	8.886	3007	2794	11429	1639
1.00	274	0.292	0.0383	8.886	4576	4559	7004	1116
1.10	332	0.242	0.0439	9.033	5815	6176	5170	906
1.20	395	0.203	0.0428	9.181	6493	7794	4097	783
1.30	463	0.173	0.0418	9.329	7263	9838	3246	672
1.40	537	0.149	0.0411	9.476	8126	11881	2688	599
1.50	617	0.130	0.0408	9.624	9130	13772	2319	554
1.60	702	0.114	0.0403	9.772	10179	15662	2039	520

H = 45000 FT a = 968.1

MACH	QBAR	CL	CDo	PI*A*e	TREQ	FUEL FLOW	TIME	RANGE (NM
0.60	78	1.034	0.0242	8.658	3985	4615	6919	661
0.70	106	0.759	0.0239	8.727	3306	3614	8836	986
0.80	138	0.581	0.0236	8.797	2975	2612	12223	1558
0.90	175	0.459	0.0245	8.797	2943	2756	11587	1662
1.00	216	0.372	0.0386	8.797	4072	4075	7836	1249
1.10	261	0.308	0.0441	8.943	4964	5307	6017	1055
1.20	310	0.258	0.0430	9.089	5436	6540	4883	934
1.30	364	0.220	0.0421	9.235	5996	8139	3923	813
1.40	422	0.190	0.0413	9.381	6638	9738	3279	731

1.50	485	0.165	0.0410	9.528	7400	11203	2850	681
1.60	552	0.145	0.0406	9.674	8204	12669	2521	643

H = 50000 FT   a =   968.1

MACH	QBAR	CL	CDo	PI*A*e	TREQ	FUEL FLOW	TIME	RANGE (NM
0.80	109	0.739	0.0238	8.709	3266	3137	10179	1297
0.90	137	0.584	0.0247	8.709	3051	2828	11291	1619
1.00	170	0.473	0.0388	8.709	3805	4080	7826	1247
1.10	205	0.391	0.0444	8.853	4399	5336	5985	1049
1.20	244	0.328	0.0432	8.998	4691	6591	4845	926
1.30	287	0.280	0.0423	9.143	5071	7370	4333	897
1.40	332	0.241	0.0415	9.288	5530	8150	3918	874
1.50	382	0.210	0.0412	9.432	6092	9265	3446	824
1.60	434	0.185	0.0408	9.577	6696	10381	3076	784

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MISSION CAPABILITY ANALYSIS   19 April 1990
Brian Cox

****
The following are constants for all missions:

Total Mission Fuel	8642 lbs
Takeoff	953 lbs
Acclerate to Climb at Sea Level	313 lbs
Accelerate to Supersonic Cruise	627 lbs
Hover	227 lbs
Landing	114 lbs
Reserves	432 lbs
Climb (lb fuel/ft)	0.0162 lbs/ft
Subsonic Cruise (lb fuel/nuatical mile)	5.71 lbs/nm
Supersonic Cruise (lb fuel/nautical mile)	26.70 lbs/nm
Low Level Dash (lb fuel/nautical mile)	15.05 lbs/nm

****

MASS INTERCEPT

		Fuel Burn
1. Takeoff		953 lbs
2. Climb to	35000 ft	880 lbs
3. Accelerate to Supersonic Cruise		627 lbs
4. Supersonic Cruise for	115 nm	2670 lbs
5. Dash at 5000 ft for	30 nm	452 lbs
6. Combat with K =	0.15 % of total fuel	1296 lbs
7. Climb to	30000 ft	486 lbs
8. Subsonic Cruise for	85 nm	485 lbs
9. Hover, Landing, and Reserves		773 lbs
	Total Fuel Burn	8622 lbs

HIGH VALUE ASSEST PROTECTION

		Fuel Burn
1. Takeoff		953 lbs
2. Climb to	45000 ft	1042 lbs
2.5 Loiter for	0.506553 hrs	1042 lbs
3. Accelerate to Supersonic Cruise		627 lbs
4. Supersonic Dash for	50 nm	1335 lbs
5. Shoot Missiles at Optimum Climbing Turn		1107 lbs
6. Supersonic Dash for	50 nm	1335 lbs
7. Subsonic Cruise for	75 nm	428 lbs

8. Hover, Landing, and Reserves	773 lbs
	7600 lbs
Total Fuel Burn	8642 lbs

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TRANSPORT (HELICOPTER) INTERCEPT
=================================

	Fuel Burn
1. Takeoff	953 lbs
2. Climb to 30000 ft	799 lbs
3. Accelerate to Supersonic Cruise	627 lbs
4. Supersonic Cruise for 180 nm	4406 lbs
6. Combat with K = 0.15 % of total fuel	1296 lbs
7. Climb to 30000 ft	486 lbs
8. Subsonic Cruise for 180 nm	1028 lbs
9. Hover, Landing, and Reserves	773 lbs
Total Fuel Burn	10368 lbs

=======================
STOVL TWO STAGE MISSION
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	Fuel Burn
PHASE 1	
1. Takeoff	953 lbs
2. Climb to 30000 ft	799 lbs
3. Subsonic Cruise for 240 nm	1370 lbs
4. Landing, Hover, and Reserves	773 lbs
PHASE 1 Fuel Burn	3895 lbs
PHASE 2	
5. Takeoff	953 lbs
6. Climb to 30000 ft	799 lbs
7. Supersonic Dash for 40 nm	1068 lbs
8. Sea Level Combat/Strafe Run at K= 0.15 % W Fuel	1296 lbs
9. Climb to 30000 ft	799 lbs
9.5 Supersonic Cruise for 60 nm	1602 lbs
10. Subsonic Cruise for 240 nm	1370 lbs
11. Hover, Landing, and Reserves	773 lbs
PHASE 2 Fuel Burn	8661 lbs

APPENDIX 3

The purpose of this appendix is to present the stability and control engineering calculations for the material presented in Chapter 9.

Appendix 3: Table of Contents

LONGITUDINAL STABILITY AUGMENTATION
PRELIMINARY CALCULATIONS

TO USE CHARTS IN MIL-F-8785C, n_α MUST BE CALCULATED. THE APPROXIMATION FOR n_α COMES FROM ROSKAM FLIGHT DYNAMICS, PART II

FC 2

$$n_\alpha \approx \frac{C_{L_\alpha}\bar{q}}{W/S} = \frac{(4.438)(1070.36)}{28395/347.9} = 58.20 \ g\text{'s}/\text{RAD}$$

FC 4

$$n_\alpha = \frac{(4.577)(677.79)}{(23980/347.9)} = 45.0 \ g\text{'s}/\text{RAD}$$

FC 7

$$n_\alpha = \frac{(2.378)(1127.01)}{(27239/347.9)} = 34.23 \ g\text{'s}/\text{RAD}$$

∴ THE REQUIREMENTS FROM MIL-F-8785C ARE:

	ω_{SP}, RAD/SEC MIN	ω_{SP}, RAD/SEC MAX	ζ MIN	ζ MAX
FC 2	4.3	16	.30	1.30
FC 4	3.8	14	.35	1.30
FC 7	1.8	12	.30	2.00

$$z = e^{(-\sigma_0 + j\omega)T} = e^{-\sigma_0 T} e^{j\omega T}; \quad \theta = \omega T, \ R = e^{-\sigma_0 T}$$

AS STATED IN CHAPTER 9, T = .01 IS TYPICAL FOR FIGHTERS. WITH T = .01, TO MEET THE MINIMUM HANDLING QUALITIES REQUIREMENTS,

	σ	R	θ (RAD)	ROOTS REAL	ROOTS IMAG.
FC2	1.35229	.9866	.043	.985656	.0424
FC4	1.4198	.9859	.038	.985190	.0375
FC7	.56607	.9944	.018	.994194	.0179

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LONGITUDINAL STABILITY AUGMENTATION (DIGITAL)

FLIGHT CONDITION 2

AIRCRAFT TRANSFER FUNCTION:

$$\frac{\dot{\theta}}{i_H} = \frac{-31903.7s^3 - 69932.8s^2 - 3992.53s}{950.6635s^4 + 4230.158s^3 - 17117.8s^2 - 994.563s - 47.0721}$$

POLES: -7.0006, 2.6088, $-0.0290 \pm 0.0433i$

ZEROS: 0, -2.1333, -0.0587

ACTUATOR: $\frac{10}{s+10}$ (THIS IS THE SLOWEST ACTUATOR THAT PRODUCES A FEASIBLE DESIGN. WHILE FASTER ACTUATORS WOULD WORK AS WELL, THEIR COST WOULD BE HIGHER)

NEW FORWARD PATH:

$$\frac{\dot{\theta}}{i_H} = \frac{-319037s^3 - 699328s^2 - 39925.3s}{950.6635s^5 + 13736.8s^4 + 25183.8s^3 - 172173s^2 - 9992.20s - 470.721}$$

T = .01 (CHOSEN IN CHAPTER 9 TO BE ADEQUATE FOR A FIGHTER)

$$\therefore \frac{\dot{\theta}}{i_H}(z) = \frac{-.0161z^4 + .0325z^3 - .0016z^2 - 0.03z + .0151}{z^5 - 4.8631z^4 + 9.4545z^3 - 9.1853z^2 + 4.4593z - .8655}$$

POLES: 1.1467, $1.0148 \pm .1479i$, $.8434 \pm .0796i$

ZEROS: $-.9600$, $1.0646 \pm .1298i$, $.8493$

COMPENSATOR: $\dfrac{(z-1.1467)(z^2-2.0296z+1.05169)(z^2-1.6868z+.717660)}{(z+.24)(z+.5)(z+.7)(z^2-1.97131z+.973316)}$

FINAL SYSTEM:

$$\frac{\dot{\theta}}{i_H}(z) = \frac{(z+.96)(z-.8493)(z^2-2.1292z+1.15022)}{(z+.7)(z+.5)(z+.24)(z^2-1.97131z+.973316}$$

FOR $K = -0.2$ $\quad x_{COORD} = .9853$, $y_{COORD.} = .0374$

	RESULT	REQUIREMENT (MIN)
ω_{SP}	4.35 RAD/SEC	4.30 RAD/SEC
ζ_{SP}	.38	.35

LONGITUDINAL STAB. AUG. (CON'T)

FLIGHT CONDITION 4

AIRCRAFT TRANSFER FUNCTION:

$$\frac{\dot{\theta}}{i_H} = \frac{-21137s^3 - 37129.5s^2 - 3113.53s}{953.2758s^4 + 3287.973s^3 - 13126s^2 - 992.541s - 264.866}$$

POLES: -5.7955, 2.4251, $-0.0394 \pm 0.1350i$ ZEROS: 0, -1.6683, -0.0883

ACTUATOR: $\frac{10}{s+10}$

NEW FORWARD PATH:

$$\frac{\dot{\theta}}{i_H} = \frac{-211370s^3 - 371295s^2 - 31135.3s}{953.2758s^5 + 12820.7s^4 + 19753.7s^3 - 132253s^2 - 10190.3s - 2648.66}$$

$T = .01$ (SEE EXPLANATION UNDER FLIGHT CONDITION 2)

$$\frac{\dot{\theta}}{i_H}(z) = \frac{-.0107z^4 + .0216z^3 - .001z^2 - .0199z + .0101}{z^5 - 4.8723z^4 + 9.4909z^3 - 9.2391z^2 + 4.4946z - .8742}$$

POLES: 1.1431, $1.0212 \pm .1472i$, $.8434 \pm .0842i$ ZEROS: $-.9611$, 1.1662, $.9068 \pm .1413i$

COMPENSATOR: $\frac{(z-1.1431)(z^2-2.0424z+1.06452)(z^2-1.6868z+.718413)}{(z+.24)(z+.5)(z+.7)(z^2-1.97038z+.972003)}$

FINAL SYSTEM:

$$\frac{\dot{\theta}}{i_H}(z) = \frac{(z+.9611)(z-1.1662)(z^2-1.8136z+.842252)}{(z+.24)(z+.5)(z+.7)(z^2-1.97038z+.972003)}$$

FOR $k=-0.2$, $x_{COOR.} = .9854$ $y_{COOR.} = .0447$

	RESULT	REQUIREMENT (MIN)
ω_{SP}	3.96 RAD/SEC	3.8 RAD/SEC
ζ_{SP}	.37 RAD/SEC	.35

LONGITUDINAL STAB. AUG. (CON'T)

FLIGHT CONDITION 7

AIRCRAFT TRANSFER FUNCTION:

$$\frac{\dot\theta}{i_H} = \frac{-49261.7s^3 - 34436.4s^2 - 413.348s}{1592.449s^4 + 2391.358s^3 + 7482.474s^2 + 85.8631s - 6.9880}$$

POLES: $-0.7449 \pm 2.0315i$, -0.0371, 0.0253

ZEROS: 0, -0.6868, -0.0122

ACTUATOR: $\frac{10}{s+10}$

NEW FORWARD PATH:

$$\frac{\dot\theta}{i_H} = \frac{-492617s^3 - 344364s^2 - 4133.48s}{1592.449s^5 + 18315.8s^4 + 31396.1s^3 + 74910.6s^2 + 851.643s - 69.880}$$

T = .01 (SEE EXPLANATION UNDER FLIGHT CONDITION 2).

$$\frac{\dot\theta}{i_H}(z) = \frac{-.0161z^4 + .0325z^3 - .0016z^2 - .03z + .0151}{z^5 - 4.8631z^4 + 9.4545z^3 - 9.185?z^2 + 4.4593z - .8655}$$

POLES: 1.1467, $1.0148 \pm .1479i$, $.8434 \pm .0796i$

ZEROS: $-.96$, $1.0646 \pm .1298i$, $.8493$

COMPENSATOR: $\dfrac{(z-1.1467)(z^2-2.0296z+1.05169)(z^2-1.6868z+.71766)}{(z+.24)(z+.5)(z+.7)(z^2-1.98839z+.988742)}$

FINAL SYSTEM:

$$\frac{\dot\theta}{i_H}(z) = \frac{(z+.96)(z-.8493)(z^2-2.1292z+1.150211)}{(z+.24)(z+.5)(z+.7)(z^2-1.98839z+.988742)}$$

FOR $K=-1.8$, $-1.98871 = -2x_{COORD.}$, $x = .994355$

$y_{COORD} = [.9887419 - (.994355)^2]^{1/2}$, $y = 1.84391^{-4}$

	RESULT	REQUIREMENT (MIN)
$\therefore\ \omega_{SP}$	1.85 RAD/SEC	1.8 RAD/SEC
ζ_{SP}	.30	.30

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The handling quality requirements of MIL-8785C are set out in APPENDIX B of PART VII.

The LIFT CONFIGURATION is:

* CLASS IV Aircraft
* To be Designed to be LEVEL 1
* Maneuvered in the following FLIGHT PHASES:

FC 2 ⇒ CATEGORY A {Terrain Following (TF)}

FC 4 ⇒ CATEGORY A {Air-to-Air Combat (CO)}

FC 7 ⇒ CATEGORY B {Cruise (CR)}

MAXIMUM ROLL-MODE TIME CONSTANT, T_r, SECONDS

Flight Phase Category	Class	LEVEL 1	LEVEL 2	LEVEL 3
A	I, IV	1.0	1.4	
	II, III	1.4	3.0	
B	ALL	1.4	3.0	10.0
C	I, II-C, IV	1.0	1.4	
	II-L, III	1.4	3.0	

SPIRAL STABILITY - MINIMUM TIME TO DOUBLE AMPLITUDE, T_{2_S}

Flight Phase Category	Level 1	Level 2	Level 3
A and C	12 sec.	8 sec.	4 sec.
B	20 sec.	8 sec.	4 sec.

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For FC 2 (TF) CATEGORY A CLASS IV

$$\frac{\phi}{\delta_A} = \frac{205.0826\, s\left[(s+.5161)^2 + 1.4578^2\right]}{s(s+3.7210)(s+1.1733)\left[(s+.1038)^2 + .5572^2\right]}$$

The roots of the characteristic eqn are:

$$\lambda_1 = -1.1733$$

$$\lambda_2 = -3.7210$$

$$\lambda_{3,4} = -.1038 \pm .5572 j$$

The spiral time constant:

$$T_1 = T_S = \frac{-1}{\lambda_1} = \frac{-1}{-1.1733} = 0.8523 \text{ sec}$$

The roll time constant:

$$T_2 = T_R = \frac{-1}{\lambda_2} = \frac{-1}{-3.7210} = 0.2687 \text{ sec}$$

The time to double the amplitude in the spiral mode is calculated using EQN (B6) of p 543, Flight Dyn Part I:

$$T_{2_S} = \frac{\ln 2}{\frac{1}{T_S}} \qquad \text{EQN (B6)}$$

$$\therefore T_{2_S} = \frac{\ln 2}{\frac{1}{.8523}} = .5908 \text{ sec}$$

∴ FC 2 satisfies *Level 1* for the roll-mode time constant, but the time to double the amplitude of the spiral mode is too quick. Currently, $T_{2_S} = .5908$ sec, and this doesn't even satisfy Level 3. To be Level 1 $T_{2_S} \geq 12$ sec

FC 4 (CO) CATEGORY A CLASS IV

$$\frac{\phi}{\delta_A} = \frac{412.576\, s\left[(s+.3915)^2 + 2.4554^2\right]}{s(s+2.0456)(s+.0569)\left[(s+1.5808)^2 + 3.8849^2\right]}$$

The characteristic roots are :

$$\lambda_1 = -.0569$$

$$\lambda_2 = -2.0456$$

$$\lambda_{3,4} = -1.5808 \pm 3.8849j$$

The spiral time constant :

$$T_1 = T_S = \frac{-1}{\lambda_1} = \frac{-1}{-.0569} = 17.57 \text{ secs}$$

$$\therefore T_{2_S} = \frac{\ln 2}{\frac{1}{T_S}} = \frac{\ln 2}{\frac{1}{17.57}}$$

$$T_{2_S} = 12.18 \text{ secs}$$

The roll time constant :

$$T_2 = T_R = \frac{-1}{-2.0456} = 0.4889 \text{ secs.}$$

∴ In FC 4 the handling qualities satisfy Level 1 for the roll mode and for the spiral time to double amplitude.

$$\therefore K_\phi = 0$$

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FC 7 (CR) CATEGORY B CLASS IV

$$\frac{\phi}{\delta_A}(s) = \frac{126.1512\, s\,(s+.9757)(s-.1348)}{s\,(s+3.242)(s+2.2447)(s-.6265)(s-.2461)}$$

The characteristic roots are :

$$\lambda_1 = +.6265$$

$$\lambda_2 = +.2461$$

$$\lambda_3 = -2.2447$$

$$\lambda_4 = -3.2420$$

The spiral time constant :

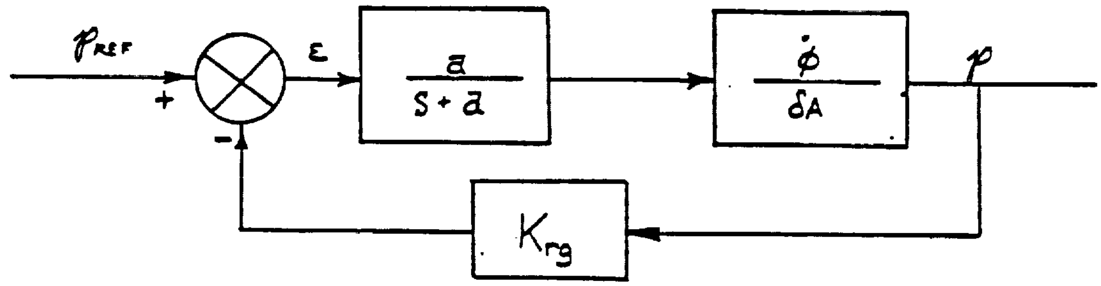
$$T_1 = T_s = \frac{-1}{-2.2447} = .4455 \text{ secs}$$

$$\therefore T_{2s} = \frac{\ln 2}{\frac{1}{T_s}} = \frac{\ln 2}{\frac{1}{.4455}} = .3088 \text{ sec}$$

The roll time constant:

$$T_2 = T_R = \frac{-1}{-3.242} = .3085 \text{ sec}$$

Thus, in FC 7, the handling qualities satisfy Level 1, as the roll mode {max 1.4 secs}, and the spiral time to double amplitude is too small as it must be ≥ 20 sec for FC 7.



$$\frac{\dot{\phi}}{\delta_A}(s) = s\,\frac{\phi}{\delta_A}(s)$$

$\frac{\phi}{\delta_A}(s)$ has been determined for flight conditions 2, 4, 7.

The open loop transfer function is:

$$G(s) = K_{rg}\,\frac{a}{s+a}\,\frac{\dot{\phi}}{\delta_A}$$

A servo of $10/(s+10)$ is examined first, than if this is not sufficient a higher frequency servo will be investigated.

FLIGHT CONDITION 2:

$$G(s) = K_{rg}\,\frac{10}{(s+10)}(s)\,\frac{205.0826\,s\left[(s+.5161)^2 + 1.4578^2\right]}{s(s+3.721)(s+1.1733)\left[(s+.1038)^2 + .5572^2\right]}$$

$$= \frac{K_{rg}\;2051\,s\left[(s+.5161)^2 + 1.4578^2\right]}{(s+10)(s+3.721)(s+1.1733)\left[(s+.1038)^2 + .5572^2\right]}$$

The total pulsed transfer function, $G(z)$:

$$G(z) = D_c(z)\,\mathcal{Z}\left\{[ZOH]\left[\tfrac{10}{s+10}\right]\left[\tfrac{\dot{\phi}}{\delta_A}\right]\right\}(z)$$

$$= D_c(z)\,\mathcal{Z}\left\{[ZOH][H(s)]\right\}(z)$$

$$H(s) = \frac{2051s^3 + 2117s^2 + 4905s}{s^5 + 15.1s^4 + 56.7s^3 + 59.51s^2 + 26.19s + 14.02}$$

Using PC-MATLAB to perform the z transform with a sample time of T = .01/sec.

$$G(z) = \frac{.0979z^4 - .1592z^3 + .0124z^2 + .1814z - .0924}{z^5 - 4.855z^4 + 9.42z^3 - 9.14z^2 + 4.434z - .8598}$$

$$G(z) = \frac{(z + .9542)(z - 1)\left[(z - .9947)^2 + .0145^2\right]}{(z - .9048)(z - .9635)(z - .9883)\left[(z - .9989)^2 + .0056^2\right]}$$

for FC 2 the time to double amplitude must be greater than or equal to 12 sec. Thus the spiral time constant must be:

$$T_{2_S} = \frac{\ln 2}{\frac{1}{T_s}} \geq 12$$

$$T_s \geq \frac{12}{\ln 2} = 17.3 \qquad T_s = \frac{-1}{\lambda_1}$$

thus the spiral root must be:

$$\lambda_1 \leq -.0578 \quad \therefore R = e^{\lambda_1 T} = .9994$$

The z plane root that is the spiral is:

$$(z - .9883)$$

at a gain of .25 the spiral root is at .9997 and the roll root is at .9274.

$$\lambda_1 = \ln .9997/.01 = -.030$$

$$T_s = \frac{-1}{\lambda_1} = 33.3 \qquad T_{2_s} = \frac{\ln 2}{\frac{1}{33.3}} = 23.1 \text{ sec} \geq 12 \text{ sec} \checkmark$$

$$\lambda_2 = \ln .9274/.01 = -7.537$$

check $T_r = \frac{-1}{\lambda_2} = .1327$ sec which is ≤ 1.0 ✓

$$\therefore K_\phi = 0.25$$

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FLIGHT CONDITION 4:

$$G(s) = K_{rg}\frac{10}{(s+10)}(s)\frac{412.58\, s\left[(s+.3915)^2 + 2.4554^2\right]}{s(s+2.0456)(s+.0569)\left[(s+1.5808)^2+3.885^2\right]}$$

$$= \frac{K_{ra}\ 4126\, s\,(s^2 + .7835 s + 6.1823)}{(s+10)(s+2.0456)(s+.0569)(s^2+3.162 s + 17.59)}$$

The total pulsed transfer function, $G(z)$:

$$G(z) = D_c(z)\, \mathcal{Z}\left\{[ZOH]\left[\tfrac{10}{s+10}\right]\left[\tfrac{\phi}{\delta_A}\right]\right\}(z)$$

$$= D_c(z)\, \mathcal{Z}\left\{[ZOH][H(s)]\right\}(z)$$

$$H(s) = \frac{4126 s^3 + 3233 s^2 + 25{,}508 s}{s^5 + 15.26 s^4 + 77 s^3 + 280.5 s^2 + 375.5 s + 20.48}$$

Using PC-MATLAB to perform the z transform with a sample time of $T = .01/\text{sec}$.

$$G(z) = \frac{.0003 z^4 + .0006 z^3 - .0019 z^2 + .0006 z + .0003}{z^5 - 4.855 z^4 + 9.424 z^3 - 9.143 z^2 + 4.434 z - .860}$$

$$G(z) = \frac{(z+3.791)(z-1.264)(z-.791)(z+.2638)}{(z-.8092)\left[(z-1.027)^2 + .0905^2\right]\left[(z-.920)^2 + (.1469)^2\right]}$$

for FC 4 Level 1 is already satisfied.

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FLIGHT CONDITION 7:

$$G(s) = K_{rg}\frac{10}{s+10}\, s\ \frac{126.1512\, s(s+.9757)(s-.1348)}{s(s+3.242)(s+2.2447)(s-.6265)(s-.2461)}$$

$$= \frac{K_{rg}\ 1262\, s(s^2+.8409s-.1315)}{(s+10)(s+3.242)(s+2.2447)(s-.6265)(s-.2461)}$$

The total pulsed transfer function, $G(z)$:

$$G(z) = D_c(z)\, \mathcal{Z}\left\{[ZOH]\left[\frac{10}{s+10}\right]\left[\frac{\phi}{\delta_a}\right]\right\}(z)$$

$$= D_c(z)\, \mathcal{Z}\left\{[ZOH][H(s)]\right\}(z)$$

$$H(s) = \frac{1262s^3+1061s^2-166s}{s^5+14.60s^4+48.76s^3+20.91s^2-53.88s+11.21}$$

Using PC-MATLAB to perform the z transform with a sample time of $T = .01/\text{sec}$.

$$G(z) = \frac{.0603z^4-.1228z^3+.0076z^2+.112z-.0571}{z^5-4.860z^4+9.443z^3-9.171z^2+4.452z-.8642}$$

$$G(z) = \frac{(z+.9552)(z-1.0013)(z-1)(z-.9903)}{(z-.9050)(z-.9680)(z-.9779)(z-1.0025)(z-1.0063)}$$

for FC 7 the Level 1 handling requirements are:

$$T_r \le 1.4 \text{ seconds}\ ,\quad T_{2_S} \ge 20 \text{ sec}$$

uncompensated: $T_r = .3085$ sec, $T_{2_S} = .3088$ sec

so the spiral time to double must be increased without making the roll time constant greater than 1.4 seconds.

keep $0 < \lambda_2 < .9929$ so that $T_r > 1.4$ sec

To make $T_{2_S} \geq 20$ sec

$$\frac{\ln 2}{\frac{1}{T_S}} \geq 20 \text{ sec}$$

$$T_S \geq \frac{20 \text{ sec}}{\ln 2} = 28.85$$

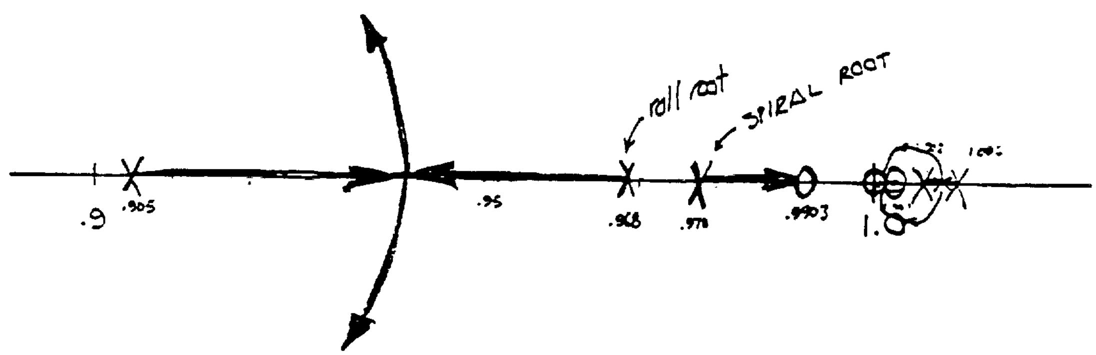
thus the spiral root must be :

$\lambda_1 \leq -.0347 \quad \therefore \quad R = e^{\lambda_1 T} = .9997$

The z plane root that is the spiral is :

$$(z - .9779)$$

sketching the roots near 1 :



Thus the spiral root can not make it to the desired .9997, so a compensator must be implemented to cancel the zero with a pole and place the zero at .99999 so that the root can get to .9997. Thus the compensator is :

$$D_c(z) = \frac{(z - 1.0009)}{(z - .9903)}$$

The compensated equation thus becomes :

$$G(z) = \frac{(z+.9552)(z-1.0013)(z-1)\cancel{(z-.9903)}(z-1.0009)}{(z-.9050)(z-.9680)(z-.9779)(z-1.0025)(z-1.0063)\cancel{(z-.9903)}}$$

$$G(z) = \frac{z^4 - 2.046z^3 + .1357z^2 + 1.867z - .9563}{z^5 - 4.86z^4 + 3.728z^3 - 3.42z^2 + 4.452z - .8641}$$

from PC-Matlab, at a gain of 14, the spiral root location is .9997, and the roll root is at .4766 ± 1.2243 i.

thus $\lambda_1 = \ln .9997/.01 = -.030$

$T_s = \frac{-1}{\lambda_1} = 33.3 \qquad T_{2_s} = \frac{\ln 2}{\frac{1}{T_s}} = 23.1 \text{ sec} \geq 20 \text{ sec}$

$$\lambda_2 = \frac{\ln .4766}{.01} = -74.1$$

$$T_r = \frac{-1}{\lambda_2} = .0135 \text{ sec} \leq 1.4 \text{ sec}$$

$\therefore$ IN FC 7 $K_\phi = 14$

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new compensator :

$$D_c(z) = \frac{(z-1.0009)}{(z-.9903)}\frac{(z-.9050)}{(z+.10)}$$

$$= \frac{z^2 - 1.9059z + .9132}{z^2 - .8903z - .09903}$$

after convoluting on PCMATLAB file → ejhawk

nt = 0 .0603 -.2377 .2966 -.0146 -.2636 .2111 -.0521

dt = 1 -5.7499 13.6705 -17.0972 11.6822 -3.9196 .3285 .0856

∴ K = [0 : .01 : 1]
K(:)
j = rlocus(nt, dt, K)

New eqn : $G(z) = \frac{(z+.9552)(z-1.0013)(z-1)(z-.9903)(z-1.0009)(z-.905)}{(z-.905)(z-.9680)(z-.9779)(z-1.0025)(z-1.0063)(z-...}$

$$= \frac{z^4 - 2.046z^3 + .1357z^2 + 1.867z - .9563}{z^4 - 3.96z^3 + 5.795z^2 + 1.959z \ldots}$$

1 -4.021 6.062 -4.06 1.021

1 -4.04 6.12 -4.12 1.04

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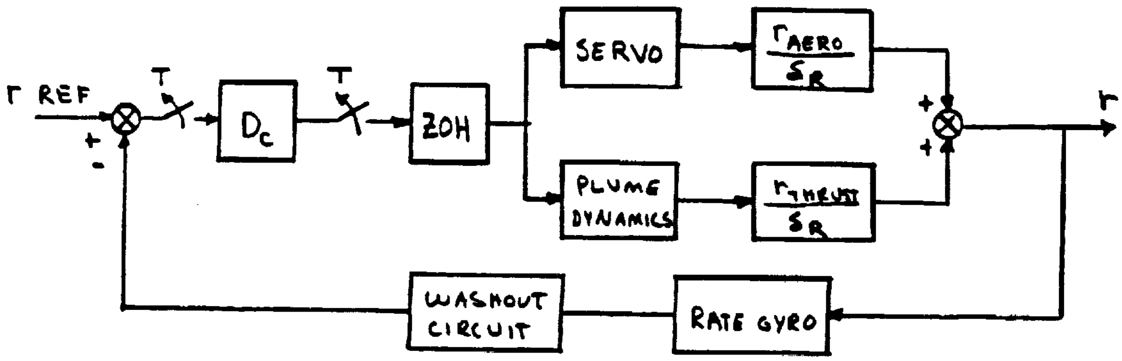

COMPONENTS:

SERVO T.F. : $\frac{10}{s+10}$

BOB KISSLINGER (McAIR STOL DEMONSTRATOR): THE THRUST PLUME REACTS INSTANTANEOUSLY. THE PLUME WILL THERFORE REACT AT THE SAME FREQUENCY AS THE SERVO:

PLUME DYNAMICS T.F. : $\frac{10}{s+10}$

THE WASHOUT CIRCUIT PREVENTS INTERFERENCE FROM THE YAW DAMPER WHEN THE PILOT INITIATES TURN ENTRIES AND PERFORMS STEADY-STATE TURNS.

WASHOUT CIRCUIT T.F. : $\frac{4s}{4s+1} = \frac{s}{s+.25}$

D_c IS THE DIGITAL COMPENSATOR IF REQUIRED.

THE r_{AERO}/S_R AND r_{THRUST}/S_R ARE DETERMIND FROM THE DIMENSIONAL DERIVATIVES FOR EACH FLIGHT CONDITION.

FC 2

$$\frac{r_{thrust}}{\delta_R} = \frac{-2056.35s^3 - 8945.01s^2 - 1790.46s - 4054.13}{948.24s^4 + 4837.81s^3 + 5407.88s^2 + 2350.14s + 1329.72}$$

$$\frac{r_{AERO}}{\delta_R} = \frac{-1576.42s^3 - 6856.05s^2 - 1372.31s - 3107.32}{948.24s^4 + 4837.81s^3 + 5407.88s^2 + 2350.14s + 1329.72}$$

DEVELOPMENT OF TOTAL PULSED T.F., $G(z)$

$$G(z) = D_c(z)\,\mathcal{Z}\left\{[ZOH]\left[\frac{10}{s+10}\right]\left[\left(r_{AERO}/\delta_R + r_{thrust}/\delta_R\right)\right]\left[\frac{s}{s+.25}\right]\right\}(z) =$$

$$= D_c(z)\,\mathcal{Z}\left\{[ZOH][H(s)]\right\}(z)$$

$$H(s) = \frac{-36327.7s^4 - 158010.6s^3 - 31627.7s^2 - 71614.5s}{948.24s^6 + 14557.27s^5 + 57366.03s^4 + 69875.44s^3 + 38938.46s^2 + 19505.0s + 3324.3}$$

USING PC-MATLAB TO PERFORM Z-TRANSFORM w/ T = .01/SEC.

$$G(z) = -.0018\;D_c(z)\,\frac{(z+.964)(z-.9584)(z-1)\left((z-.9995)^2 + .0068^2\right)}{(z-.9975)(z-.9883)(z-.9635)(z-.9048)\left((z-.9989)^2 + .0056^2\right)}$$

DUTCH ROLL: $\left((z-.9989)^2 + .0056^2\right)$

$$z = Re^{\theta i} = e^{\sigma T}e^{\omega T i}$$

$$R = \sqrt{.9981^2 + .0056^2} = .9989 = e^{\sigma T}$$

$$-.0011 = \sigma T = \zeta\omega T$$

$$\zeta\omega = -.0011/.01 = -.1085$$

$$\theta = \tan^{-1}(.0056/.9989) = .32° = .0056\text{ RAD} = \omega T$$

$$\omega_D = .0056/.01 = .56\text{ RAD/SEC}$$

$$\zeta_D = .1085/.56 = .194$$

MIL-F-8785 C REQUIREMENTS:

$\zeta_D \geq .40$

$\omega_{n_D} \geq 1.0$ RAD/SEC

FOR INERTIA COUPLING CRITERIA, THE FOLLOWING DUTCH ROLL VALUES ARE DESIRED AS DETERMINED BY P.B.:

$$\zeta_D > .48 \qquad \omega_{n_D} > 2.4 \text{ RAD/SEC}$$

FOR THESE VALUES THE ROOT LOCATION IS:

$$R = e^{-(.48)(2.4)(.01)} = .98855$$

$$\theta = 2.4(.01) = .024 \text{ RAD} = 1.375°$$

$$z = .98855\left(\cos(1.375) + \sin(1.375)j\right) = .9883 \pm .0237j$$

$$D_c = \frac{(z-.9989)^2 + .0056^2}{(z-.9883)^2 + .0237^2} = \frac{z^2 - 1.9978z + .9978}{z^2 - 1.9766z + .9773}$$

THIS COMPENSATOR PLACES NEW POLES AT THE DESIRED LOCATION.

THE MIL-F-8785C REQUIREMENTS ARE SATISFIED FOR A GAIN RANGE OF 0 TO -1.5. THE INERTIA CRITERIA IS NOT SATISFIED.

$K = -1.5 \qquad \omega_n = 1.0 \qquad \zeta_D = .45$

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FC 4

$$\frac{r_{thrust}}{\delta_R} = \frac{-3903.82s^3 - 17701.3s^2 - 2434.62s - 19408.9}{894.17s^4 + 4708s^3 + 21777.37s^2 + 33400.15s + 1830.89}$$

$$\frac{r_{AERO}}{\delta_R} = \frac{-942.33s^3 - 4265.1s^2 - 587.36s - 4670.5}{894.17s^4 + 4708s^3 + 21777.37s^2 + 33400.15s + 1830.89}$$

DEVELOPMENT OF TOTAL PULSED T.F., G(z)

$$G(z) = KD_c(z)\, \mathcal{Z}\{[ZOH][H(s)]\}(z)$$

$$H(s) = \frac{-48461.5s^4 - 219664s^3 - 30219.8s^2 - 240794s}{894.17s^6 + 13873.24s^5 + 72269.77s^4 + 268388.19s^3 + 398625.82s^2 + 102266.92s + 4577.2}$$

USING PC-MATLAB TO PERFORM Z-TRANSFORM w/ T=.01/SEC.

$$G(z) = -.0026KD_c(z)\frac{(z+.9641)(z-1)(z-.9548)((z-1.004)^2+.0104^2)}{(z-.9975)(z-.9994)(z-.9797)(z-.9048)((z-.9836)^2+.0382^2)}$$

DUTCH ROLL: $((z-.9836)^2+.0382^2)$

$$z = Re^{\theta j} = e^{\sigma T}e^{\omega T j}$$

$$R = \sqrt{.9836^2+.0382^2} = .9843 = e^{\sigma T}$$

$$-.0158 = \sigma T = \zeta\omega T$$

$$\zeta\omega = -1.5825$$

$$\theta = \tan^{-1}(.0382/.9836) = 2.22^\circ = .039\ \text{RAD} = \omega T$$

$$\omega_D = .039/.01 = 3.88\ \text{RAD/SEC}$$

$$\zeta_D = 1.5825/3.88 = .408$$

MIL-F-8785C REQUIREMENTS

$\zeta_D \geq .19$

$\omega_D \geq 1.0$ RAD/SEC

$\zeta_D\omega_D \geq .35$

REQUIREMENTS ARE MET

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FC 7

$$\frac{r_{thrust}}{\delta_R} = \frac{-6809.42s^3 - 26866s^2 - 3214.15s - 10535.5}{1591.69s^4 + 7344.05s^3 + 4207.7s^2 - 8761.19s + 1786.34}$$

$$\frac{r_{AERO}}{\delta_R} = \frac{-2975.53s^3 - 11733s^2 - 1402.29s - 4664.48}{1591.69s^4 + 7344.05s^3 + 4207.7s^2 - 8761.19s + 1786.34}$$

DEVELOPMENT OF TOTAL PULSED T.F., G(z)

$$G(z) = KD_c(z)\,\mathcal{Z}\{[ZOH][H(s)]\}(z)$$

$$H(s) = \frac{-97849.5s^4 - 385990s^3 - 46164.4s^2 - 151999.8s}{1591.69s^6 + 23658.87s^5 + 83463.44s^4 + 52727.86s^3 - 77496.61s^2 - 3592.99s + 4465.85}$$

USING PC-MATLAB TO PERFORM z-TRANSFORM w/ T = .01/sec.

$$G(z) = -.0030\,KD_c(z)\,\frac{(z+.9643)(z-.9615)(z-1)((z-.9999)^2+.0063^2)}{(z-1.0063)(z-1.0025)(z-.9975)(z-.9778)(z-.9681)(z-.9048)}$$

WASH SPIRAL [illegible] SERVO

DUTCH ROLL: $(z-1.0063)(z-1.0025)$ → UNSTABLE

MIL-F-8785C REQUIREMENTS

$\zeta_D \geq .08$

$\omega_{n_D} \geq .40$ RAD/SEC

$\zeta_D \omega_{n_D} \geq .15$

FOR INERTIA COUPLING CRITERIA, THE FOLLOWING DUTCH ROLL VALUES ARE DESIRED AS DETERMINED BY P.B.:

$\zeta_D > .096 \qquad \omega_n > 1.62$

FOR THESE VALUES THE ROOT LOCATION IS:

$$R = e^{-(.096)(1.62)(.01)} = .9985$$

$$\theta = 1.62(.01) = .0162 \text{ RAD} = .928^\circ$$

$$z = .9985(\cos(.928) + \sin(.928)j) = .9984 + .0162j$$

COMPENSATOR TO PLACE POLES AT THE DESIRED LOCATION:

$$D_c = \frac{(z-1.0063)(z-1.0025)}{(z-.9984)^2 + .0162^2} = \frac{z^2 - 2.0088z + 1.0088}{z^2 - 1.9968z + .99707}$$

Following the procedure of pages 59-61 of PART VII:

Step 1: MIL-F-8785C applies

max roll mode time constants are found in Table VII, p 297 of Appendix B, PART VII

Step 2: Flight conditions and configurations have been selected.

Step 3: M, $\bar{q}$ and I_{xx_S} is known for each f.c.

Step 4: C_{ℓ_p} is known for each f.c.

Step 5: Determine roll control power derivative due to lateral cockpit control, $C_{\ell_{\delta_{cpt}}}$

The Monarch uses ailerons for roll control.

$$C_{\ell_{\delta_{cpt}}} \delta_{cpt_{max}} = C_{\ell_{\delta_a}} \delta_{a_{max}} + C_{\ell_{\delta_s}} \delta_{s_{max}} \quad \text{EQN (2.63)}$$

$C_{\ell_{\delta_a}}$ = aileron control power derivative

$\delta_{a_{max}}$ is max available aileron deflection $= 25° = .436$ rad

The MONARCH does not have spoilers,

$\therefore C_{\ell_{\delta_s}} = \delta_s = 0$

FC 2

$$C_{\ell_{\delta_{cpt}}} \delta_{cpt_{max}} = .252 \text{ rad}^{-1} (.436 \text{ rad}) = .10996$$

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$\delta_{cpt_{max}}$ is the max available deflection of the lateral cockpit controller in radians.

δ_{cpt} follows from the cockpit design, and is gotten from CHARLIE to be:

TABLE 2.3, PART III

$$\delta_{cpt} \cong 15^\circ = .262 \text{ rad}$$

$$\therefore C_{l_{\delta_{cpt}}} = .10996 / .262 \text{ rad}$$

$$C_{l_{\delta_{cpt}}} = .420 \text{ rad}^{-1} \quad .00733 \text{ deg}^{-1}$$

Step 6: Determine roll angle performance capability of the airplane using EQN (2.66), PART VII:

$$\phi(t) = \{(-L_{\delta_{cpt}}\delta_{cpt})t/L_p\} + [(L_{\delta_{cpt}}\delta_{cpt})/\{(L_p)^2\}](e^{L_p t} - 1) \qquad \text{EQN (2.66)}$$

$\phi(t)$ is the bank angle reached at t seconds after full movement of lateral cockpit controller, δ_{cpt}.

t is the time elapsed from full movement of the cockpit controller, δ_{cpt} to the time at which the bank angle is to be measured.

FC 2A (TF) category TABLE IXb, Appendix E, p 300 PART VII, Speed Range M

$\phi(t) = 90^\circ$ in 1.3 seconds Level 1

$$L_p = -4.0978$$

$$L_{\delta_{cpt}} = (C_{l_{\delta_{cpt}}}\bar{q} S b)/I_{xx_S} \qquad \text{EQN (2.62)}$$

$$= .420(1067)(348)(33.67)/15370 = 341.6$$

$$\phi(t) = \{(-341.6(.262)(1.3)/(-4.0978)\} +$$

$$+ [(341.6(.262)/\{-4.0978\}^2](e^{-4.0978(1.3)} - 1)$$

$28.4 + 5.33(-.995)$

$$\phi(t) = 23.1 \quad 1.323$$

Step 7 : Calculate the airplane roll time constant, τ_R from EQN (2.67) :

$$\tau_R = \frac{-1}{L_p} \qquad \text{EQN (2.67)}$$

$$FC\ 2 \quad \tau_R = \frac{-1}{-4.0978} = +.244$$

Max $T_R \leq 1.00$ second ✓

	FC	M V ← speed range	$\bar{q}$ [psf]	I_{xx_s} [slug $\times$ ft²]	$L_{\delta_{apt}}$	$\phi(t)$	τ_R
Category A (TF)	2	.65 M	1067	15370	341.6	18.7	+.244
Category A (CO)	4	.90 M	678	10631	313.8		
Category B (CR)	7	1.60 H	1127	10152	546.3		

REQUIREMENTS : LEVEL 1

FC 2 90° in 1.3 seconds

FC 4 360° rolls : 90° in 1 sec 180° in 1.6 sec 360° in 2.8 sec
Flight phase CO : 90° in 1.1 sec 180° in 2.2 sec

FC 7 50° in 1.1 second

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INERTIA COUPLING CALCULATIONS

FROM ROSKAM FLIGHT DYNAMICS, PART II:

THE BOUNDARIES OF THE INERTIA COUPLING DIAGRAM ARE GIVEN BY:

$$\frac{\omega_D}{P_1} = 1 - \frac{\zeta_D \zeta_{SP}}{(\omega_{SP}/P_1 - 1)}, \quad 0 \leq \frac{\omega_{SP}}{P_1} \leq 1$$

$$\frac{\omega_{SP}}{P_1} = 1 - \frac{\zeta_D \zeta_{SP}}{\left(\frac{\omega_D}{P_1} - 1\right)}, \quad 0 \leq \frac{\omega_D}{P_1} \leq 1$$

THE SLOPE OF THE LINE THROUGH THE DIAGRAM IS THE RATIO OF DUTCH ROLL FREQUENCY TO SHORT PERIOD FREQUENCY.

THE SPREADSHEETS THAT FOLLOW SHOW THE CALCULATIONS FOR THE INERTIA COUPLING DIAGRAMS FOR FLIGHT CONDITIONS 2, 4, AND 7.

AE 622 5-3-90 PAUL BORCHERS

INERTIA COUPLING CALCULATIONS, FC 2 (OPEN LOOP)

Flight Condition 2 M = 0.85 h = 100 ft.
Open Loop

Short Period Frequency	0.050	rad/sec
Short Period Damping Ratio	0.160	--
Dutch Roll Frequency	0.560	rad/sec
Dutch Roll Damping Ratio	0.180	--

Omega S. P. /Roll Rate	Omega D/Roll Rate	Omega D/Omega S. P.	P1 (deg/sec)
0.01	1.0290909090909	0.11	286.478228
0.05	1.0303157894737	0.56	57.2956455
0.10	1.032	1.12	28.6478228
0.15	1.0338823529412	1.68	19.0985485
0.20	1.036	2.24	14.3239114
0.25	1.0384	2.80	11.4591291
0.30	1.0411428571429	3.36	9.54927426
0.35	1.0443076923077	3.92	8.18509222
0.40	1.048	4.48	7.16195569
0.45	1.0523636363636	5.04	6.36618284
0.50	1.0576	5.60	5.72956455
0.55	1.064	6.16	5.20869505
0.60	1.072	6.72	4.77463713
0.65	1.0822857142857	7.28	4.40735735
0.70	1.096	7.84	4.09254611
0.75	1.1152	8.40	3.8197097
0.80	1.144	8.96	3.58097785
0.85	1.192	9.52	3.37033209
0.90	1.288	10.08	3.18309142
0.91	1.32	10.19	3.14811239
0.92	1.36	10.30	3.11389378
0.93	1.4114285714286	10.42	3.08041105
0.94	1.48	10.53	3.04764072
0.95	1.576	10.64	3.01556029
0.96	1.72	10.75	2.9841482
0.97	1.96	10.86	2.95338379
0.98	2.44	10.98	2.92324722
0.99	3.8800000000001	11.09	2.89371947
1.0290909090909	0.01	11.53	2.78379903
1.0303157894737	0.05	11.54	2.78048954
1.032	0.10	11.56	2.77595182
1.0338823529412	0.15	11.58	2.77089774
1.036	0.20	11.60	2.76523386
1.0384	0.25	11.63	2.75884272
1.0411428571429	0.30	11.66	2.75157463
1.0443076923077	0.35	11.70	2.74323583
1.048	0.40	11.74	2.73357087
1.0523636363636	0.45	11.79	2.7222361
1.0576	0.50	11.85	2.70875783
1.064	0.55	11.92	2.69246455

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FLIGHT CONDITION 2, OPEN LOOP, CON'T

1.072	0.60	12.01	2.67237153
1.0822857142857	0.65	12.12	2.64697412
1.096	0.70	12.28	2.61385244
1.1152	0.75	12.49	2.56885068
1.144	0.80	12.81	2.50418031
1.192	0.85	13.35	2.40334084
1.288	0.90	14.43	2.22420984
1.32	0.91	14.78	2.1702896
1.36	0.92	15.23	2.10645756
1.4114285714286	0.93	15.81	2.02970404
1.48	0.94	16.58	1.9356637
1.576	0.95	17.65	1.81775525
1.72	0.96	19.26	1.66557109
1.96	0.97	21.95	1.46162361
2.44	0.98	27.33	1.1740911
3.8800000000001	0.99	43.46	0.73834595

AE 622 5-3-90 PAUL BORCHERS

INERTIA COUPLING CALCULATIONS, FC 2 (CLOSED LOOP)

Flight Condition 2 M = 0.85 h = 100 ft.

Short Period Frequency	4.300	rad/sec
Short Period Damping Ratio	0.350	--
Dutch Roll Frequency	2.400	rad/sec
Dutch Roll Damping Ratio	0.480	--

Omega S. P. /Roll Rate	Omega D/Roll Rate	Omega D/Omega S. P.	P1 (deg/sec)
0.01	1.169696969697	0.01	24637.1276
0.05	1.1768421052632	0.03	4927.42552
0.10	1.1866666666667	0.06	2463.71276
0.15	1.1976470588235	0.08	1642.47517
0.20	1.21	0.11	1231.85638
0.25	1.224	0.14	985.485103
0.30	1.24	0.17	821.237586
0.35	1.2584615384615	0.20	703.917931
0.40	1.28	0.22	615.928189
0.45	1.3054545454545	0.25	547.491724
0.50	1.336	0.28	492.742552
0.55	1.3733333333333	0.31	447.947774
0.60	1.42	0.33	410.618793
0.65	1.48	0.36	379.032732
0.70	1.56	0.39	351.958965
0.75	1.672	0.42	328.495034
0.80	1.84	0.45	307.964095
0.85	2.12	0.47	289.84856
0.90	2.68	0.50	273.745862
0.91	2.8666666666667	0.51	270.737666
0.92	3.1	0.51	267.794865
0.93	3.4	0.52	264.91535
0.94	3.8	0.52	262.097102
0.95	4.36	0.53	259.338185
0.96	5.2	0.54	256.636746
0.97	6.6000000000001	0.54	253.991006
0.98	9.4000000000001	0.55	251.399261
0.99	17.800000000001	0.55	248.859875
1.169696969697	0.01	0.65	210.628293
1.1768421052632	0.05	0.66	209.349474
1.1866666666667	0.10	0.66	207.616244
1.1976470588235	0.15	0.67	205.712755
1.21	0.20	0.68	203.612625
1.224	0.25	0.68	201.283722
1.24	0.30	0.69	198.686513
1.2584615384615	0.35	0.70	195.771796
1.28	0.40	0.71	192.477559
1.3054545454545	0.45	0.73	188.724515
1.336	0.50	0.75	184.409638
1.3733333333333	0.55	0.77	179.39656

FLIGHT CONDITION 2, CON'T

ω_{SP}/P_1	ω_D/P_1	ω_D/ω_{SP}	P_1 (DEG/SEC)
1.42	0.60	0.79	173.500898
1.48	0.65	0.83	166.467078
1.56	0.70	0.87	157.930305
1.672	0.75	0.93	147.351241
1.84	0.80	1.03	133.897432
2.12	0.85	1.18	116.212866
2.68	0.90	1.50	91.9295805
2.8666666666667	0.91	1.60	85.9434683
3.1	0.92	1.73	79.4746051
3.4	0.93	1.90	72.4621399
3.8	0.94	2.12	64.8345463
4.36	0.95	2.43	56.5071733
5.2	0.96	2.90	47.3790915
6.600000000001	0.97	3.68	37.3289812
9.400000000001	0.98	5.25	26.2097102
17.800000000001	0.99	9.93	13.8410829

AE 622 5-3-90 PAUL BORCHERS

INERTIA COUPLING CALCULATIONS, FC 4 (OPEN LOOP)

Flight Condition 4 M = 0.90 h = 15000 ft.
Open Loop

Short Period Frequency	0.140	rad/sec
Short Period Damping Ratio	0.280	--
Dutch Roll Frequency	4.190	rad/sec
Dutch Roll Damping Ratio	0.370	--

Omega S. P. /Roll Rate	Omega D/Roll Rate	Omega D/Omega S. P.	P1 (deg/sec)
0.01	1.1046464646465	0.30	802.139037
0.05	1.1090526315789	1.50	160.427807
0.10	1.1151111111111	2.99	80.2139037
0.15	1.1218823529412	4.49	53.4759358
0.20	1.1295	5.99	40.1069519
0.25	1.1381333333333	7.48	32.0855615
0.30	1.148	8.98	26.7379679
0.35	1.1593846153846	10.48	22.9182582
0.40	1.1726666666667	11.97	20.0534759
0.45	1.1883636363636	13.47	17.8253119
0.50	1.2072	14.96	16.0427807
0.55	1.2302222222222	16.46	14.5843461
0.60	1.259	17.96	13.368984
0.65	1.296	19.45	12.3406006
0.70	1.3453333333333	20.95	11.4591291
0.75	1.4144	22.45	10.6951872
0.80	1.518	23.94	10.026738
0.85	1.6906666666667	25.44	9.43692985
0.90	2.036	26.94	8.91265597
0.91	2.1511111111111	27.24	8.8147147
0.92	2.295	27.53	8.71890258
0.93	2.48	27.83	8.62515094
0.94	2.7266666666667	28.13	8.53339402
0.95	3.072	28.43	8.44356882
0.96	3.59	28.73	8.35561497
0.97	4.4533333333334	29.03	8.26947461
0.98	6.1800000000001	29.33	8.18509222
0.99	11.36	29.63	8.10241452
1.1046464646465	0.01	33.06	7.26150006
1.1090526315789	0.05	33.19	7.23265077
1.1151111111111	0.10	33.37	7.19335526
1.1218823529412	0.15	33.58	7.14993899
1.1295	0.20	33.80	7.10171791
1.1381333333333	0.25	34.06	7.04784768
1.148	0.30	34.36	6.98727385
1.1593846153846	0.35	34.70	6.91866208
1.1726666666667	0.40	35.10	6.84029878
1.1883636363636	0.45	35.57	6.749946
1.2072	0.50	36.13	6.64462423
1.2302222222222	0.55	36.82	6.52027758

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FLIGHT CONDITION 4, OPEN LOOP, CON'T

1.259	0.60	37.68	6.37123938
1.296	0.65	38.79	6.18934442
1.3453333333333	0.70	40.26	5.96238135
1.4144	0.75	42.33	5.67123188
1.518	0.80	45.43	5.28418338
1.6906666666667	0.85	50.60	4.74451323
2.036	0.90	60.93	3.93977916
2.1511111111111	0.91	64.38	3.72895214
2.295	0.92	68.69	3.4951592
2.48	0.93	74.22	3.2344316
2.7266666666667	0.94	81.61	2.94183021
3.072	0.95	91.94	2.61112968
3.59	0.96	107.44	2.23437058
4.4533333333334	0.97	133.28	1.80121041
6.1800000000001	0.98	184.96	1.29795961
11.36	0.99	339.99	0.70610831

AE 622 5-3-90 PAUL BORCHERS

INERTIA COUPLING CALCULATIONS, FC 4

Flight Condition 4 M = 0.90 h = 15000 ft.

Short Period Frequency	3.800	rad/sec
Short Period Damping Ratio	0.350	--
Dutch Roll Frequency	4.180	rad/sec
Dutch Roll Damping Ratio	0.380	--

Omega S. P. /Roll Rate	Omega D/Roll Rate	Omega D/Omega S. P.	P1 (deg/sec)
0.01	1.1343434343434	0.01	21772.3453
0.05	1.14	0.06	4354.46906
0.10	1.1477777777778	0.11	2177.23453
0.15	1.1564705882353	0.17	1451.48969
0.20	1.16625	0.22	1088.61727
0.25	1.1773333333333	0.28	870.893812
0.30	1.19	0.33	725.744843
0.35	1.2046153846154	0.39	622.067009
0.40	1.2216666666667	0.44	544.308633
0.45	1.2418181818182	0.50	483.829896
0.50	1.266	0.55	435.446906
0.55	1.2955555555556	0.61	395.860824
0.60	1.3325	0.66	362.872422
0.65	1.38	0.72	334.959158
0.70	1.4433333333333	0.77	311.033504
0.75	1.532	0.83	290.297937
0.80	1.665	0.88	272.154316
0.85	1.8866666666667	0.94	256.145239
0.90	2.33	0.99	241.914948
0.91	2.4777777777778	1.00	239.256542
0.92	2.6625	1.01	236.655927
0.93	2.9	1.02	234.11124
0.94	3.2166666666667	1.03	231.620695
0.95	3.66	1.05	229.182582
0.96	4.325	1.06	226.795264
0.97	5.4333333333334	1.07	224.457168
0.98	7.6500000000001	1.08	222.166789
0.99	14.3	1.09	219.92268
1.1343434343434	0.01	1.25	191.937862
1.14	0.05	1.25	190.985485
1.1477777777778	0.10	1.26	189.691295
1.1564705882353	0.15	1.27	188.265448
1.16625	0.20	1.28	186.686776
1.1773333333333	0.25	1.30	184.92932
1.19	0.30	1.31	182.960885
1.2046153846154	0.35	1.33	180.741053
1.2216666666667	0.40	1.34	178.218379
1.2418181818182	0.45	1.37	175.326353
1.266	0.50	1.39	171.977451
1.2955555555556	0.55	1.43	168.054123

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FLIGHT CONDITION 4, CON'T

ω_{SP}/P_1	ω_D/P_1	ω_D/ω_{SP}	P_1 (DEG/SEC)
1.3325	0.60	1.47	163.394711
1.38	0.65	1.52	157.770618
1.4433333333333	0.70	1.59	150.847658
1.532	0.75	1.69	142.117136
1.665	0.80	1.83	130.764837
1.8866666666667	0.85	2.08	115.401124
2.33	0.90	2.56	93.4435421
2.4777777777778	0.91	2.73	87.8704519
2.6625	0.92	2.93	81.7740669
2.9	0.93	3.19	75.0770528
3.2166666666667	0.94	3.54	67.6860476
3.66	0.95	4.03	59.4872822
4.325	0.96	4.76	50.3406828
5.4333333333334	0.97	5.98	40.0718012
7.6500000000001	0.98	8.42	28.4605821
14.3	0.99	15.73	15.2254163

AE 622 5-3-90 PAUL BORCHERS

INERTIA COUPLING CALCULATIONS, FC 7 (OPEN LOOP)

Flight Condition 7 M = 1.60 h = 30000 ft.
Open Loop

Short Period Frequency	2.160	rad/sec
Short Period Damping Ratio	0.345	--
Dutch Roll Frequency	3.150	rad/sec
Dutch Roll Damping Ratio	0.120	--

Omega S. P. /Roll Rate	Omega D/Roll Rate	Omega D/Omega S. P.	P1 (deg/sec)
0.01	1.0418181818182	0.01	12375.8594
0.05	1.0435789473684	0.07	2475.17189
0.10	1.046	0.15	1237.58594
0.15	1.0487058823529	0.22	825.057296
0.20	1.05175	0.29	618.792972
0.25	1.0552	0.36	495.034377
0.30	1.0591428571429	0.44	412.528648
0.35	1.0636923076923	0.51	353.595984
0.40	1.069	0.58	309.396486
0.45	1.0752727272727	0.66	275.019099
0.50	1.0828	0.73	247.517189
0.55	1.092	0.80	225.015626
0.60	1.1035	0.88	206.264324
0.65	1.1182857142857	0.95	190.397837
0.70	1.138	1.02	176.797992
0.75	1.1656	1.09	165.011459
0.80	1.207	1.17	154.698243
0.85	1.276	1.24	145.598346
0.90	1.414	1.31	137.509549
0.91	1.46	1.33	135.998455
0.92	1.5175	1.34	134.520211
0.93	1.5914285714286	1.36	133.073757
0.94	1.69	1.37	131.658079
0.95	1.828	1.39	130.272205
0.96	2.035	1.40	128.915202
0.97	2.38	1.41	127.58618
0.98	3.07	1.43	126.28428
0.99	5.1400000000001	1.44	125.008681
1.0418181818182	0.01	1.52	118.790972
1.0435789473684	0.05	1.52	118.590543
1.046	0.10	1.53	118.316056
1.0487058823529	0.15	1.53	118.010775
1.05175	0.20	1.53	117.669213
1.0552	0.25	1.54	117.28449
1.0591428571429	0.30	1.54	116.847877
1.0636923076923	0.35	1.55	116.348114
1.069	0.40	1.56	115.770434
1.0752727272727	0.45	1.57	115.095074
1.0828	0.50	1.58	114.294971
1.092	0.55	1.59	113.332046

FLIGHT CONDITION 7, OPEN LOOP, CON'T

1.1035	0.60	1.61	112.150969
1.1182857142857	0.65	1.63	110.668135
1.138	0.70	1.66	108.750962
1.1656	0.75	1.70	106.17587
1.207	0.80	1.76	102.534047
1.276	0.85	1.86	96.989494
1.414	0.90	2.06	87.5237584
1.46	0.91	2.13	84.7661605
1.5175	0.92	2.21	81.5542632
1.5914285714286	0.93	2.32	77.7657236
1.69	0.94	2.46	73.2299375
1.828	0.95	2.67	67.701638
2.035	0.96	2.97	60.8150341
2.38	0.97	3.47	51.9994094
3.07	0.98	4.48	40.3122457
5.1400000000001	0.99	7.50	24.0775475

AE 622 5-3-90 PAUL BORCHERS

INERTIA COUPLING CALCULATIONS, FC 7

Flight Condition 7 M = 1.60 h = 30000 ft.

Short Period Frequency	1.850	rad/sec
Short Period Damping Ratio	0.300	--
Dutch Roll Frequency	1.650	rad/sec
Dutch Roll Damping Ratio	0.100	--

Omega S. P. /Roll Rate	Omega D/Roll Rate	Omega D/Omega S. P.	P1 (deg/sec)
0.01	1.030303030303	0.01	10599.6944
0.05	1.0315789473684	0.04	2119.93888
0.10	1.0333333333333	0.09	1059.96944
0.15	1.0352941176471	0.13	706.646295
0.20	1.0375	0.18	529.984721
0.25	1.04	0.22	423.987777
0.30	1.0428571428571	0.27	353.323147
0.35	1.0461538461538	0.31	302.848412
0.40	1.05	0.36	264.992361
0.45	1.0545454545455	0.40	235.548765
0.50	1.06	0.45	211.993888
0.55	1.0666666666667	0.49	192.721717
0.60	1.075	0.54	176.661574
0.65	1.0857142857143	0.58	163.072222
0.70	1.1	0.62	151.424206
0.75	1.12	0.67	141.329259
0.80	1.15	0.71	132.49618
0.85	1.2	0.76	124.702287
0.90	1.3	0.80	117.774382
0.91	1.3333333333333	0.81	116.480158
0.92	1.375	0.82	115.21407
0.93	1.4285714285714	0.83	113.975209
0.94	1.5	0.84	112.762707
0.95	1.6	0.85	111.575731
0.96	1.75	0.86	110.413464
0.97	2	0.87	109.2752
0.98	2.5	0.87	108.160147
0.99	4.0000000000001	0.88	107.06762
1.030303030303	0.01	0.92	102.879387
1.0315789473684	0.05	0.92	102.75214
1.0333333333333	0.10	0.92	102.577686
1.0352941176471	0.15	0.92	102.383412
1.0375	0.20	0.93	102.165729
1.04	0.25	0.93	101.920139
1.0428571428571	0.30	0.93	101.640905
1.0461538461538	0.35	0.93	101.320608
1.05	0.40	0.94	100.949471
1.0545454545455	0.45	0.94	100.514344
1.06	0.50	0.95	99.9971172
1.0666666666667	0.55	0.95	99.3721352

AE 622 3-27-90 PAUL BORCHERS

FLIGHT CONDITION 7, CON'T

ω_{SP}/P_1	ω_D/P_1	ω_D/ω_{SP}	P_1 (DEG/SEC)
1.075	0.60	0.96	98.6018086
1.0857142857143	0.65	0.97	97.6287644
1.1	0.70	0.98	96.3608584
1.12	0.75	1.00	94.6401288
1.15	0.80	1.03	92.1712559
1.2	0.85	1.07	88.3307869
1.3	0.90	1.16	81.5361109
1.3333333333333	0.91	1.19	79.4977082
1.375	0.92	1.23	77.0886867
1.4285714285714	0.93	1.27	74.197861
1.5	0.94	1.34	70.6646295
1.6	0.95	1.43	66.2480901
1.75	0.96	1.56	60.5696824
2	0.97	1.78	52.9984721
2.5	0.98	2.23	42.3987777
4.0000000000001	0.99	3.57	26.4992361

C_{n_β} DYNAMIC

FROM ROSKAM PT. 3, P. 271:

$$C_{n_{\beta_{dyn}}} = \{ C_{n_{\beta_B}} - (I_{zz_B}/I_{xx_B}) C_{\ell_{\beta_B}} \tan\alpha \} \cos\alpha$$

TO PREVENT INHERENT SPIN DEPARTURES, $C_{n_{\beta_{DYN}}} > 0$

$$C_{\ell_{\beta_B}} = (C_{\ell_\beta})_{wf_B} + (C_{\ell_\beta})_{v_B} + (C_{\ell_\beta})_{h_B}$$

FROM THE C_{ℓ_β} CALCULATIONS, ONLY $(C_{\ell_\beta})_{v_B}$ WILL CHANGE FROM THE ORIGINAL C_{ℓ_β} CALCULATIONS.

$$(C_{\ell_\beta})_{v_B} = (C_{Y_\beta})_v (.188 \cos(0°)) = .188\, C_{Y_{\beta_V}}$$

FC	$(C_{Y_\beta})_v$	$(C_{\ell_\beta})_{v_B}$
1	−.249	−.0468
2	−.320	−.0602
3	−.383	−.0720
4	−.383	−.0720
5	−.383	−.0720
6	−.399	−.0750
7	−.271	−.0509
8	−.312	−.0587

FC	$(C_{\ell_\beta})_{v_B}$	$C_{\ell_{\beta_{wf}}}$	$(C_{\ell_\beta})_B$
1	−.0468	−.246	−.2928
2	−.0602	−.0132	−.0734
3	−.0720	−.0145	−.0865
4	−.0720	−.1148	−.1868
5	−.0720	−.1681	−.2400
6	−.0750	−.0193	−.0943
7	−.0509	−.0120	−.0629
8	−.0587	−.0846	−.1433

$C_{n_{\beta}}$ DYNAMIC (CON'T)

ONLY THE VERTICAL TAIL CONTRIBUTION TO C_{n_β} IS DEPENDENT ON ANGLE OF ATTACK.

SUBSONIC: $C_{n_{\beta_V}} = -(C_{Y_{\beta_V}})(.4975)$

SUPERSONIC: $C_{n_{\beta_V}} = -(C_{Y_{\beta_V}})(.5272)$

FC	$C_{Y_{\beta_V}}$	$(C_{n_{\beta_V}})_B$	$(C_{n_{\beta_F}})_B$	$(C_{n_\beta})_B$
1	-.392	.1950	-.2083	-.01328
2	-.503	.2502	-.2406	9.643×10^{-3}
3	-.601	.2990	-.2345	.0645
4	-.601	.2990	-.2317	.0630
5	-.601	.2990	-.2217	.0773
6	-.626	.3300	-.2284	.1016
7	-.425	.2241	-.2345	-.0104
8	-.491	.2443	-.2094	.0349

SUBSTITUTING INTO THE ORIGINAL EQUATION:

FC	$(I_{zz})_B$ (SLUG·FT²)	$(I_{xx})_B$ (SLUG·FT²)	α_{STALL} (DEG)	$(C_{n_\beta})_{DYN}$
1	85219	11129	12.0*	.4532
2	86833	15388	12.0	.0955
3	81025	9919	12.0	.2099
4	81025	9919	12.0	.3789
5	84047	10448	12.0	.4770
6	78662	6733	12.0	.3285
7	83515	10183	12.0	.0908
8	83751	12796	12.0	.2291

* DEPENDENT ON LIFT ENGINE/THRUST VECTORING AT TAKEOFF FLIGHT CONDITION

AS THE SIGN OF $C_{n_{\beta_{DYN}}}$ IS POSITIVE FOR THE EIGHT FLIGHT CONDITIONS, THE AIRCRAFT DOES NOT EXHIBIT ANY INHERENT SPIN TENDENCIES AT STALL ANGLE OF ATTACK.

RIDE AND COMFORT CHARACTERISTICS

FROM ROSKAM, PT. VII, P. 107:

$$\bar{A} = (0.5 \rho U_1 C_{L_\alpha}) K_1 K_2 / (W/S)$$

$$K_1 = 0.66 + (0.39/\bar{c})\, \ell_{CREW}$$

FROM THE ACADS MODEL,

ℓ_{CREW} = DISTANCE FROM CREW STATION TO LEADING EDGE OF WING M.G.C.

= 20.04 FT.

$\bar{c}$ = 12.0 FT.

$$K_1 = 0.66 + (0.39/12.0)(20.04)$$

$$K_1 = 1.311$$

S = 347.9 SQ. FT.

FC	ALT. (FT)	WT. (LB.)	ρ	U (FT/SEC)	C_{L_α} (RAD^{-1})
1	0	30744	2.377^-3	223.28	3.458
2	100	28395	2.377^-3	948.94	4.438
3	10000	23980	1.7553^-3	969.96	4.577
4	15000	23980	1.4956^-3	951.57	4.577
5	30000	28223	.8893^-3	895.23	4.577
6	30000	24534	.8893^-3	1193.64	3.798
7	30000	27239	.8893^-3	1591.52	2.378
8	40000	29917	.5851^-3	774.48	4.319

RIDE AND COMFORT CHARACTERISTICS (CON'T)

FOR FIG. 4.4:

$$L_V = 2500 \text{ FT}$$

$$\bar{c}/L_V = 12/2500 = .0048$$

$$K = 4W/(32.2\pi\rho S\bar{c}) = 4W/(32.2\pi\rho(347.9)(12.0))$$

$$K = \frac{W}{\rho}(9.47150^{-6})$$

FC	W (LB)	ρ (SLUGS/FT³)	K	K_2
1	30744	2.377^{-3}	122.5	.365
2	28395	2.377^{-3}	113.1	.345
3	23980	1.7553^{-3}	129.4	.370
4	23980	1.4956^{-3}	151.9	.385
5	28223	$.8893^{-3}$	300.6	.525
6	24534	$.8893^{-3}$	261.3	.485
7	27239	$.8893^{-3}$	290.1	.520
8	29917	$.5851^{-3}$	484.3	.625

SUBSTITUTING INTO THE FIRST EQUATION,

$$\bar{A} = 0.5(347.9)\rho U_1 C_{L_\alpha} K_1 K_2/W$$
$$= 173.95\rho U_1 C_{L_\alpha} K_1 K_2/W$$

FC	$\bar{A}$
1	.00497
2	.02774
3	.02742
4	.02385
5	.01546
6	.01818
7	.01465
8	.009324

$\bar{A} < 0.005$ IS CONSIDERED ACCEPTABLE FOR TURBULENCE RESPONSE AT THE CREW STATION. AS INDICATED FROM THE VALUES FOR $\bar{A}$ ABOVE, THE AIRCRAFT WILL NEED A RIDE CONTROL SYSTEM.

ANALYSIS FOR VERTICAL TAIL REMOVAL

FROM MIL-F-8785C, FOR CLASS IV (FIGHTER) AIRCRAFT, THE CROSSWIND VELOCITY FOR LATERAL-DIRECTIONAL CONTROL ANALYSIS IS

30 KNOTS = 50.7 FT/SEC

THIS WILL BE USED TO PRODUCE A SIDESLIP ANGLE FOR EACH FLIGHT CONDITIONS THROUGH THE EQUATION:

$$\beta_g \approx \tan^{-1}\frac{V_g}{U_1}$$

FOR THE EIGHT FLIGHT CONDITIONS CHOSEN FOR THE STABILITY AND CONTROL ANALYSIS:

FC	U_1 (FT/SEC)	ρ (SLUGS/FT^3)	β_g (RAD)	$\bar{q}$ (LBS/FT^2)
1	223.28	2.377×10^{-3}	.2233	59.2
2	948.94	2.377×10^{-3}	.0534	1070.2
3	969.96	1.7553×10^{-3}	.0522	825.7
4	951.57	1.4956×10^{-3}	.0532	677.1
5	895.23	$.8893\times10^{-3}$	.0566	356.4
6	1193.64	$.8893\times10^{-3}$	.0424	633.5
7	1591.52	$.8893\times10^{-3}$	.0318	1126.3
8	774.48	$.5851\times10^{-3}$	.0654	175.5

$C_{n_{\beta_{WB}}}$ (EXCLUDING THE VERTICAL TAIL) HAS BEEN DETERMINED. FOR STATIC DIRECTIONAL STABILITY, A C_{n_β} = +.001 DEG^{-1} = +.0573 RAD^{-1} IS DESIRED. THEREFORE, THE CHANGE IN C_{n_β} ($\Delta C_{n_{\beta_{VT}}}$) THAT MUST BE MADE UP THROUGH THRUST DEFLECTION CAN BE CALCULATED.

FC	$C_{n_{\beta_{WB}}}$ (RAD^{-1})	$\Delta C_{n_{\beta_{VT}}}$ (RAD^{-1})
1	-.2083	.2656
2	-.2406	.2979
3	-.2345	.2918
4	-.2317	.2890
5	-.2217	.2790
6	-.2284	.2857
7	-.2345	.2918
8	-.2094	.2667

THRUST WILL BE DIVERTED FOR YAW AT F.S. 625

$$\therefore\ x_T = \frac{625 - x_{CG}}{12}\ (FT)$$

$$\Delta C_{n_{\beta VT}} \beta_g = \frac{F_{Y_T} x_T}{\bar{q} S b} \quad OR \quad F_{Y_T} = \frac{\Delta C_{n_{\beta VT}} \beta_g \bar{q} S b}{x_T}$$

$$Sb = (347.9)(33.67) = 11713.8\ FT^4$$

FC	$x_T = \frac{625 - x_{CG}}{12}$ (FT)	ΔC_{n_β} (RAD^{-1})	$\bar{q}$ (LB/FT^2)	F_{Y_T} (LB)
1	15.92	.2656	59.2	2583
2	15.25	.2979	1070.2	13077
3	15.17	.2918	825.7	9712
4	15.17	.2890	677.1	8038
5	15.58	.2790	356.4	4231
6	15.17	.2857	633.5	5926
7	15.42	.2918	1126.3	7939
8	15.33	.2667	175.5	2339

ANALYSIS FOR RUDDER REMOVAL

AS A MEASURE OF STATIC CONTROL POWER,

$$C_{n_\beta} \beta_g = -C_{n_{\delta r}} \delta_r$$

WITH THE ORIGINAL VERTICAL TAIL ON THE AIRCRAFT,

C_{n_β} (RAD^{-1})	β_g (RAD)	$C_{n_{\delta r}}$ (RAD^{-1})	δ_r (RAD)	δ_r (DEG)
-.0034	.2233	-.071	-.00834	-.478
.0112	.0534	-.111	.00539	.309
.0665	.0522	-.133	.0261	1.49
.0802	.0532	-.137	.0311	1.78
.0941	.0566	-.139	.0383	2.19
.1048	.0424	-.139	.0320	1.83
-.0083	.0318	-.094	-.00281	-.161
.0436	.0654	-.111	.0257	1.47

THE SIDE FORCE CAN BE CALCULATED FROM THIS INFORMATION:

$$C_{Y_{\delta r}} \delta_r = \frac{F_{Y_T} x_T}{\bar{q} S b}$$

$$F_{Y_T} = \frac{C_{Y_{\delta r}} \delta_r \bar{q} S b}{x_T}$$

$$Sb = 11713.8\ FT^4$$

$$x_T = \frac{625 - x_{cg}}{12}\ (FT),$$ REPRESENTING THE LOCATION OF YAW THRUST

FC	x_T (FT)	δ_r (RAD)	$C_{y\delta_r}$ (RAD^{-1})	$\bar{q}$ (LB/FT^2)	F_{Y_T} (LB)
1	15.92	-.00834	.1692	59.2	-61.5
2	15.25	.00539	.2173	1070.2	962.8
3	15.17	.0261	.2595	825.7	4318.3
4	15.17	.0311	.2595	677.1	4219.5
5	15.58	.0383	.2595	356.4	2663.2
6	15.17	.0320	.2704	633.5	4232.7
7	15.42	-.00281	.1836	1126.3	-441.4
8	15.33	.0257	.2118	175.5	729.9

FLIGHT CONDITION #7

$M = 1.6$ $h = 30{,}000$ FT

$V = 1592$ FPS $\rho = .0008893$ SLUG/FT^3

$S = 364$ FT^2

$T_{req} = C_D \bar{q} S$

$T_{REQ} = 16075$ lbs FROM MISSION ANALYSIS SPREADSHEET

- 5% BLEED FOR OTHER SYSTEMS

$(5.26)(1.2)$
$A_N = 6.312$ ft^2 $(144) = 909$ IN^2

$A_{YV} = 200$ IN^2 = 20 IN · 10 IN

$A_{YV} = .22 A_N$

$T_N = 26808$ lbs

TOO MUCH THRUST.

$A_N = (4.393)(1.2)(144) = 759$ IN^2

$A_{YV} = 200$ IN^2

$A_{YV} = .358 A_N$

$T_{AV} = 20694$ lbs ASSUME 5% LOSSES = 19660 lbs

$T_N = \frac{19660}{1.358} = 14480$ lbs; $T_{YV} = 5183$

$T_X = 18145$ lbs $T_Y = 3665$ lbs

T_{REQ_Y} FOR RUDDER REMOVAL : 442 lbs

$\dot{W}_F = 38627$ lbf / hr

ORIGINAL PAGE IS
OF POOR QUALITY

10% ENGINE DECK BLEED

5% RCS BLEED FOR YAW CONTROL

$T_{REQ} = 16075$ lb

$T_{AV} = (14432)(1.2) = 17318$ lbs SFC = .33673 lb_{fuel} hr / lb_{thr}

FOR RCS BLEED OF 5%

$T = \dot{m} V_e - (p_e - p_\infty)A$

$\dot{m} = a^* \rho^* A$

$V_e = a^*$

$\therefore \; T = a^{*2} \rho^* A - (p_e - p_\infty)A$

ATMOSPHERE

$T = 412\,°R$

$P = 628$ psf

$\rho = .0008873$ slug/ft^3

TEMP & PRES. BEHIND FAN WILL BE:

$P = (fpr)\, 628$ psf

$$T = 412\left(\frac{(fpr)^{.4/1.4} - 1}{\eta_f} + 1\right)$$

FOR THE ENGINE fpr = 5.5 $\eta_f = .88$

$P = (5.5)\, 628 = 3454$ psf

$T = 706\,°R$

ORIGINAL PAGE IS
OF POOR QUALITY

FOR COMPRESSOR

$$\eta_c = .90 \qquad fpr = \frac{30}{5.5} = 5.46$$

$$T = 706\left(\frac{5.46^{.386/1.386} - 1}{.9} + 1\right) = 1180\ ^\circ R$$

$$P = 3454\ psf\ (5.46) = 18860\ psf$$

$$\rho = \frac{P}{RT} = \frac{18860}{(1717)1180} = .00931\ sl/ft^3$$

A CHOKED NOZZLE IS ASSUMED:

$$a_o = \sqrt{\gamma RT} = [(1.35(1717)\,1180]^{1/2} = 1654\ fps$$

THROAT CONDITIONS

$$p^* = p\left(\frac{2}{\gamma+1}\right)^{\gamma/\gamma-1} = 18860\left(\frac{2}{2.35}\right)^{1.35/.35} = 10125\ psf$$

$$t^* = \left(\frac{2}{\gamma+1}\right)t = 1180\left(\frac{2}{2.35}\right) = 1004\ ^\circ R$$

$$\rho^* = \frac{p^*}{Rt^*} = \frac{10125}{(1717)(1004)} = .00587\ slugs/ft^3$$

$$a^* = a_o\left(\frac{2}{\gamma+1}\right)^{1/2} = 1567\ fps$$

$$T = \dot{m}V_e - (p_e - p_o)A \qquad \dot{m} = a^*\rho^*A \qquad V_e = a^*$$

$$T = a^{*2}\rho^*A + (p_e - p_o)A$$

$$\dot{m}_{des} = 190\ (1.2) = 228\ lbm/s$$

$$\dot{m}_A = \frac{\dot{m}_{des}}{(1+\beta)} = \frac{228}{1.8} = 126.7\ lbm/s$$

ORIGINAL PAGE IS OF POOR QUALITY

$\dot{m}_{RCS} = .05\,\dot{m}_A = (.05)\,126.7\ lb_m/s = 6.34\ lb_m/s$

$\dot{m}_{RCS} = .1967\ slug/s$

$\dot{m}_{RCS} = a^* \rho^* A$

$.1967\ slug/s = 1567\ ft/s\ .00587\ slug/ft^3\ A$

$A = \frac{(.1967)}{(1567)(.00587)}\ ft^2 = .0214\ FT^2$

$T = a^{*2} \rho^* A + (p_e - p_o) A$

$T = 1567^2 .00587^2 .0214 + (10125 - 628) .0214$

$T = 203\ lbs = F_y$

$T_{AV} = 17318\ lbs$ w/ 5% losses $= 16452\ lbs$

$A_N = 4.037(1.2) = 4.84\ FT^2 = 698\ IN^2$

$A_{YV} = 200\ IN^2$

$A_{YV} = .287\,A_N$

$T_X = .977\ T_{AV} = \boxed{16075\ lbs}$

$T_Y = (1 - .977)\,T_{AV} = 368\ lbs + 203 = \boxed{571\ lbs}$

$\dot{W}_F = 33673\ lb_f/hr$

ORIGINAL PAGE IS
OF POOR QUALITY

15% ENGINE DECK BLEED

10% RCS BLEED FOR YAW CONTROL

$T_{REQ} = 16075$ lb

$T_{AV} = 13500(1.2) = 16200$ lb   SFC = 30290(1.2) = 36348 lbm/hr

$T = 412\,°R$

$P = 628$ psf

$\rho = .0008893$ slug/ft³

T & P BEHIND FAN:

$P = 628$ psf $(5.5) = 3454$ psf

$T = 706\,°R$

AFTER COMP:

$T = 1180\,°R$

$P = 18360$ psf

$\rho = .00931$ slug/ft³

AT THROAT

$P^* = 10125$ psf

$t^* = 1004\,°R$

$\rho^* = .00587$ slug/ft³

$a^* = 1557$ fps

$\dot{m}_{des} = 174.5(1.2) = 209.4$ lbm/s

$$\dot{m}_A = \frac{209.4}{1.8} = 116.3 \text{ lbm/s}$$

ORIGINAL PAGE IS
OF POOR QUALITY

$\dot{m}_{RCS} = .10\,(116.7)\ lb_m/s = 11.67\ lb_m/s$

$\dot{m}_{RCS} = .3613\ slug/s$

$\dot{m}_{RCS} = a^* \rho^* A$

$A = \dfrac{.3617}{(1567)(.00587)} = .03928\ FT^2$

$T = a^{*2} \rho^* A + (P_e - P_o) A$

$T = 1567^2(.00587)(.03928) + (5125 - 628)\,.03928$

$T = \boxed{939\ lbs = F_y}$

$\boxed{F_x = 16200\ lbs}$

$\dot{W}_F = 36348\ lb_f / hr$

20° VECTORING NOZZLE

$T_{REQ} = 16075\ lbs$

$T_y = 442\ lbs$

$\tan 20° = .364 \therefore$ $T_y \le .364\, T_{REQ}$ FOR NOZZLE TO WORK

$T_{AV} = \sqrt{T_{REQ}^2 + T_y^2} = \sqrt{16075^2 + 442^2} = 16081\ lbs$

$T_{AV} = 14107\,(1.2) = 16928\ lbs$

$SFC = 24915\ lb_f / hr$

$16928 \sin \phi = 442\ lbs$

$\phi = \sin^{-1}\left(\dfrac{442}{16928}\right) = 1.5°$ DEFLECTION WILL WORK

FLIGHT COND. 3

$M = .9 \quad h = 10{,}000 \text{ FT} \quad S = 364 \text{ ft}^2$

$V = 945 \text{ fps} \quad \rho = .0017553 \text{ slug/ft}^3 \quad C_L = .083$

$C_D = .02500 + .1003\, C_L^2$

$C_D = .02569$

$T_{REQ} = D = C_D\, q\, S = .02569(.0017553)(945)^2\, 364$

$T_{REQ} = 14660 \text{ lbs}$

-5% bleed for other systems

$A_N = 3.436\,(1.2) = 4.12 \text{ FT}^2 = 594 \text{ IN}^2$

$T_{AV} = 15469\,(1.2) = 18563 \text{ lbs}$ w/ 5% losses = 17635 lbs

$T_{REQ} = T_{AV}(f)$

$f = \frac{14660}{17635} = .831$

$T_X = T_{AV}(.831-(1-.831)) + (1-.831)T_{AV} = \boxed{14{,}660 \text{ lbs}}$

$T_Y = (1-.831)\,T_{AV} = \boxed{2980 \text{ lbs}}$ NOT ENOUGH

$A_N = 4.384\,(1.2) = 5.26 \text{ FT}^2 = 758 \text{ IN}^2$

$T_{AV} = 21522\,(1.2) = 25826 \text{ lbs}$ w/ 5% losses = 24,535 lbs

$$\frac{A_W}{A_N} = \frac{200}{758-200} = .358$$

$$\frac{T_{yW}}{T_N} = .358$$

$$T_{yW} = .358\ T_N$$

$$T_N = \frac{24{,}535}{1.358} = 18067 \text{ lbs}$$

$$T_{yW} = 6468$$

$$T_x = 18067 + \cos 45^\circ\ 6468 = \boxed{22641 \text{ lbs}} \Rightarrow \text{NOT ACCEPTABLE}$$

$$T_y = \sin 45^\circ\ 6468 = \boxed{4574 \text{ lbs}}$$

$T_{AV} \ggg T_{req}$

NEED 4318 lbs

USING 15% bleed

$T_{req} = 14660$ lbs

$T_{AV} = 12{,}777$ lbs $= 15{,}332$ lbs

SFC $= 23.375\ \text{lb}_f/\text{hr} = 28{,}050\ \text{lb}_f/\text{hr}$

$T = 483\ °R$

$P = 1456$ psf

$\rho = .0017553\ \text{slug}/\text{ft}^3$

TEMP | PRS. BEHIND FAN

$P = 5.5\,(1456\ \text{psf}) = 8008$ psf

$$T = 483\left(\frac{5.5^{.4/1.4} - 1}{.88} + 1\right) = 827.4\ °R$$

ORIGINAL PAGE IS OF POOR QUALITY

FOR COMPRESSOR

$\eta_c = .90 \qquad fpr = \frac{30}{5.5} = 5.46$

$$T = 827.1\left(\frac{5.46^{.386/1.386} - 1}{.9} + 1\right) = 1383°R$$

$P = 8008 \text{ psf } (5.46) = 43724 \text{ psf}$

$$\rho = \frac{P}{RT} = \frac{43724}{1717(1383)} = .01841 \text{ slug/ft}^3$$

ASSUME CHOKED NOZZLE

$$a_0 = \sqrt{\gamma R T} = [(1.35)(1717)1383]^{1/2} = 1790 \text{ fps}$$

THROAT CONDITIONS

$$p^* = p\left(\frac{2}{\gamma+1}\right)^{\gamma/\gamma-1} = 43724\left(\frac{2}{2.35}\right)^{1.35/.35} = 23473 \text{ psf}$$

$$t^* = \left(\frac{2}{\gamma+1}\right) t = \left(\frac{2}{2.35}\right)1383 = 1177°R$$

$$\rho^* = \frac{p^*}{R t^*} = \frac{23473}{(1717)(1177)} = .01162 \text{ slug/ft}^3$$

$$a^* = a_0\left(\frac{2}{\gamma+1}\right)^{1/2} = 1696 \text{ fps}$$

$\dot{m}_{du} = 200 \; (1.2) = 240 \text{ lbm/s}$

$$\dot{m}_A = \frac{\dot{m}_{du}}{(1+\beta)} = \frac{240}{1.8} = 133.3 \text{ lbm/s}$$

$\dot{m}_{RCS} = .05\,\dot{m}_A = (.05)\,133.3 = 6.67 \text{ lbm/s}$

$\dot{m}_{RCS} = .207 \text{ slug/s}$

ORIGINAL PAGE IS
OF POOR QUALITY

$\dot{m}_{RCS} = a^* \rho^* A$

$A = \frac{\dot{m}_{RCS}}{a^* \rho^*} = \frac{.201}{(1696)(.01162)} = .0105 \ ft^2$

$T_y = a^{*2} \rho^* A + (p_c - p_o) A$

$T_y = 1696^2 (.01162) .0105 + (23473 - 1456) .0105$

$T_y = 582$ lbs

$T_{y_{REQ}} = 4318$ lbs

$T_y \ll T_{y_{REQ}}$ WILL NOT WORK

20° VECTORING NOZZLE

$T_{x_{REQ}} = 14660$ lbs

$T_{y_{REQ}} = 4318$ lbs

$\frac{T_{y_{REQ}}}{T_{REQ}} = \frac{4318}{14660} = .245 < .364$ WILL BE LESS THAN 20°

$T_{RES} = \sqrt{14660^2 + 4318^2} = 15283$ lbs

$T_{AV} = 31833$ lbs w/ AFTERBURNER

$T_{AV} = 18563$ DRY

$\phi = \sin^{-1}\left(\frac{4318}{14660}\right) = 17°$ CAN BE DONE

FLIGHT CONDITION 4

$20°$ VECTORING NOZZLE

$M = .9 \qquad h = 15{,}000$ FT

$V = 970$ fps

$$T_{X_{REQ}} = C_D \bar{q} S = (C_{D_0} + C_L^2 K)\bar{q} S$$

$C_{D_0} = .02336 \qquad K = .1003 \qquad C_L = .661$

$\bar{q} = 970^2(.0014956)\,1/2 = 704$ psf $\qquad S = 364$ ft^2

$$T_{X_{REQ}} = (.02336 + .1003(.661)^2)\,704\,(364) = 17206 \text{ lb}$$

$T_{Y_{REQ}} = 4219$ lbs

$$T_{REQ} = \sqrt{17206^2 + 4219^2} = 17716 \text{ lbs}$$

$T_{AV} = 28550$ lbs w/ AFTERBURNER

$T_{AV} = 16783$ lbs DRY

WILL NEED AFTERBURNER

$$\phi = \tan^{-1}\left(\frac{4219}{17206}\right) = 14°$$ CAN BE DONE

ORIGINAL PAGE IS
OF POOR QUALITY

FLIGHT CONDITION 6

$M = 1.2 \qquad h = 30{,}000$ ft

$V = 1194$ fps $\qquad \bar{q} = 635$ psf

$S = 364$ ft^2 $\qquad C_L = .111$

$$T_{X_{REQ}} = (C_{D_o} + K C_L^2)\, \bar{q}\, S$$

$$T_{X_{REQ}} = (.0401 + .1003(.111)^2)(635)(364) = 9554 \text{ lbs}$$

$$T_{Y_{REQ}} = 4232 \text{ lbs}$$

$$T_{REQ} = \sqrt{9554^2 + 4232^2} = 10450 \text{ lbs}$$

$T_{AV} = 22460$ lbs w/ AFTERBURNER

$T_{AV} = 11977$ lbs DRY

$$\beta = \tan^{-1}\left(\frac{4232}{9554}\right) = 24°$$

ORIGINAL PAGE IS
OF POOR QUALITY

FLIGHT CONDITION 1

$M = .2 \qquad h = 0$ ft

$V = 223$ fps $\qquad \bar{q} = 59.2 \qquad S = 364\ ft^2$

$T_{x_{REQ}} = (C_{D_0} + K C_L^2)\bar{q} S$

$T_{x_{REQ}} = (.02295 + .1143(1.495)^2)(59.2)(364)$

$T_{x_{REQ}} = 6000$ lbs

$T_{y_{REQ}} = 62$ lbs

$T_{REQ} = \sqrt{6000^2 + 62^2} = 6000$ lbs

$T_{AV} = 34250$ lbs w/ AFTERBURNER

$T_{AV} = 23063$ lbs DRY

$\phi = \tan^{-1}\left(\frac{62}{6000}\right) = .6°$

FLIGHT COND 2

$M = .85 \qquad h = 0$ ft

$V = 950$ fps $\qquad \bar{q} = 1070$ psf $\qquad S = 364\ ft^2$

$T_{x_{REQ}} = (C_{D_0} + K C_L^2)\bar{q} S = (.0213 + .1014(.076)^2)(1070)(364) = 8524$ lbs

$T_{y_{REQ}} = 963$ lbs

$T_{REQ} = \sqrt{8524^2 + 963^2} = 8580$ lbs

$T_{AV} = 37,284$ lbs w/ AB $\qquad T_{AV} = 22,108$ lbs DRY

$\phi = \tan^{-1}\left(\frac{963}{8524}\right) = 6.4°$

FLIGHT CONDITION 5

$M = .9$ $h = 20{,}000$ ft $S = 364$ ft^2

$V = 895$ fps $\bar{q} = 356$ psf $C_L = .968$

$$T_{x_{REQ}} = (C_{D_0} + KC_L^2)\bar{q}S = (.0235 + .1103(.968)^2)\,356\,(364)$$

$$T_{x_{REQ}} = 16438 \text{ lbs}$$

$$T_{Y_{REQ}} = 2663 \text{ lbs}$$

$$T_{REQ} = \sqrt{16438^2 + 2663^2} = 16{,}653 \text{ lbs}$$

$$T_{AV} = 18{,}566 \text{ lbs w/ AB}$$

$$T_{AV} = 7489 \text{ lbs DRY}$$

$$\phi = \tan^{-1}\left(\frac{2663}{16438}\right) = 9^\circ$$

FLIGHT CONDITION 8

$M = .80$ $h = 40{,}000$ ft $S = 364$ ft^2

$V = 775$ lbs $\bar{q} = 176$ psf $C_L = .48$

$$T_{x_{REQ}} = (C_{D_0} + KC_L^2)\bar{q}S = (.0235 + .1103(.48)^2)(176)(364)$$

$$T_{x_{REQ}} = 3133 \text{ lbs}$$

$$T_{Y_{REQ}} = 730 \text{ lbs}$$

$$T_{REQ} = \sqrt{3133^2 + 730^2} = 3217 \text{ lbs}$$

$T_{AV} = 10323$ w/ AB $T_{AV} = 6455$ lbs DRY

$$\phi = \tan^{-1}\left(\frac{730}{3133}\right) = 13^\circ$$

FC	Tx_{REQ} (lbs)	Ty_{REQ} (lbs)	ϕ (YAW VANE DEFLECTION) (DEG)
1	6000	62	1
2	8524	963	4
3	14660	4318	12
4	17206	4219	6
5	18566	2663	10
6	9554	4232	17
7	16075	442	1
8	3133	730	9

FC	$X_T = \frac{695 - X_{cg}}{12}$ (FT)	β_g (RAD)	$\Delta C_{n_{\beta_{VT}}}$ (RAD^{-1})	$\bar{q}$ (psf)
1	21.75	.2233	.2656	59.2
2	21.08	.0534	.2979	1070.2
3	21.00	.0522	.2918	825.7
4	21.00	.0532	.2890	677.1
5	21.41	.0566	.2790	356.4
6	21.00	.0424	.2857	633.5
7	21.25	.0318	.2918	1126.3
8	21.16	.0654	.2667	175.5

NOTE: 695 IS F.S. LOCATION OF THRUST DEFLECTION

$\Delta C_{n_{\beta_{VT}}}$ & β_g IS FROM PAUL BORCHERS

$$F_{Y_T}(\text{lbs}) = \frac{\Delta C_{n\beta_{VT}} \beta_g \bar{q} S b}{X_T}$$ (FROM PAUL BORCHERS 2-27-90)

$$Sb = (347.9)(33.67) = 11714 \text{ FT}^3$$

$\therefore$

FC	F_{Y_T} (lbs) REQUIRED TO REMOVE VERTICAL TAIL
1	1916
2	9459
3	7018
4	5806
5	3079
6	4281
7	5760
8	1700

THE FOLLOWING IS $T_{x_{REQ}}$ & T_{AV} FOR THE FLIGHT FLIGHT CONDITIONS.

FC	$T_{x_{REQ}}$ (lbs)	T_{AV} (lbs)
1	6000	23063 DRY
2	8524	22108 DRY
3	14660	18563 DRY
4	17206	28550 W/ AB
5	16438	18566 W/ AB
6	9554	11977 DRY
7	16075	16928 DRY
8	3133	6455 DRY

ORIGINAL PAGE IS OF POOR QUALITY

ASSUME MAX. THRUST DEFLECTION IS 20°
FOR ALL FLIGHT CONDITIONS.

FC1

$T_{x_{REQ}} = 6000$ lbs

$T_{Y_{MAX}} = 2052$ lbs

$T_{REQ} = \sqrt{6000^2 + 2052^2} = 6341$ lbs

$T_{AV} = 23063$ lbs DRY

✓ $T_{AV} > T_{REQ}$

$T_{Y_{REQ}} = 1916$ lbs

$T_{Y_{MAX}} > T_{Y_{REQ}}$ ∴ V-TAIL CAN BE REMOVED IN FLIGHT CONDITION 1

FC2

$T_{X_{REQ}} = 8524$ lbs

$T_{Y_{MAX}} = 2915$ lbs

$T_{REQ} = 9008$ lbs

$T_{AV} = 22108$ lbs DRY

✓ $T_{AV} > T_{REQ}$

$T_{Y_{REQ}} = 9459$

∴ $F_{Y_{VT}} = 9459 - 2915 = 6544$ lbs FOR FC 2

ORIGINAL PAGE IS
OF POOR QUALITY

FC 3

$T_{X_{REQ}} = 14660$ lbs

$T_{Y_{MAX}} = 5014$ lbs

$T_{REQ} = 15494$ lbs

$T_{AV} = 18563$ lbs DRY

✓ $T_{AV} > T_{REQ}$

$T_{Y_{REQ}} = 7018$ lbs

$\therefore$

$F_{Y_{VT}} = 7018 - 5014 = 2004$ lbs FOR FC3

FC4

$T_{X_{REQ}} = 17206$ lbs

$T_{Y_{max}} = 5885$ lbs

$T_{REQ} = 18185$ lbs

$T_{AV} = 28550$ lbs w/ ~~A/B~~

✓ $T_{AV} > T_{REQ}$

$T_{Y_{REQ}} = 5806$ lbs

$T_{Y_{MAX}} > T_{Y_{REQ}} \therefore$ V-TAIL CAN BE REMOVED FOR F.C.4

FC 5

$T_{x_{REQ}} = 16438$ lbs

$T_{y_{MAX}} = 5622$ lbs

$T_{REQ} = 17373$ lbs

$T_{AV} = 18566$ lbs W/ AB

✓ $T_{AV} > T_{REQ}$

$T_{y_{REQ}} = 3079$ lbs

$T_{y_{MAX}} > T_{y_{REQ}}$ ∴ V-TAIL CAN BE REMOVED FOR F.C. 5

F.C. 6

$T_{x_{REQ}} = 9554$ lbs

$T_{y_{MAX}} = 3268$ lbs

$T_{REQ} = 10097$ lbs

$T_{AV} = 11977$ lbs

✓ $T_{AV} > T_{REQ}$

$T_{y_{REQ}} = 4281$ lbs

∴

$F_{y_{VT}} = 4281 - 3268 = 1013$ lbs FOR FC6

FC 7

$T_{x_{REQ}} = 16075$ lbs

$T_{y_{max}} = 5498$ lbs

$T_{REQ} = 16980$ lbs

$T_{AV} = 16928$ lbs DRY

✓ $T_{AV} \approx T_{REQ}$

$T_{y_{REQ}} = 5760$ lbs

$\therefore$

$F_{y_{VT}} = 5760 - 5498 = 262$ lbs FOR FC 7

FC 8

$T_{x_{REQ}} = 3133$ lbs

$T_{y_{max}} = 1072$ lbs

$T_{REQ} = 3311$ lbs

$T_{AV} = 6455$ lbs DRY

✓ $T_{AV} > T_{REQ}$

$T_{y_{REQ}} = 1700$ lbs

$\therefore$

$F_{y_{VT}} = 1700 - 1072 = 628$ lbs FOR FC 8

FC	$F_{Y_{VT}}$ (lbs)
1	0
2	6544
3	2004
4	0
5	0
6	1013
7	262
8	628

$\Delta C_{n_{\beta_{TV}}} \equiv$ CHANGE IN C_{n_β} DUE TO THRUST VECTORING

FC 2

$$\Delta C_{n_{\beta_{TV}}} = \frac{F_{y_{max}}\, x_T}{\bar{q}\, S b\, \beta_g} = \frac{(2915)(21.08)}{1070.2\,(11714)(.0534)}$$

$$\Delta C_{n_{\beta_{TV}}} = .0918$$

FC 3

$$\Delta C_{n_{\beta_{TV}}} = \frac{(5014)(21.00)}{(826)(11714)(.0522)}$$

$$\Delta C_{n_{\beta_{TV}}} = .2085$$

FC 6

$$\Delta C_{n_{\beta_{TV}}} = \frac{(3268)(21.00)}{(633.5)\,11714\,(.0424)}$$

$$\Delta C_{n_{\beta_{TV}}} = .2181$$

FC 7

$$\Delta C_{n_{\beta_{TV}}} = \frac{(5498)(21.25)}{(1126)(11714)(.0318)}$$

$$\Delta C_{n_{\beta_{TV}}} = .2785$$

FC 8

$$\Delta C_{n_{\beta_{TV}}} = \frac{(1072)(21.16)}{(175.5)(11714)(.0654)}$$

$$\Delta C_{n_{\beta_{TV}}} = .1687$$

$$C_{n_{\beta_{VT}}} = \Delta C_{n_{\beta_{VT}}} - \Delta C_{n_{\beta_{TV}}}$$

$\therefore$

FC	$C_{n_{\beta_{VT}}}$ (RAD^{-1})
1	0
2	.2061
3	.0833
4	0
5	0
6	.0676
7	.0133
8	.0980

$$C_{n_{\beta_{VT}}} = C_{L_{\alpha_V}} (S_V/S)(X_V/b)$$

VERTICAL TAIL GEOMETRY

$\Lambda_{c/2} = 31.4°$

$A = 2.44$

$S = 363 \ FT^2$

$b = 33.67 \ FT$

$X_V = 16.25$ FT (SUBSONIC)

$X_V = 17.0$ FT (SUPERSONIC) ASSUMED

$C_{L_{\alpha_V}} = 2.80 \ RAD^{-1}$ @ $M = .2$

$C_{L_{\alpha_V}} = 3.51 \ RAD^{-1}$ @ $M = .8$

$C_{L_{\alpha_V}} = 3.60 \ RAD^{-1}$ @ $M = .85$

$C_{L_{\alpha_V}} = 4.30 \ RAD^{-1}$ @ $M = .9$

$C_{L_{\alpha_V}} = 4.48 \ RAD^{-1}$ @ $M = 1.2$

$C_{L_{\alpha_V}} = 3.04 \ RAD^{-1}$ @ $M = 1.6$

FC	$C_{n_{\beta_{VT}}}$ $(RAD \ FT^2)^{-1}$	$C_{n_{\beta_{VT\,REQ}}}$ (RAD^{-1})	$S_{V_{REQ}}$ (FT^2)
2	$.004786 \ S_V$	.2061	43.1
3	$.005717 \ S_V$	.0833	14.6
6	$.006231 \ S_V$	.0676	10.8
7	$.004228 \ S_V$	.0133	3.1
8	$.004882 \ S_V$	.0980	20.1

$\therefore$ $S_V = 43.1 \ FT^2$

$A_V = 1.61$ (OLD V-TAIL)
$\Lambda_{V_{c/2}} = 31.4°$ (OLD V-TAIL)

HOLD THE SAME SO THAT $C_{L_{\alpha_V}}$ REMAINS THE SAME

$$A = \frac{b^2}{S}$$

$$b = \sqrt{AS} = \sqrt{43.1(1.61)} = 8.34 \ FT$$

$$\lambda = .355$$

$$S = \frac{(C_t + C_r)b}{2}$$

$$C_t = .355 C_r$$

$$S = \frac{(.355C_r + C_r)b}{2}$$

$$(2)\ 43.1 \ FT^2 = 1.355 C_r (8.34) FT$$

$$C_r = 7.63 \ FT$$

$$C_t = 2.71 \ FT$$

$$\bar{c} = C_r \frac{2}{3}\left(\frac{\lambda^2 + \lambda + 1}{\lambda + 1}\right)$$

$$\bar{C} = 7.63\left(\frac{2}{3}\right)\left(\frac{.355^2 + .355 + 1}{.355 + 1}\right)$$

$$\bar{C} = 5.56 \ FT$$

$S_v = 43.1 \ FT^2$

$A_v = 1.61$

$\Lambda_{LE} = 40.5°$

$\lambda = .355$

$b = 8.34 \ FT$

$C_r = 7.63 \ FT$

$C_t = 2.71 \ FT$

$\bar{c} = 5.56 \ FT$

APPENDIX 4

The purpose of this appendix is to show the calculations for the landing gear sizing and the horizontal stabilator actuator sizing as discussed in Chapter 11.

Appendix 4: Table of Contents

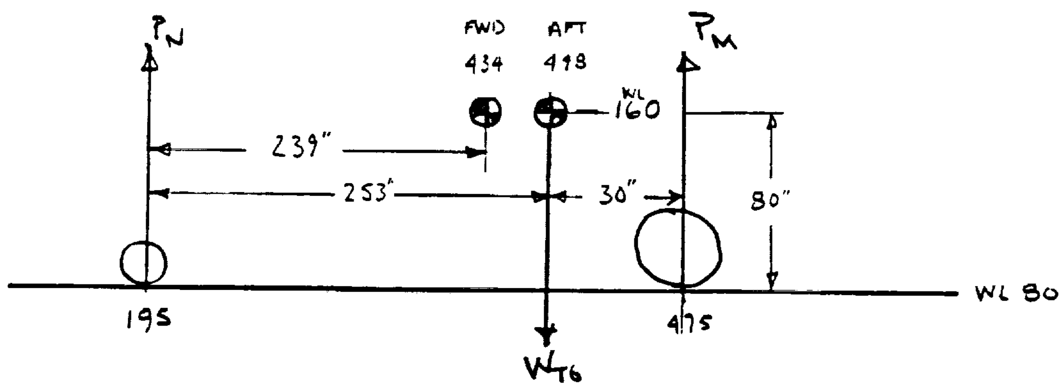
LANDING GEAR SIZING | 27 MAR 90 | AE627 | TAVIS JACOBS

FWD CG CA TAKEOFF FS 434, WL 160 } CONFIG BOOK 27 MAR 90

AFT CG CA HOVER FS 448, WL 158 } CONFIG BOOK 27 MAR 90

MAIN GEAR CONTACT POINT FS 475
NOSE GEAR CONTACT POINT FS 195 } FROM DRAWING 27 MAR 90

$W_{TO_{MAX}}$ 34,786 lb BAI #1 CONFIG BOOK 27 MAR 90



MAXIMUM STATIC MAIN GEAR LOAD = $\frac{(34{,}786)(283-30)}{2(283)}$

P_M = 15,550 lb

MAXIMUM STATIC NOSE GEAR LOAD = $\frac{(34{,}786)(283-239)}{283}$

$P_{NS_{MAX}}$ = 5,400 lb

MINIMUM STATIC NOSE GEAR LOAD = $\frac{(34{,}786)(283-253)}{283}$

$P_{N_{MIN}}$ = 3,690 lb

DYNAMIC NOSE GEAR LOAD = 5,400 lb + $\frac{10(80)(34{,}786)}{(32.2)(283)}$

P_{ND} = 8,450 lb

STROKE LENGTH

$$S_s = \left[\frac{\frac{1}{2} W_L \, w_t^2}{g \eta_s P_m N_g} - \eta_t \, s_t\right] \Big/ \eta_s \qquad \frac{lb \;\; ft^2}{s^2 \; lb}$$

$W_L = 25{,}035$ lb   BA1 #2   CONFIG BOOK

$W_t =$ 10 ft/s   USAF
13 ft/s   USAF TRAINERS
17 ft/s   USN NON-CARRIER BASED
22 ft/s   USN
20 ft/s   AIAA/GD COMPETITION

WE USE 15 ft/s TO ALLOW FOR NO FLARE LANDINGS

NOSE GEAR TIRE	MAIN GEAR TIRE
$D_o = 22''$	$D_o = 31''$
$W = 8''$	$W = 13''$
$D = 10''$	$D = 12''$
$P_{ress} = 110$ psi	$P_{ress} = 135$ psi
$P_{MAX} = 8{,}500$ lb	$P_{MAX} = 17{,}200$ lb
$V_{max} = 190$ mph	$V_{max} = 210$ mph
LOADED R = 9"	LOADED R = 12.4"

$\eta_s = 0.85$ FOR LIQUID SPRINGS

$\eta_t = 0.47$

$N_g = 3.0$ $S_t = 0.243$ ft $= 2.91$ in

$$S_s = \frac{\frac{(0.5)(25{,}035)(15)^2}{(2)(15{,}550)(3)(32.2)} - (0.47)\left(\frac{31 - 2(12.4)}{12}\right)}{0.85}$$

$S_s = 9.8''$

$S_{s_{DESIGN}} = 11$ IN   MAIN GEAR

$d_s = \left[0.041 + 0.0025(15{,}550)^{1/2}\right]12 = 4.23$ IN

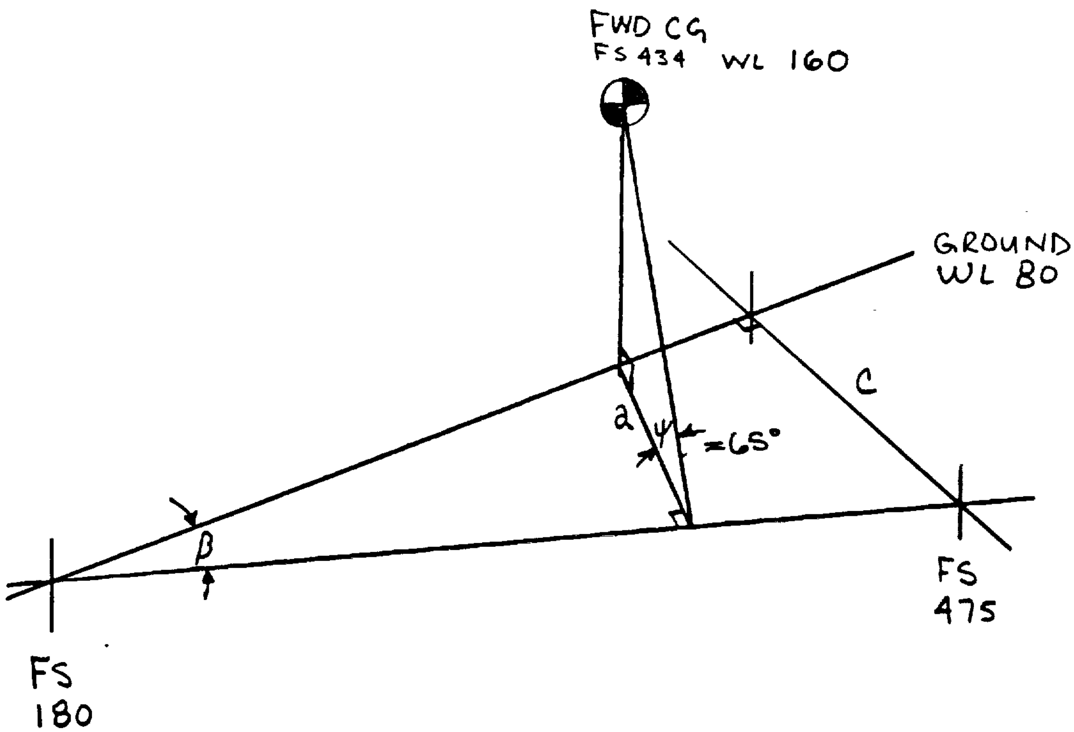
$$S_s = \frac{\frac{(0.5)(25,035)(15)^2}{(32.2)(8,450)(8.3)} - (0.47)(22-18)}{0.85} \qquad S_t = .157$$

$$S_s = 15.4 \text{ IN}$$

$$S_{s_{DESIGN}} = 16 \text{ IN}$$

$$d_s = \left[0.041 + 0.0025(8,450)^{1/2}\right]12 = 3.25 \text{ IN}$$

THESE STRUT SIZES ARE REFLECTED IN FIGURES OF CHAPTER 12



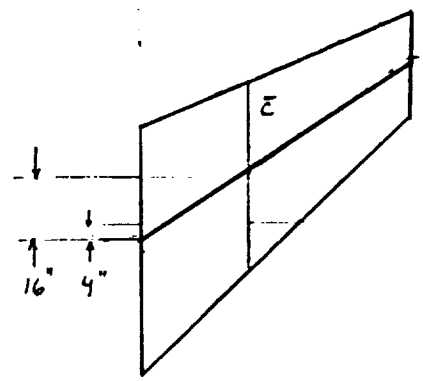
$\tan 65 = (160-80)/a \qquad a = \dfrac{80}{\tan 55} = 37.3''$

$\sin \beta = a/(434-180) \qquad \beta = 8.44°$

$\tan \beta = \dfrac{c}{475-180} \qquad c = 44''$

MAIN GEAR AT LEAST 44 " OUTBOARD

(SEE FIGURE 11.3)



SCALE 1/50

$\bar{c}_h = 3.67$ ft

$S_h = 40$ ft^2

FC		$\bar{q}$ (PSF)	$C_{L\alpha_h}$ (1/RAD)	$\alpha_{h_{MAX}}$ (DEG)	L (lb)
1	SUB	59.25	3.457	12	858
2	SUB	1069.46	4.380	4	6540
3	SUB	825.20	4.510	6	7794
4	SUB	677.12	4.510	7	7462
5	SUB	356.36	4.510	12	6732
6	SUP	633.53	3.860	8	6822
7	SUP	1126.27	3.002	6	7082
8	SUB	175.48	4.220	12	3102

MOMENT ARM = IN

MAX MOMENT = 7082 lb × 16" = 113,312 lb in

MAX FORCE = 22,662 lb

ASSUME 5000 PSI ELECTROHYDROSTATIC

22662 = 5000 · A

A = 4.53 DIAM = 2.41 IN

FROM AIRPLANE FLIGHT DYNAMICS: Pt 1 ROSKAM

$$HM = \bar{q} S_f \bar{c}_f C_h$$

FROM FIG 5.22 ASSUME MAX $C_h = 0.25$

$$S_f = 13.75 \text{ ft}^2$$

$$\bar{C}_f = 32 \text{ in}$$

$$\bar{q} = 633.53 \text{ psf}$$

$$HM = 70{,}000 \text{ lb in}$$

FROM SMITH, H., <u>AEROSPACE MATERIALS + PROCESSES</u>, LAWRENCE, KS, 1988

$$F_{su} = 75{,}000 \text{ psi}$$

BAR RADIUS = 0.67 in

INSIDE DIAM = 1.34 in

WING THICKNESS = OUTSIDE DIAM ≈ 4"

O.D - I.D. = 2.66 in

ASSUME 5000 psi

SOLVE FOR LENGTH

$$2.66 \times L \times 5000 \times 0.67 = 70{,}000$$

$$L = 8 \text{ in}$$

APPENDIX 5

The purpose of this appendix is to present calculations to determine the estimated increase in drag due to the open internal weapons bay and to estimate the maximum yawing moment that is induced by the firing of the Vulcan cannon.

Appendix 5: Table of Contents

ESTIMATION OF DRAG INCREASE DUE TO WEAPONS BAY & DOORS

FLIGHT CONDITION

$M = 1.6$
$H = 30{,}000\ ft$
$V_a = 994.7\ ft/s$
$\rho = 0.0008893\ slug/ft^3$
$\mu = 3.106 \times 10^{-7}\ lb\cdot sec/ft^2$

DOOR DIMENSIONS

EACH PANEL : $L = 156\ in$ (SEE FIGURE 12.10)
$W = 9\ in$

REYNOLDS NUMBER CALCULATION

$$R_{N_L} = \frac{(0.0008893)(1.6)(994.7)(156)(\frac{1}{12})}{(3.106 \times 10^{-7})}$$

$$R_{N_L} = 5.924 \times 10^{7}$$

FLAT PLATE DRAG ESTIMATION

$$C_D = 0.455(\log R_{N_L})^{-2.58} \quad \text{(TURBULENT FLOW)}$$

$$C_D = 0.455(\log 5.924 \times 10^{7})^{-2.58}$$

$$C_D = 0.002293$$

MULTIPLY BY 2 TO ACCOUNT FOR BOTH SIDES OF PANEL AND BY 2 TO ACCOUNT FOR CAVITY DRAG (REF. DR. ROSKAM)

$$C_D = 0.0092$$

NORMALIZE TO REFERENCE AREA

$$S_{DOOR} = (2)(156\text{ in})(9\text{ in})\left(\frac{1}{144}\right) = 19.5\text{ ft}^2$$

$$S_{WING} = 347.9\text{ ft}^2$$

$$C_D = (0.0092)(19.5/347.9)$$

$$C_D = 0.00052$$

COMPARISON TO AIRPLANE ZERO-LIFT DRAG

$$C_{D_0} = 0.04038$$

$$\%\, C_{D_0} = \frac{0.00052}{0.04038}(100) = 1.3\,\%\ \text{INCREASE}$$

MAXIMUM CANNON RECOIL INDUCED YAWING MOMENT ESTIMATION

MAXIMUM RECOIL FORCE = 3,980 lb @ 6,000 RPM REF 12.1

FROM FIG 12.1

YAWING MOMENT = (3,980 lb)(35 in)(1 ft/12 in)

N_{GUN} = 11,600 ft·lb

MAX FIRING TIME ESTIMATION

GUN SPOOL-UP TIME = 1/3 SECOND REF. 12.2

MEAN FIRING RATE DURING SPOOL-UP = $\sqrt{2}$ (100/s)

= 70 RPS

(70 RPS)($\frac{1}{3}$ SEC) = 23.1 ROUNDS

AMMO REMAINING = 376.9 RDS

FIRING TIME = $\frac{376.9}{100} + \frac{1}{3}$

= 4.10 SECONDS