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A Preliminary Design Proposal for a Maritime Patrol Strike Aircraft:

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MPS-2000 Condor

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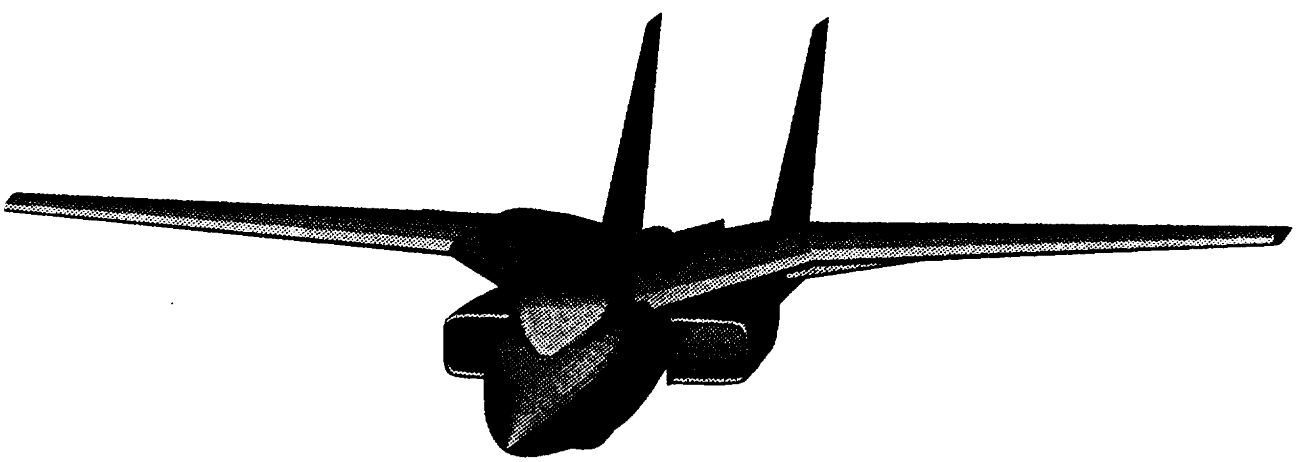

In Response to 1993/1994 AIAA / Mcdonnell Douglas Corp.
Graduate Team Aircraft Design RFP

The University of Kansas Graduate Team

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List of Symbols

Symbol	Description	Units
A,B	Regression Coefficients	-
A	Aspect Ratio	-
C_D	Drag Coefficient	-
C_L	Lift Coefficient	-
C_μ	Momentum Coefficient	-
C	Cost	USD ($)
e	Oswald's Efficiency Factor	-
E	Endurance	hours
h	Altitude	ft
L/D	Lift to Drag Ratio	-
m_dot	Mass Flow Rate	lbm/s
R	Range	nm
S	Reference Area (Wing)	ft^2
S	Field Length	ft
S"	Effective Blown Flap Area	ft^2
T	Thrust	lbs
V	Velocity	ft/s or kts
W	Weight	lbs

1. Introduction and Interpretation of RFP

The four member graduate design team assembled to submit a proposal for the 1993/1994 RFP at the University of Kansas has designed a four seat, variable swept wing, twin turbofan aircraft with STOL capabilities. The aircraft is named the *MPS-2000 Condor* and is capable of carrying air-to-surface or air-to-air weapon systems along with attack and surveillance radar and IFR systems. The aircraft has a cruise range of 800 nautical miles, a loiter of 4 hours, and a dash speed of 500 kts.

The Request for Proposal (RFP) requirements and the Mission Profile for the *Condor* are summarized in Sub-sections 1.1 and 1.2 respectively.

1.1 RFP Requirements

The AIAA Request for Proposal calls for a Maritime Patrol Strike Aircraft design to meet the needs of the U.S. Navy to contain regional conflicts and deter large scale aggression. The aircraft must be able to conduct surveillance over large expansions of water and land with little support for long periods of time. The RFP dictates that the proposed design must be a four seat multi-mission combat aircraft able to operate from a TARAWA or Wasp class amphibious assault ship. The aircraft must be able to intercept small, fast surface crafts and other small, armed aircraft. Other variants may included capabilities to conduct drug interdiction, law enforcement and search and rescue missions.

1.2 Mission Profile

The rigorous mission profile for the Maritime Patrol Strike Aircraft as dictated in the RFP can be summarized in Table 1.1:

Table 1.1: *Mission Specification of the Condor*

Phase	Description	Requirement
Phase 1:	Prepare for Launch	15 minutes
Phase 2:	Warm-up & takeoff	10 minutes
Phase 3:	Cruise 400 nm. to patrol station	
Phase 4:	Loiter on station	4 hours / Speed ≤ 200 kts.
Phase 5:	Detect targets and accelerate to intercept speed	10 minutes
Phase 6:	Dash at intercept speed 100 nm.	Mach 0.9 or 500 kts @ S/L
Phase 7:	Attack and destroy target	Two passes maximum
Phase 8:	Return 400 nm. to vessel	

To meet the mission profile requirements, the *MPS-2000 Condor* is designed with the following payload capabilities:

- Anti-air Weapons: 2 Raytheon & Ford Instrument AIM-9 Sidewinder missiles
  2 Raytheon & Hughes AIM-120A Amraam missiles
- Anti-surface Weapons: 2 McDonnell Douglas RGM-84 Harpoon missiles
- Standard flare & Chaff Dispensers
- ESM/ECM
- IR equipment
- UHF/VHF
- Control displays / Video Display Terminals
- Radar: 250" x 40" elliptical / 120 deg. coverage / 360 deg total coverage / 250 KVA
- Crew: 4 members (pilot, navigator/co-pilot, weapons systems officer, tactical officer)

The performance constraints imposed on the *Condor* during completion of the mission profile are listed in Table 1.2.

Table 1.2: *Performance Constraints for the Condor*

Flight Condition	Requirement
Critical Field Length (Ground run)	500 ft
Minimum initial cruise altitude	38,000 ft.
Cruise Speed	≥ 250 kts.
Loiter Speed	≤ 200 kts.
Dash speed	Mach 0.9 @ alt.
	500 kts. @ S/L
Return from loiter station after four hours with full payload	
Launch and recovery from WASP or TARAWA class amphibious assault ship	

This proposal presents the preliminary design aspects of the *Condor* as the apply to restrictions and requirements set forth by the RFP. The following chapters are keyed according to the RFP requirements.

2. Technical Approach to Meet RFP Requirements

The approach taken by the this design team toward the final design of the *Condor* presented in this proposal is based on methods found in References 2 to 9. After a full understanding of the RFP is believed to be achieved, the preliminary design of the aircraft entails weight sizing, performance sizing, aerodynamic characteristics, high lift capabilities, weight and balance and performance. This step-by-step process is presented in Sections 2.1 through 2.6 respectively.

2.1 Weight Sizing

Preliminary weight sizing of the *Condor* is based on methods found in Reference 2. The first approach involves studying similar aircraft and using statistical data to estimate takeoff and empty weights. This analysis, as conducted for the *Condor*, is described in Section 2.1.1. The fuel weight of the aircraft can be estimated from the mission specification as presented in Section 2.1.2.

2.1.1 Regression Coefficients and Plots.

To find an initial estimation of the takeoff and empty weight, the following relationship was used:

$$W_E = \mathrm{Log}_{10}\left[\frac{\mathrm{Log}_{10}(W_{TO}) - A}{B}\right] \qquad \text{Eq. 2.16 Ref. 2}$$

A = -0.5269 B = 1.177

The constants **A** and **B** were found from a linear-logarithmic regression analysis of comparable fighters with variable sweep wing. Table 2.1.1 shows the airplanes considered with their takeoff and empty weights, and Figure 2.1.1 shows the linear-logarithmic regression extrapolation.

Table 2.1.1: *Similar Airplanes*

Aircraft	W_{TO} (lbs)	W_E (lbs)
F-14	74,349	41,780
Tornado	61,620	30,620
Mig-23	39,250	22,485
Su-24M	87,520	41,885
Mig-27	44,750	26,252
F-111A	91,500	46,172

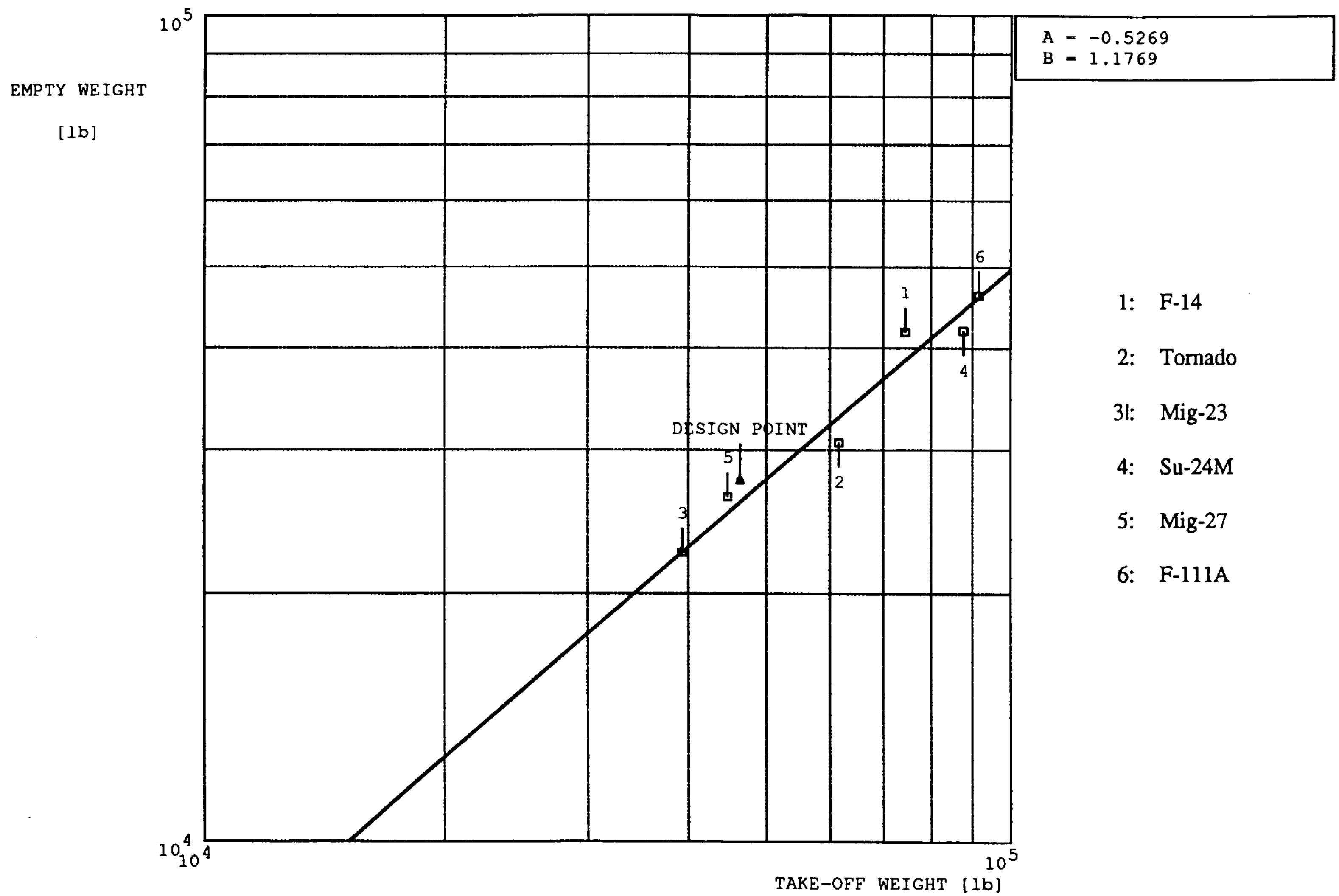

Figure 2.1.1: Weight Trends of Similar Aircraft

2.1.2 Mission Weights

In this sub-section the mission weights for each flight phase will be presented. Referring to Chapter 2 of Reference 2, typical values of mission weight fractions can be found. Some of these values are fairly constant across designs and were assumed to apply to the *Condor*. The values dependent upon the design of the *Condor* are those for cruise, loiter, and dash. These weight fractions were obtained using Class II methods found in Reference 8. The mission profile that defines these weights is shown in Figure 2.1.2. The mission weights, averaged for each flight phase, resulting from this analysis are:

- W_{TO} = 46,500 lbs
- $W_{cruise1}$ = 43,600 lbs
- W_{loiter} = 39,500 lbs
- W_{dash} = 35,400 lbs
- $W_{cruise2}$ = 33,900 lbs
- W_{land} = 34800 lbs (max.)
- W_{land} = 32500 lbs (normal)

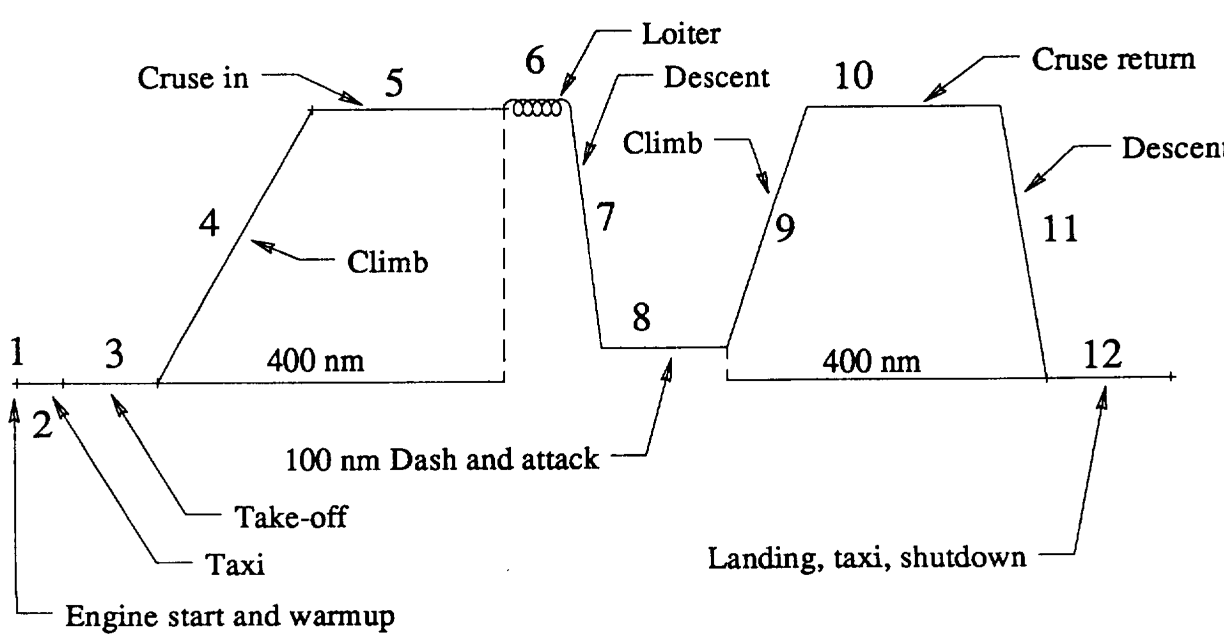

Figure 2.1.2: Mission Profile for the Condor

2.2 Performance Sizing

From simple analysis of the performance requirements as they apply to a particular aircraft, wing area and engine size can be estimated as described in Sub-sections 2.2.1 and 2.2.2.

2.2.1 Wing Loading

This sub-section contains the selection rationale for the wing loading of the *Condor* attack jet. Attempting to determine a wing loading that would be preferable in each of the flight phases is difficult. The cruise and loiter phase performance is better for medium to high values of wing loading, while the takeoff and landing conditions prefer low values. Fortunately, performance in each phase is better with high aspect ratio wings. Two things were done to meet the demands for wing loading. A variable geometry wing was used and blowing over the flaps was employed in the takeoff and landing phases. The wing loading selected was tied closely to the takeoff conditions as they were found to be flight critical in terms of the lift coefficient and the thrust to weight required by the *Condor*. This is shown in Figure 2.2.3. Always attempting to find a better compromise that would save weight and complexity led to the selection of a wing loading of 66 pounds per square foot in takeoff. This value is typical for aircraft with similar missions and capabilities. It is not a low value, but is medium when compared.

2.2.2 Thrust to Weight Ratio

This sub-section contains the selection rationale for the thrust to weight ratio of the *Condor* attack jet. Determining the thrust to weight ratio was full of recurring design changes and the source of much frustration. The process has the single purpose of decreasing the ratio as low as possible while still being able to sufficiently complete the mission requirements. Inherent in this process is the engine selection process. This is often the most difficult job. Using the performance relationships in Reference 2, the selection process for thrust to weight ratio was tied closely to that of wing loading. This can be seen in Figure 2.2.3. Once the desired wing loading and lift coefficient were obtained, the corresponding thrust to weight ratio was determined. The problem was then to find an engine that could deliver the required performance at the least weight and smallest size. The thrust to weight ratio required by the *Condor* is approximately 60%. The engine selected to power the *Condor* is the BMW 710-15. This engine has the "best" combination of available thrust, specific fuel consumption across the flight regime, mass flow, and engine weight.

2.2.3 Summary

The airplane and engine sizing process obviously is geared toward the most beneficial performance combination, but there are economic considerations as well. Two of the most important and expensive components of aircraft are the wing and powerplant. Most all performance relations are in terms of the wing loading and thrust to weight ratio. As well as meeting the performance requirements, it is also desirable to minimize cost. This is another reason to have a high wing loading and low thrust to weight ratio. The values for wing loading and thrust to weight ratio for the *Condor* are:

- W/S_{TO} = 66 psf
- T/W_{TO} = 0.61

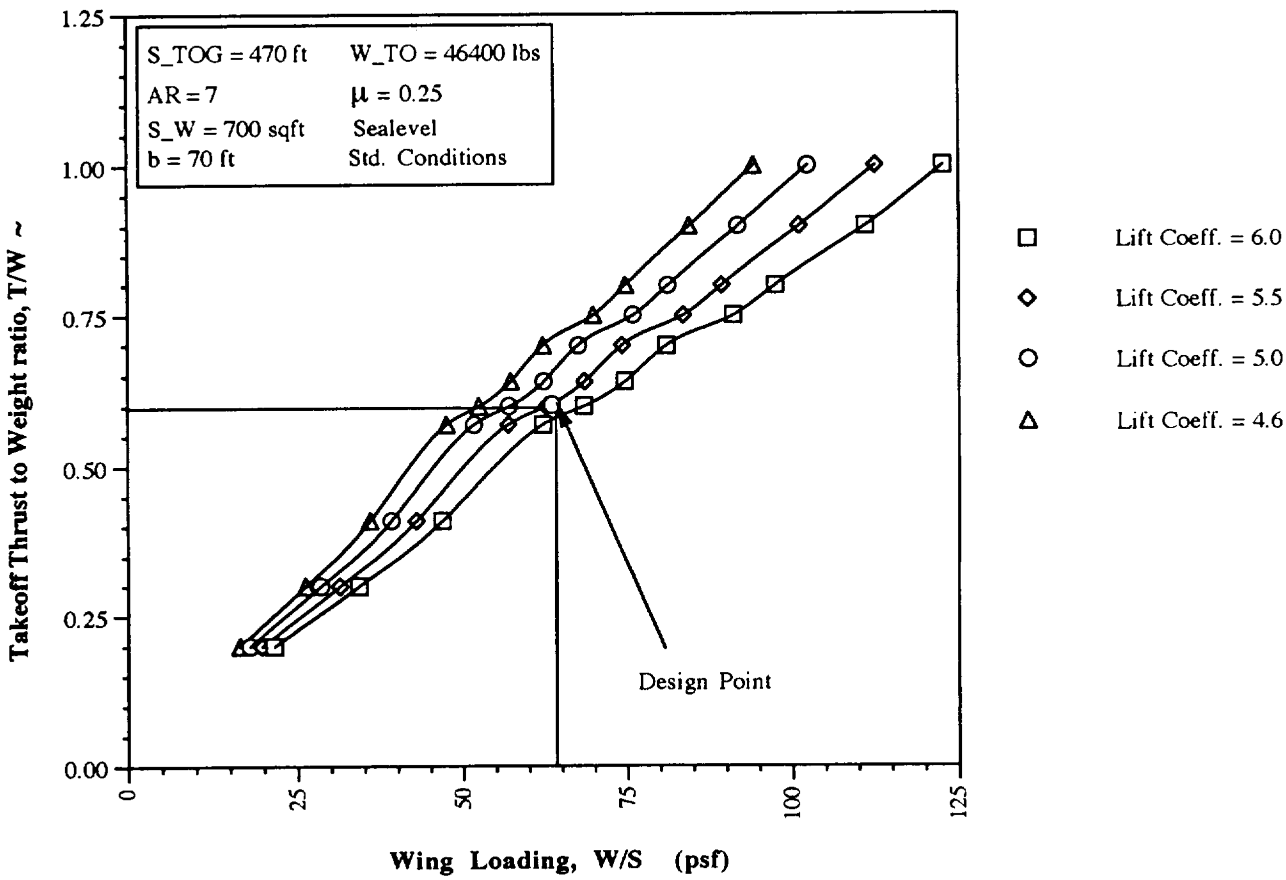

Figure 2.2.3: Takeoff Wing Loading Variation with Thrust to Weight Ratio for the Condor

2.3 Aerodynamics

Aerodynamic considerations for the *Condor*, in the scope of this preliminary design proposal, include drag analysis and stability and control analysis. From the drag analysis, lift-to-drag ratios can be determined for each flight condition to aid in the prediction of performance characteristics of the airplane. In addition, a component drag breakdown is included, with the aid of the AAA program, to display the drag contribution of each component of the airplane per flight condition. The stability and control analysis for the *Condor* includes methods to size the empennage and control surfaces. Once the required geometry of the aircraft is estimated for flight, the AAA program can be used to determine longitudinal and lateral-directional stability and control derivatives. From the stability and control derivatives, the dynamic flying qualities of the airplane can be predicted. The drag analysis and the stability and control analysis for the *Condor* can be found in Sections 2.3.1 and 2.3.2 respectively.

2.3.1 Drag

The purpose of this chapter is to summarize a Class II drag polar analysis for the *Condor*. The method for drag analysis can be found in Reference 7. Further analysis for the drag polars were computed with the AAA program of Reference 10.

2.3.1.1. Drag Breakdown

The total airplane drag coefficient for the *Condor* is broken down into the following components:

- Wing Drag
- Horizontal Tail Drag
- Vertical Tail Drag
- Fuselage Drag
- Flap Drag*
- Gear Drag*
- Canopy Drag
- Stores Drag
- Trim Drag
- Miscellaneous Drag

(* take-off and landing)

The following flight scenarios are individually examined for the *Condor*:

- Take-off
- Cruise1
- Loiter
- Dash
- Cruise2
- Landing

The drag coefficients were calculated for the total drag of each flight conditions. The variable wing selection causes a significant change in wing area in the dash condition. The unswept wing area was found to be 700 ft^2 and the fully swept wing area for the dash was found to be 823 ft^2 . This change in wing area was accounted for in the dash drag calculation. Table 2.3.1.1 shows the drag breakdown in each flight conditions.

Table 2.3.1.1 *The Drag Breakdown in Each Flight Conditions for the Condor*

	Take-off	Cruise 1	Loiter	Dash $S = 823\ ft^2$	Cruise 2	Landing
$C_{D\text{-}O\text{-}w}$	0.0062	0.0136	0.0061	0.0029	0.0080	0.0058
$C_{D\text{-}I\text{-}w}$	1.2190	0.0043	0.0101	0.0002	0.0029	0.2244
$C_{D\text{-}O\text{-}h}$	0.0022	0.0020	0.0021	0.0016	0.0020	0.0021
$C_{D\text{-}I\text{-}h}$	0.0372	0.0055	0.0096	0.0000	0.0048	0.0715
$C_{D\text{-}O\text{-}v}$	0.0016	0.0015	0.0015	0.0012	0.0015	0.0016
$C_{D\text{-}I\text{-}v}$	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
$C_{D\text{-}O\text{-}fus}$	0.0084	0.0083	0.0077	0.0069	0.0078	0.0078
$C_{D\text{-}I\text{-}fus}$	0.0231	0.0000	0.0000	0.0000	0.0000	0.0004
$C_{D\text{-}flap}$	0.0966	0.0000	0.0000	0.0000	0.0000	0.0911
$C_{D\text{-}gear}$	0.0111	0.0000	0.0000	0.0000	0.0000	0.0112
$C_{D\text{-}can}$	0.0006	0.0006	0.0006	0.0000	0.0006	0.0006
$C_{D\text{-}store}$	0.0005	0.0006	0.0005	0.0005	0.0006	0.0005
$C_{D\text{-}trim}$	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
$C_{D\text{-}misc}$	0.0028	0.0028	0.0028	0.0028	0.0028	0.0028
TotalDrag	**1.4032**	**0.0392**	**0.0410**	**0.0161**	**0.0311**	**0.4198**

For the take-off flight condition, the benefit of using a blowing system was realized. The blowing system will be discussed in Section 2.4.2.

2.3.1.2. Drag Polar

With the results of drag breakdown, the drag polar was computed using the AAA program. Assuming that the equation is parabolic, the drag polar equations for each flight phase were determined as follows:

- Take-off: $A = 7,\ e = 0.959$

  $C_D = 0.0925 + 0.0474\ C_L^2$ Eq. 2.3.1

- Cruise 1: $A = 7,\ e = 0.082$

  $C_D = 0.0304 + 0.0557\ C_L^2$ Eq. 2.3.2

- Loiter: $A = 7,\ e = 0.93$

  $C_D = 0.0220 + 0.0488\ C_L^2$ Eq. 2.3.3

- Dash: $S = 823\ ft^2,\ A = 3,\ e = 0.39$

  $C_D = 0.0158 + 0.1163\ C_L^2$ Eq. 2.3.4

- Cruise 2: $A = 7,\ e = 0.80$

  $C_D = 0.0241 + 0.0567\ C_L^2$ Eq. 2.3.5

- Landing: $A = 7,\ e = 0.89$

  $C_D = 0.0804 + 0.0512\ C_L^2$ Eq. 2.3.6

The drag polars are shown in graphical form in Fig. 2.3.1.1. The lift-to-drag ratios for each flight conditions were found as following Table 2.3.1.2

Table 2.3.1.2 *The Lift-to-Drag Ratios for Each Flight Conditions*

	Takeoff	Cruise 1	Loiter	Dash $S = 823\ ft^2$	Cruise 2	Landing
Weight (lbs)	46,400	43,600	37,400	35,400	33,900	32,500
Lift Coeff.	5.258	0.398	0.624	0.051	0.351	2.574
Altitude (ft)	0	38,000	38,000	0	38,000	0
Mach Number	0.09	0.72	0.53	0.76	0.68	0.11
L/D	3.8	10.1	15.2	3.2	11.3	6.1

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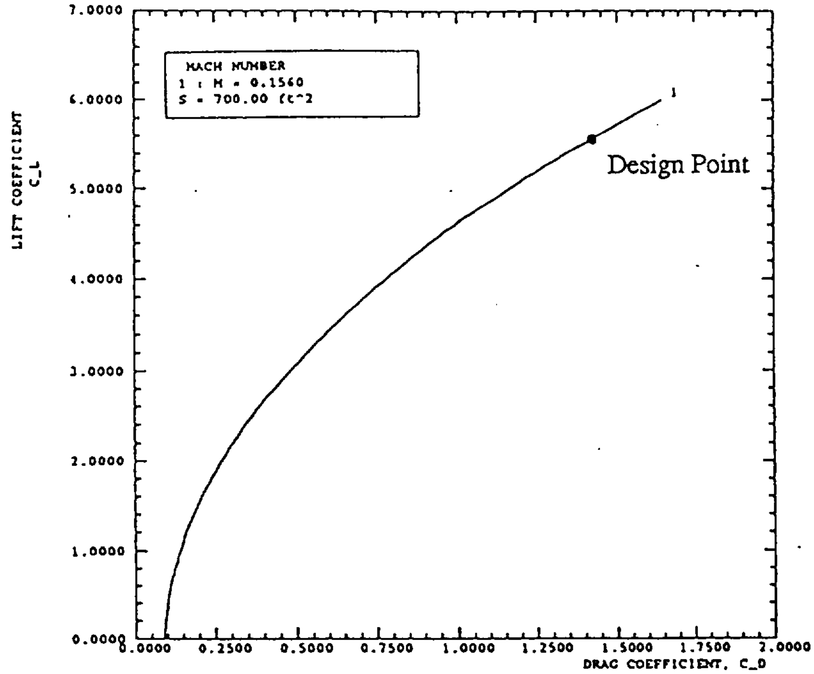

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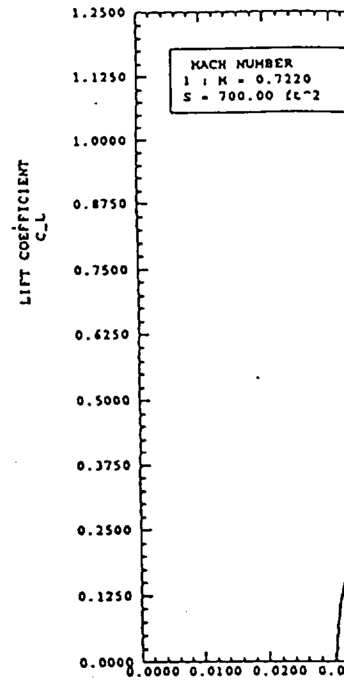

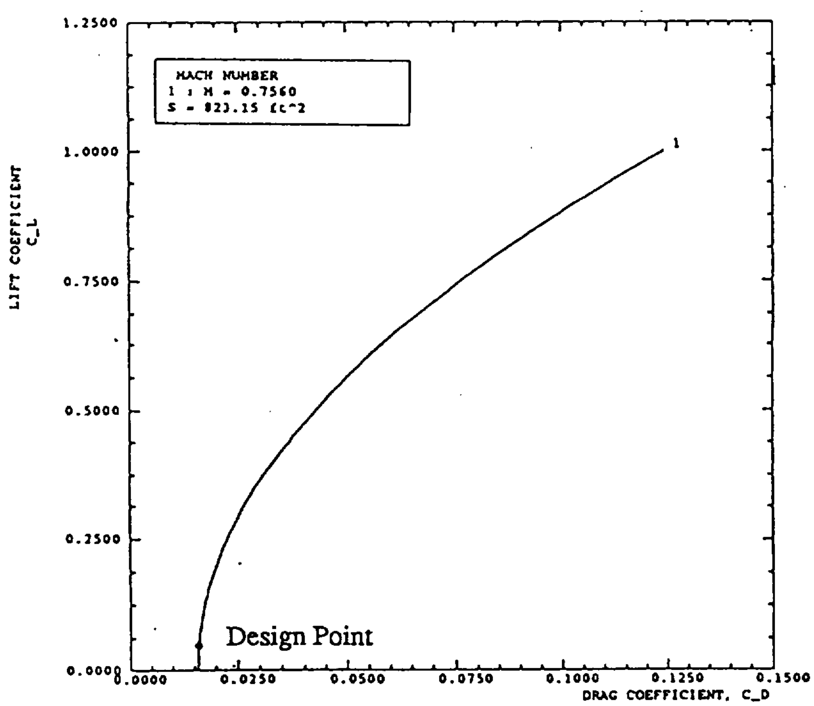

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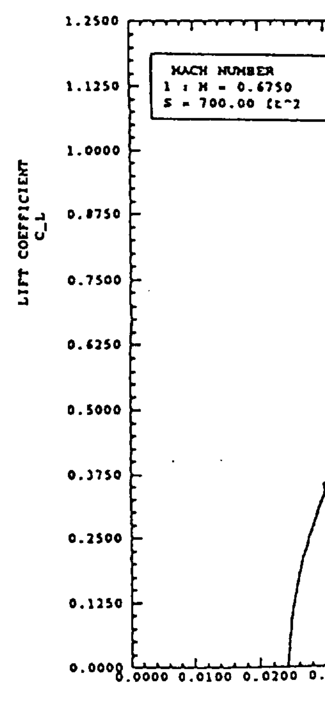

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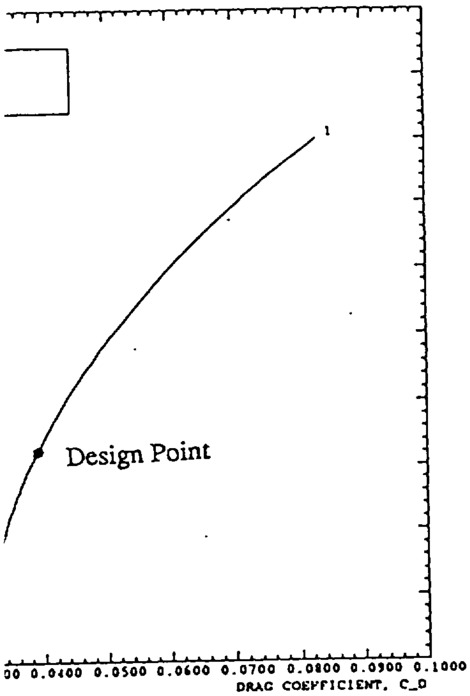

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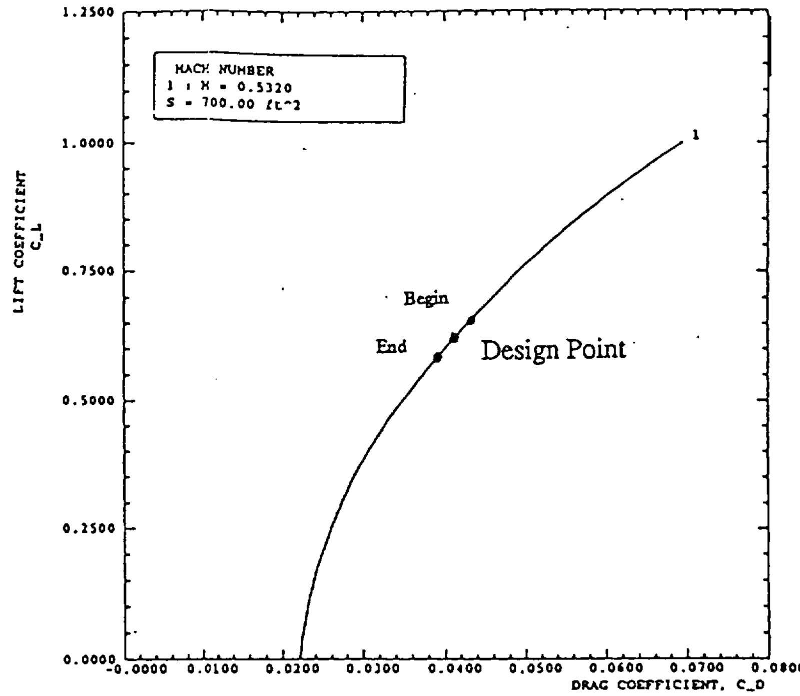

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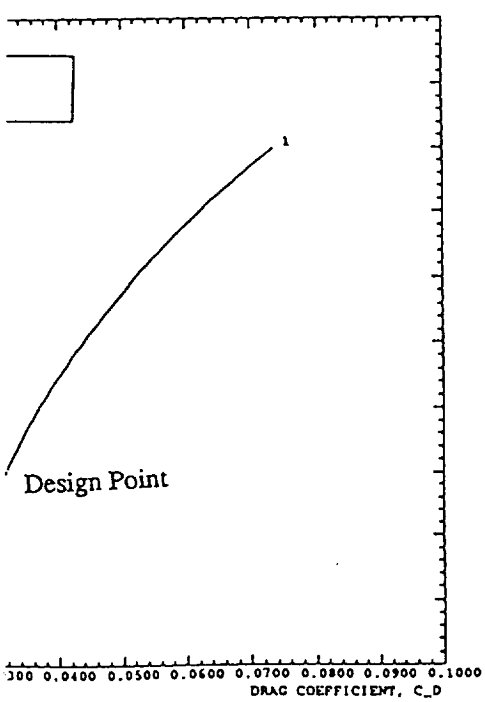

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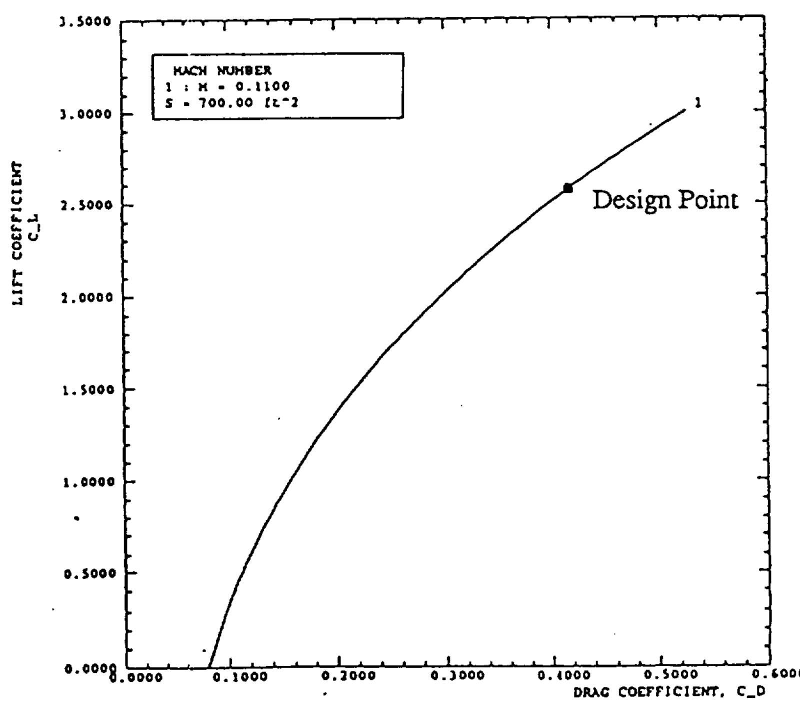


< Landing >

: Drag Polars for the Condor

2.3.2 Stability and Control

The stability and control analysis for the *Condor* includes preliminary methods to predict empennage and control surface sizes (Section 2.3.2.1) and estimating longitudinal and lateral-directional stability and control (S&C) derivatives (Section 2.3.2.2). The S&C derivatives are compared with similar aircraft to validate initial sizing. The methods for stability and control analysis are primarily based on References 7,8,10 and 11.

2.3.2.1 Empennage/Control Surface Sizing

Preliminary empennage and control surface sizing for the *Condor* is based on comparison with similar aircraft. The first step in the sizing process is to determine the airplane configuration. For the *Condor*, a conventional two surface, tail aft configuration is selected. Longitudinal control is achieved with the use of variable incidence horizontal stabilizers. Ailerons provide lateral control while directional control is obtained with rudders.

For the horizontal and vertical tail sizing, a volume method is used as suggested by Reference 3. The tail surfaces are sized from statistical relations with similar aircraft based on wing reference area and empennage moment arms. A second approximation to the horizontal and vertical tail sizing for the *Condor* includes preliminary estimation of the longitudinal and directional stability of the aircraft based on center of gravity and and/or aerodynamic center locations. A more involved S&C derivative analysis (Section 2.3.2.2) confirms or discredits the tail size assumptions. The results of the tail sizing iterations are listed as follows:

- Horizontal Tail Area: 240 ft^2
- Vertical Tail Area (total): 200 ft^2

Specific information on the remaining tail geometry parameters can be found in Section 3.1.

Typical of a military fighter or attack aircraft, the *Condor* exhibits relatively large vertical and horizontal tail projections due to the short coupled fuselage. A vertical tail span constraint exists for the *Condor* due to the requirement listed in the RFP for operation on a TARAWA class assault ship. For the ability to store the aircraft under the deck, the total height of the aircraft must be under 18.5 feet for the elevator. Therefore, to meet this requirement, the vertical tail area is divided evenly into two surfaces to reduce the physical span of the tails. The total height of the aircraft to meet the elevator requirement can be confirmed in Section 3.1.

For longitudinal stability analysis and horizontal tail sizing, the effects of the flap blowing system on the horizontal tail must be accounted for in the takeoff condition. As suggested by Dr. Roskam of the University of Kansas, it is assumed that the downwash acting on the horizontal tail is equal to the flap deflection angle with the blowing system operating. An arbitrary assumption is made that the dynamic pressure acting on the horizontal tail is approximately 20% of that found at the exit of the blowing nozzle. This assumption renders a horizontal tail dynamic pressure five times greater than the free-stream dynamic pressure.

Sizing methods for the control surfaces for the *Condor* follow assumptions made from statistical data as suggested in Reference 3. Longitudinal and lateral-directional S&C derivatives with the assumed control surfaces geometries are compared with those found in similar aircraft.

- **Longitudinal Control:**

  Variable Incidence Stabilizer: Area: 240 ft^2

- **Lateral Control:**

  Flaperons: Full Span

  30% local chord

  Differential Stabilizer: Full Span

- **Directional Control:**

  Rudders (two): 85% span

  30% local chord

A variable incidence stabilizer, as opposed to an elevator, is found to be consistent with similar aircraft for longitudinal control power and trim (See Section 3.2.2.2). Full span flaperons and differential stabilizer have been found to be needed for adequate roll performance in the takeoff condition (See Section 2.3.2.3). This assumption for roll performance is primarily based on C_{l_p} in the takeoff condition with the flap blowing system operating. With circulation control, the lift curve slope of the airfoil is 15 rad^{-1}. This lift curve slope yields a C_{lp} of approximately -1.0 rad^{-1}(See Section 2.3.2.2). This relatively large derivative requires a large amount of roll control power for adequate roll performance[11].

2.3.2.2 Stability and Control Derivatives

In the stability and control analysis for the *Condor*, as conducted on the AAA program, the following six forces and moments are assumed to be acting on the aircraft in all flight conditions:

- Drag
- Side Force
- Lift
- Roll
- Pitch
- Yaw

Two flight scenarios are assumed for the six force and moment expressions: steady state and perturbed state flight. In steady state flight, the aerodynamic forces and moments are analyzed as derivatives dependent on angle of attack, sideslip angle (small angles), and control surface deflection. Perturbed state flight stability studies change in the aerodynamic forces and moments of an airplane in a steady state flight due to a sudden change in the following motions:

- Forward Velocity
- Side Velocity
- Downward Velocity
- Roll Rate
- Pitch Rate
- Yaw Rate

The stability derivatives are estimate by summing the various component contributions of the wing, horizontal tail, etc. for both flight scenarios.

Longitudinal Stability and Control:

Analysis for longitudinal stability and control derivatives for the *Condor* include the following forces and moments:

- Drag
- Lift
- Pitching Moment

The derivatives are analyzed for takeoff, cruise, loiter, dash and landing conditions. As expected, the circulation control, when operating, has significant effects on the longitudinal stability and control of the aircraft. Table 2.3.2.1 displays the longitudinal stability and control derivatives for takeoff and landing configurations. In the takeoff configuration, the circulation control is operational while in the landing configuration it is not. For comparison, typical values for similar aircraft are also supplied from Reference 13.

Table 2.3.2.1 *Longitudinal Stability and Control Derivatives for the Condor*

Derivative	Takeoff Value (1/rad)	Landing Value (1/rad)	Recommended Range
Drag due to Angle of Attack	6.75	0.48	0.02 to 0.20
Airplane Lift Curve	13.9	4.88	1.0 to 7.0
Lift due to Horizontal Stabilizer	4.24	1.12	Not Available
Lift due to Rate of Angle of Attack	9.04	2.48	-5.0 to 5.0
Lift due to Speed	0.054	0.35	-0.1 to 0.3
Lift due to Pitch Rate	26.3	7.76	0.0 to 8.0
Pitching Moment due to Angle of Attack	-1.06	0.87	-3.0 to 0.5
Pitching Moment due to Horizontal Stabilizer	-8.46	-1.87	Not Available
Pitching Moment due to Rate of Angle of Attack	-18.0	-4.14	-10.0 to 3.0
Pitching Moment due to Speed	0.10	0.15	-0.2 to 0.5
Pitching Moment due to Pitch Rate	-43.5	-7.05	-20.0 to 0.0

A method to analyze the longitudinal trim capabilities of the aircraft is to produce a figure relating the lift curve to the pitching moment due to angle of attack derivatives for the airplane. The effects of angle of attack and elevator deflection can then be added to determine the conditions necessary to maintain a stable airplane. For the *Condor*, two trim diagrams are displayed for the takeoff and landing conditions in Figures 2.3.2.2.1 and 2.3.2.2.2 respectively. As can be seen from the figures, the *Condor* achieves trim in the takeoff condition with a blown flap lift coefficient of 4.4 at 5.5 degrees of angle of attack and 7.0 degrees of horizontal tail incidence. Without the circulation control, the *Condor* can be trimmed at 0.0 degrees angle of attack and approximately 2.0 degrees of horizontal tail incidence. The trim diagrams in Figures 2.3.2.2.1 and 2.3.2.2.2 were constructed with the use of the AAA program.

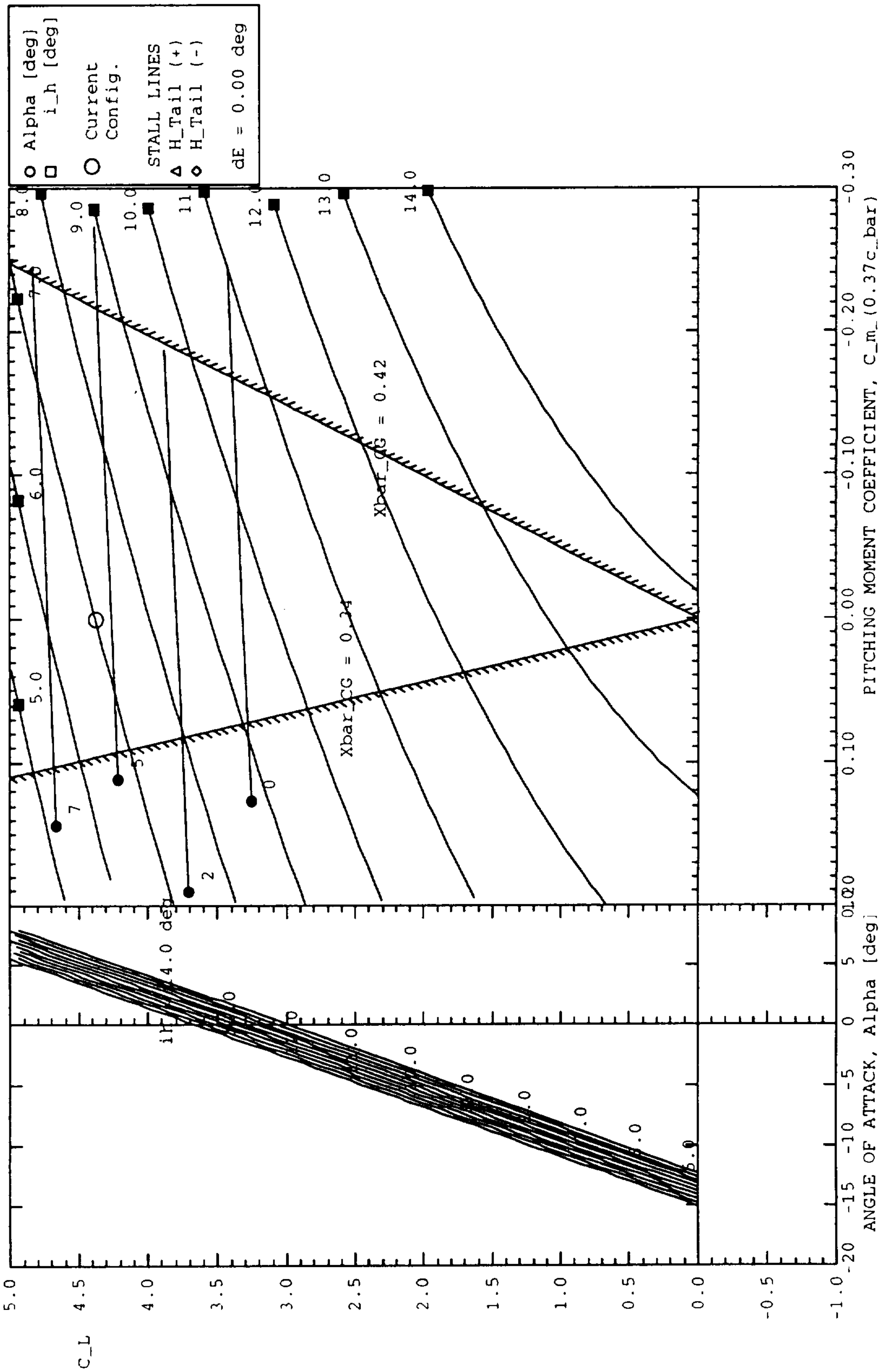


Figure 2.3.2.2.1: Trim Diagram for the Takeoff Condition

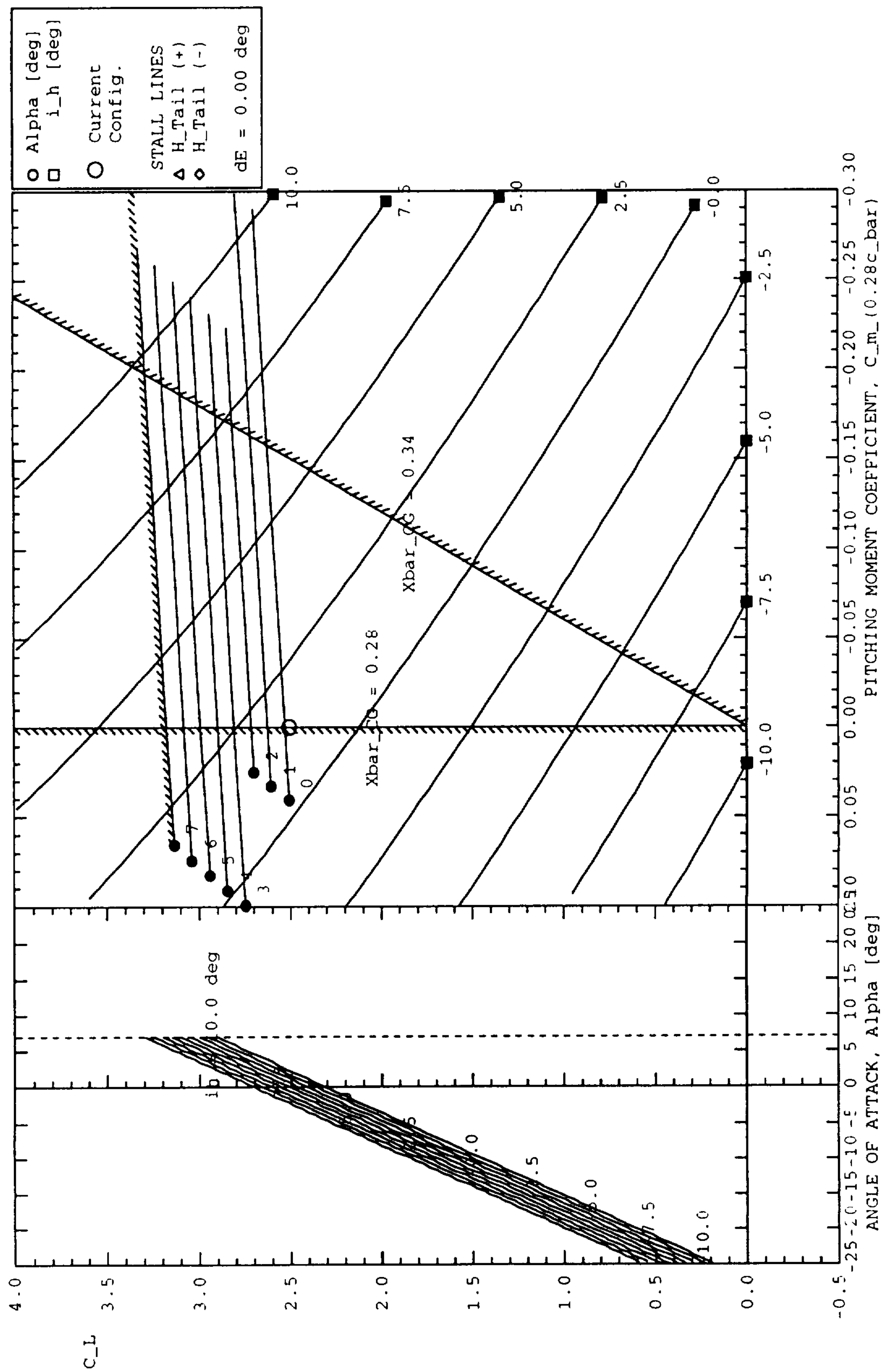


Figure 2.3.2.2.2: Trim Diagram for the Landing Condition

Lateral and Directional Stability

Lateral and directional stability includes the following aerodynamic forces and moments acting on the aircraft:

- Side Force
- Rolling Moment
- Yawing Moment

The steady state and perturbed state derivatives for the *Condor* are displayed in Table 2.3.2.2 for the takeoff condition only. This condition was found to be the most critical for lateral and directional stability due to the large takeoff weight and low dynamic pressure. For simplicity, the derivatives for the remaining flight conditions have been omitted from this report. For comparison, corresponding derivatives for similar aircraft are also display from Reference 13.

Table 2.3.2.2 *Lateral and Directional Derivatives for the Condor in Takeoff Condition*

Derivative Name	Takeoff Value (1/rad)	Recommended Range
Side Force due to Rudder	0.256	0 to 0.5
Side Force due to Sideslip	-0.446	-0.1 to -1.5
Side Force due to Roll Rate	-0.042	0 to 1.2
Rolling Moment due to Rudder	0.012	-0.04 to 0.04
Rolling Moment due to Aileron	0.067	0.0 to 0.3
Rolling Moment due to Sideslip	-0.150	-0.30 to 0.06
Rolling Moment due to Roll Rate	-0.713	-0.1 to -0.8
Rolling Moment due to Yaw Rate	1.109	0.0 to 0.4
Yawing Moment due to Rudder	-0.075	0.0 to -0.15
Yawing Moment due to Aileron	-0.057	-0.08 to 0.08
Yawing Moment due to Sideslip	0.080	0.0 to 0.4
Yawing Moment due to Roll Rate	-0.527	-0.5 to 0.1
Yawing Moment due to Yaw Rate	-0.302	0.0 to -1.0

2.4 High Lift

To meet the requirements listed in the RFP of a 500 ft. field length, a description of the development and analysis of the high lift system for the *Condor* is presented in Sub-sections 2.4.1 and 2.4.2.

2.4.1 Flap Sizing and Placement

This sub-section contains the details of the flap sizing and placement for the *Condor* Attack jet. The flap sizing was driven by the takeoff requirement of 500 feet. It was apparent from the flap and takeoff analyses that there was no capability to meet this requirement using conventional full-span flaps. The required lift coefficients were out of the attainable range for conventional flap technology. An augmentation method was selected in that of a blown flap system which will be discussed in the following sub-section. The flaps were sized using the AAA program to produce a lift coefficient of 5.4, with the blowing assistance in takeoff. Using methods for blown flaps analysis and flap analysis from Reference 7, the following results were obtained:

- Full span Fowler flaps
- Takeoff flap deflection of 24 degrees
- Takeoff angle of attack of 14 degrees

While the AOA may seem high, it is within the range outlined in Chapter 7 of Reference 14 when blowing over the flaps is involved.

2.4.2 Circulation Control

This sub-section contains the details of the circulation control system, or blowing system. The blown flap is a technology that has been known for some time, but is new to the market. McCormick states in his book that the concept was first investigated in 1933[14]. It works by blowing a stream of air over the flap at high speed and is sometimes referred to as a "jet flap.". When the flap is deflected, the stream of air bends with it. This has the effect of keeping the flow attached and allows for greater angles of deflection and/or higher lift coefficients and angles of attack[14]. The circulation control system was sized around the takeoff requirement of 500 feet. At the known takeoff weight, the required lift coefficient was 5.3. An analysis showed that the blowing system:

- must be full span
- requires 50 lbm/sec of air
- blows at a velocity of 470 fps
- produces a negative pitching moment of -1.8 about the quarter chord

Iterations were performed to decrease the induced stable pitching moment to a level that was manageable by the flight control system and to insure that the mass flow was deliverable by the engine or APU. In this case the APU serves as the source of the blown air.

2.5 Weight and Balance

Once preliminary weight and sizing parameters of the aircraft are determined, a more in-depth weight and balance analysis is needed to justify a configuration. If a more detailed weight analysis (see Section 2.5.1) varies significantly from preliminary weight and sizing, then the whole process must be reiterated. In a similar fashion, if the aircraft cannot be balanced appropriately, as described in Section 2.5.2, then the preliminary configuration of the aircraft must be modified.

2.5.1 MIL-STD-1374 Weight Breakdown

This sub-section contains the MIL-STD-1374 weight breakdown for the *Condor* attack jet. The MIL-STD-1374 reporting forms were used as a guideline for reporting the weight and balance statements. The methods of reference 6 as well as the AAA program were used in the determination of the aircraft component weights. Some assumptions were made in the Class II weight estimation:

Structural Weight

- Main wing: 22% reduction due to advanced materials
- Vertical Tail: 10% reduction due to advanced materials
- Horizontal Tail: 10% reduction due to advanced materials

Flight Control System: 10% reduction due to advanced systems

Landing Gear: 10 % reduction due to advanced materials

The final result of the Class II weight estimation is presented in Table 2.5.1a. The analysis resulted in a takeoff weight of 46,500 lbs. This is within 5% of the initial Class I takeoff weight so no performance iterations were made.

The propulsion weight estimation is also presented in Table 2.5.1a. The engine weight of 3,500 lbs was obtained from manufacturer's data. The engine weight listed in Table 2.5.1.a is 4,211 lbs. The 20% increase in weight is a result of converting the commercial version of the BMW BR710 engine to a military variant. The structural integrity of the engines were strengthened to withstand the increase in flight loads experienced by the military engines over the commercial engines. Methods outlined in reference 6 were used to calculate the following:

- Weight of Engine Controls
- Weight of Engine Starting Systems
- Weight of the Thrust Reverses

The results of the weight calculations for the propulsion system are listed in Table 2.5.1b.

Table 2.5.1a: *MIL-STD-1374A Weight Statement*

MIL-STD-1374 PART 1	GROUP WEIGHT STATEMENT, WEIGHT EMPTY				
Wing Group					W (lbs)
Main Wing					5,279.00
Tail Group					
Struct. -Stabilizer-incl. Elevator					1,262.00
Struct. -Fin-incl. Rudder					887.40
Body Group					
Struct. -Basic-Secondary					2,880.00
Alighting Gear Group, Tricycle					
Location					
Main					1,260.00
Nose					500.00
Engine Group					
Left					4,211.00
Right					4,211.00
Air Induction Group					
Nozzles					800.00
TOTAL STRUCTURE					21,290.40
Propulsion Group					
Propulsion System					577.00
Fuel System					752.00
Auxiliary Powerplant Group					
APU					300.00
Hydraulic & Pneumatic Group					320.00
Electrical Group					431.00
Avionics Group					
Equipment & Installation					762.00
Search Radar: Heracles II					350.00
Attack Radar: Hughes APG-65					550.00
ECM, IR, Communications					975.00
Furnishing & Equipment Group					
Furnishing & Accomodations & Emerg. Equipm.					578.00
Oxygen Equipment Group					134.00
Miscellaneous					220.00
Airconditioning Group					
A.C. & Pressure System & Anti-icing System					281.00
TOTAL WEIGHT EMPTY					27,520.40
	GROUP WEIGHT STATEMENT,				
	USEFUL LOAD AND GROSS WEIGHT				
Load Condition					
Crew (No. 4)					800.00
Trapped Fuel & Oil					211.00
Trapped Fuel & Oil					
Trapped Engines & Fuel Tanks					211.00
Fuel Tanks					
Location	Type				
Tank Group #1	Integral Tank				6,000.00
Tank Group #2	Integral Tank				6,000.00
Tank Group #3	Integral Tank				1,800.00
Tank Group #4	Integral Tank				1,800.00
Cargo					
Harpoon Missles (No. 2)					2,320.00
TOTAL USEFUL LOAD					18,931.00
WEIGHT EMPTY					27,520.40
TAKEOFF WEIGHT					46,451.40

Table 2.5.1b: *MIL-STD-1374A Propulsion Weight Statement*

MIL-STD 1374 Part II: Propulsion Group	Weight (lbs)
Engine Controls	175
Starting System	450
Thrust Reverses	1000
TOTAL PROPULSION GROUP	1625

2.5.2 Center of Gravity Locations

This sub-section contains the center of gravity information for the *Condor* attack jet. The weight-c.g. excursion diagram is shown in Figures 2.5.2. Center of gravity excursion studies were completed for each flight phase. The component center of gravity locations were estimated from the three view structural and system layouts. Since the fuel weight is such a large percentage of the total weight, the fuel was located as near to the center of gravity to reduce the c.g. travel due to fuel consumption. There are two tanks in the wing torque box and one tank in the fuselage as described in Section 3.2.9. An analysis of the effects of sweeping the wings fully while on the ground fully fueled showed that the *Condor* is very close to being a "tail sitter." The c.g. is at F. 591 and the main gear is at F. 604. In this case, the fuel should be loaded in the torque box tanks first, and the weaponry should be loaded next, before the main fuel group is loaded. Upon inspection it is apparent that the total operational travel of the center of gravity is quite small when compared to the chord length of the wing. The fuel is loaded first and was checked for the wings swept forward and aft condition. In both cases, the c.g. is very small. The results of the center of gravity studies follow:

Table 2.5.2 *Center of Gravity Locations*

	Weight	F.	Static Margin
Most aft c.g.	W_{TO} (wings swept)	591 in	15%
Most aft operational c.g.	W_{CR2}	567 in	-11%
Most forward c.g.	W_{OE}	552 in	-22%

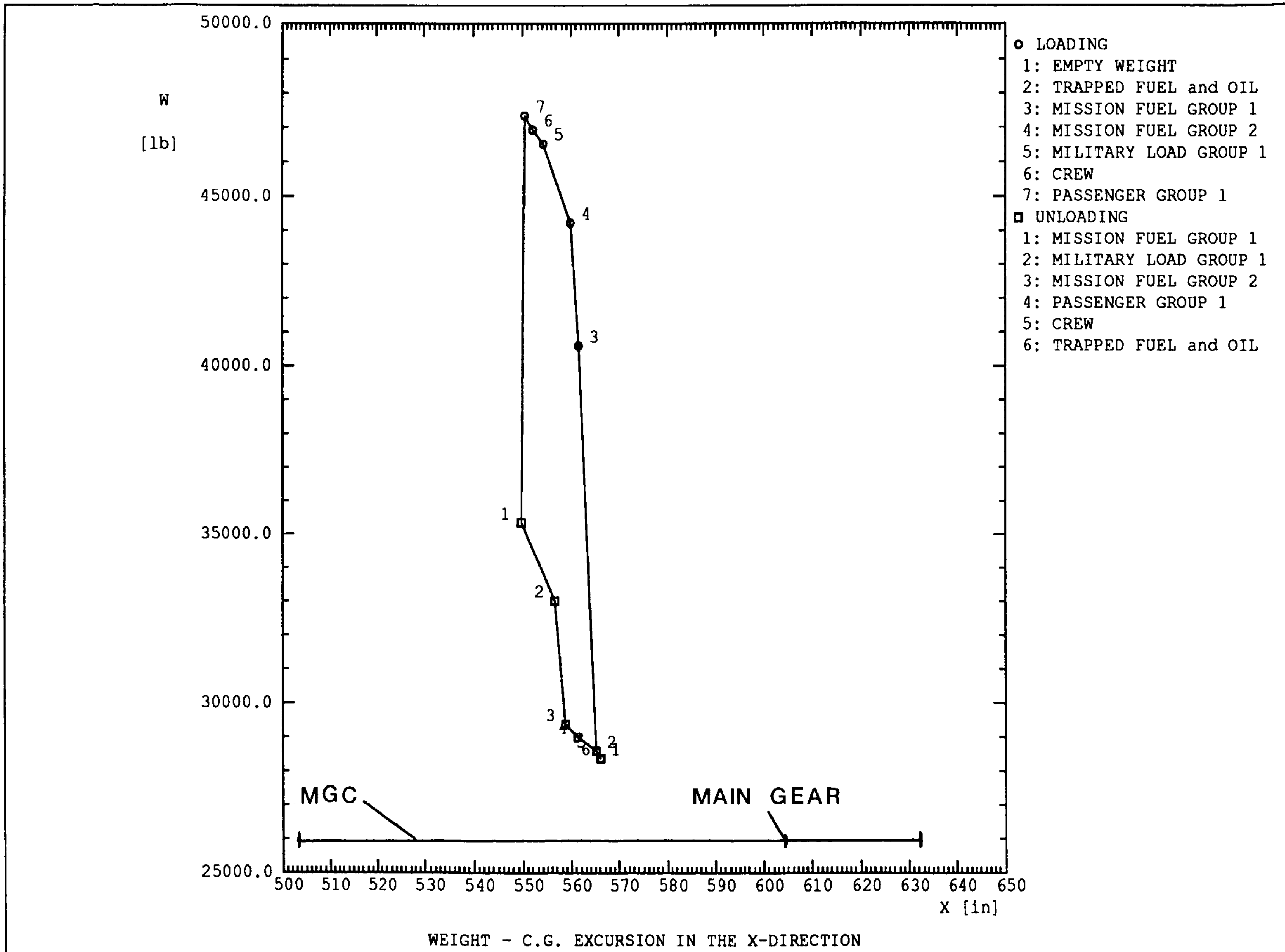


Figure 2.5.2: Center of Gravity Excursion in the X-Direction for the Condor

2.6 Performance

The purpose of this section is to present the performance analysis information of the Navy *Condor* attack jet. The following requirements were considered design drivers:

- Takeoff distance
- Cruise range
- Loiter endurance
- Dash capability
- Landing distance

These requirements fit into each of the flight phases of the *Condor*. The following sub-sections contain the details of the analysis for each flight phase.

2.6.1 Takeoff

The RFP calls for a maximum takeoff distance of 500 feet. This is due to the requirement for operating off a TARAWA Class ship where the available runway length is approximately 500 feet. Reference 2 contains methods for evaluating the takeoff performance of aircraft. The approach used here was to set the takeoff distance as a known value since it was specifically stated in the mission specification. However, a slightly more conservative number, 470 feet, was chosen to allow a safety margin, albeit small. Now, with the geometric and weight quantities known, the lift coefficient and thrust-weight ratio in takeoff can be easily obtained. Iterations in this process were made to find a takeoff distance that would yield better performance.

Because of the short field requirement, it was decided to use circulation control to augment the takeoff. This mechanism produces the capability for high lift coefficients at high angles of attack and very low stall speeds. It was decided to be conservative and assume that the total airplane lift coefficient was equal to the coefficient achieved through blowing over the flaps. In actuality the airplane coefficient will be augmented by the lift produced by the horizontal tail. However, this change will only serve to make the ground run more conservative. At this stage there is the assumption that the ship is stationary, no wind over the deck. The takeoff speed, 115% of stall speed, was selected to allow for acceleration of the airplane in the 470 foot ground run. The resulting takeoff conditions are:

- S_{TOG} = 470 feet
- C_L = 5.4
- W/S_{TO} = 66 psf
- T/W_{TO} = 0.61

The details of the circulation control, referred to hence forth as the blowing system, are addressed in Sub-section 2.4.2.

2.6.2 Cruise, Loiter, and Dash

From the Class II drag polar analysis detailed in Section 2.3.1.2, lift-drag ratios of 10.1 and 12.4 were found for the first and second cruise phases, respectively. These values translate well for the mission weight fractions which are determined to be 0.94 and .95 respectively (i.e.. the weight at the end of cruise is 94% of the weight at the beginning of the cruise phase). The cruise weight was calculated by averaging the weights at the beginning and the end of cruise which were found by using the weight fractions. This process was iterated for convergence to 5 percent of takeoff weight. The results of the cruise analysis are:

- $C_{Lcruise1}$ = 0.65
- $C_{Lcruise2}$ = 0.35
- $W/S_{cruise1}$ = 62 psf
- $W/S_{cruise2}$ = 48 psf
- $L/D_{cruise1}$ = 10.1
- $L/D_{cruise2}$ = 12.4
- $V_{cruise1}$ = 387 knots
- $V_{cruise2}$ = 415 knots
- h = 38,000 ft
- h = 38,000 ft
- W_5/W_4 = 0.94
- W_{10}/W_9 = 0.95

The loiter flight phase was considered mission critical due to the large amount of fuel burned and was, therefore, set to be the phase to maximize performance. A Class II drag polar analysis resulted in a lift-drag ratio of 15.3 in loiter. Again, using weight information obtained from the weight fractions, the speed for best L/D was calculated at the known lift coefficient. The results follow:

- E_{loiter} = 4 hrs
- $C_{Lloiter}$ = 0.66
- W/S_{loiter} = 56 psf
- L/D_{loiter} = 15.3
- V_{loiter} = 305 knots
- W_6/W_5 = 0.86
- h = 38,000 ft.

It is appropriate to mention here that the speed in loiter, while it is larger than given in the specifications, is the most efficient speed and the change in the specification was approved by Mr. Patrick Guhin of the AIAA.

The weight in dash was known, as was the speed, so a lift coefficient was obtained. The L/D was obtained, using Class II methods for drag prediction, for the configuration with the wings swept 60 degrees. The performance analysis showed that this flight phase is not critical, but it does have a mission weight fraction equal to that of the second cruise phase. The results are listed below.

- $R_{dash} = 100$ nm
- $C_{L dash} = 0.05$
- $W/S_{dash} = 47$ psf
- $L/D_{dash} = 3.1$
- $V_{dash} = 500$ knots
- $W_8/W_7 = 0.95$

It should be noted, however, that this phase played an important role in sizing the engine. This relationship will be addressed in Chapter 6.

2.6.3 Landing

The same requirement that specified the takeoff distance inherently applies the same restriction, 500 feet, to the landing ground run. It should be noted here that the landing analysis was performed assuming a static ship. In other words there is no accounting for wind over the deck or for the motion of the ship. As mentioned in Section 3.5, modifications could be made to the TARAWA Class ships to improve the takeoff and landing operation efficiency of this aircraft. The selected angled deck concept is a fairly inexpensive solution and offers great flexibility in utility. The design landing weight was set to be the critical emergency landing. In this scenario the aircraft experiences an emergency on takeoff climb and must land immediately. It is assumed that a fuel purge of 25% of the takeoff weight can be completed prior to landing. The results of this analysis are:

- $S_{LG} = 372$ feet
- $V_A = 82$ knots
- $C_L = 3.1$
- $W/S = 50$ psf

2.6.4 Summary

In this sub-section the results of the performance analysis for the *Condor* will be summarized. Table 2.6.3 contains a bulletized listing of the major results of this analysis. Figure 2.6.3 is a matching plot for the performance of this aircraft showing the relationship between each of the flight phases relative to the takeoff condition. The importance of the stall speed selection can be seen as the design point lies just to the left of the stall line. This indicates the criticality of achieving the appropriate lift coefficient in landing. It is also seen that the aircraft thrust-weight ratio required is driven by the takeoff requirement. It has been shown in this chapter that the performance requirements for each of the flight phases have been met, and in certain instances, exceeded. It can be concluded that the design at this point is a successful one from a performance standpoint. The complete results of the performance analysis and verification are tabulated below.

Table 2.6.3: *Performance Summary for the Condor Attack Jet*

Flight Condition	W (lbs)	C_L	W/S (psf)	S (ft)	V (kts)	W_{i+1}/W_i	L/D
Takeoff	46,500	5.4	66	470	62		
Cruise 1		0.4	62		387	0.94	10
Loiter		0.66	56		305	0.86	15.3
Dash ($S=823$ ft^2)		0.05	40		500	0.95	3.1
Cruise 2		0.35	48		415	0.94	12.4
Landing	35,000	3.1	50	270	82		

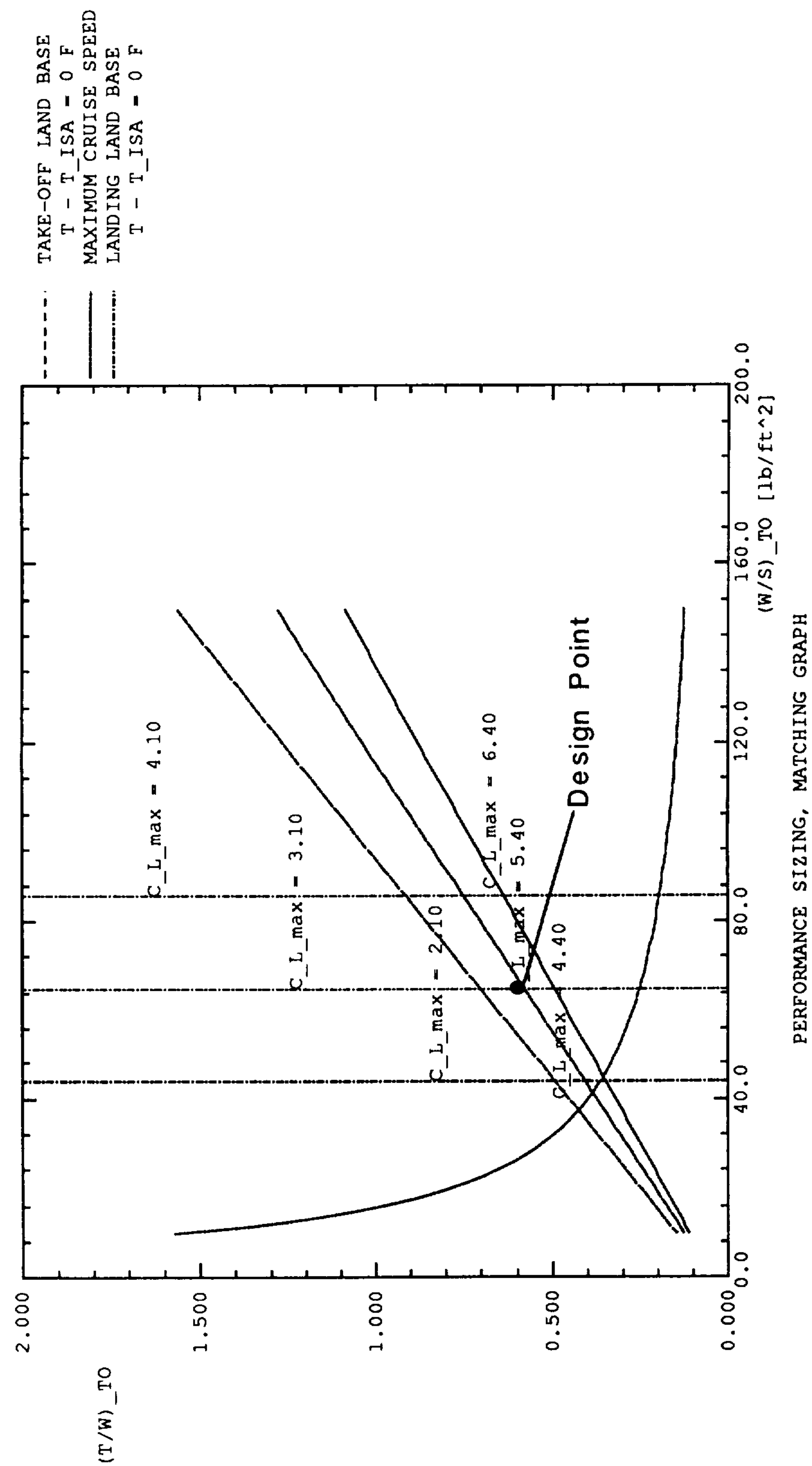


Figure 2.6.3: Matching Plot for the Condor

3. Technical Solutions to RFP

The detail design of the *Condor* includes development of a three-dimensional model, systems design and structural design to meet the requirements of the RFP. The overall configuration of the aircraft can be represented in a three-view and an inboard profile displayed in Section 3.1. The systems are described in varying detail in Section 3.2. The three dimensional model of the aircraft and the systems inside are supported by a structural layout described in Section 3.3. In today's market, not only must every component of the aircraft be successfully incorporated within the aerodynamic shell of the aircraft, but the overall configuration must also be designed to provided adequate access to necessary high maintenance areas and systems. The accessibility of the *Condor* is described in Section 3.4. In some cases, research or trade studies conducted during the development of the design of the *Condor* lead the design team to explore possible exceptions to the RFP if certain requirements were found to be unclear or unjustifiable to our design. The exceptions to the RFP as they apply to the *Condor* can be found in Section 3.5.

3.1 Configuration

The proposal submitted by this design team in response to the 1993/1994 RFP consists of a four seat, variable swept wing, twin turbofan aircraft with STOL capabilities. The design is capable of carrying air-to-surface or air-to-air weapon systems along with elaborate system of attack and surveillance radar and IFR systems. The Mission Specification listed in the RFP dictates that the aircraft must be able to cruise a total of 800 nautical miles, loiter for up to 4 hours, and dash at intercept speed 100 miles (500 kts).

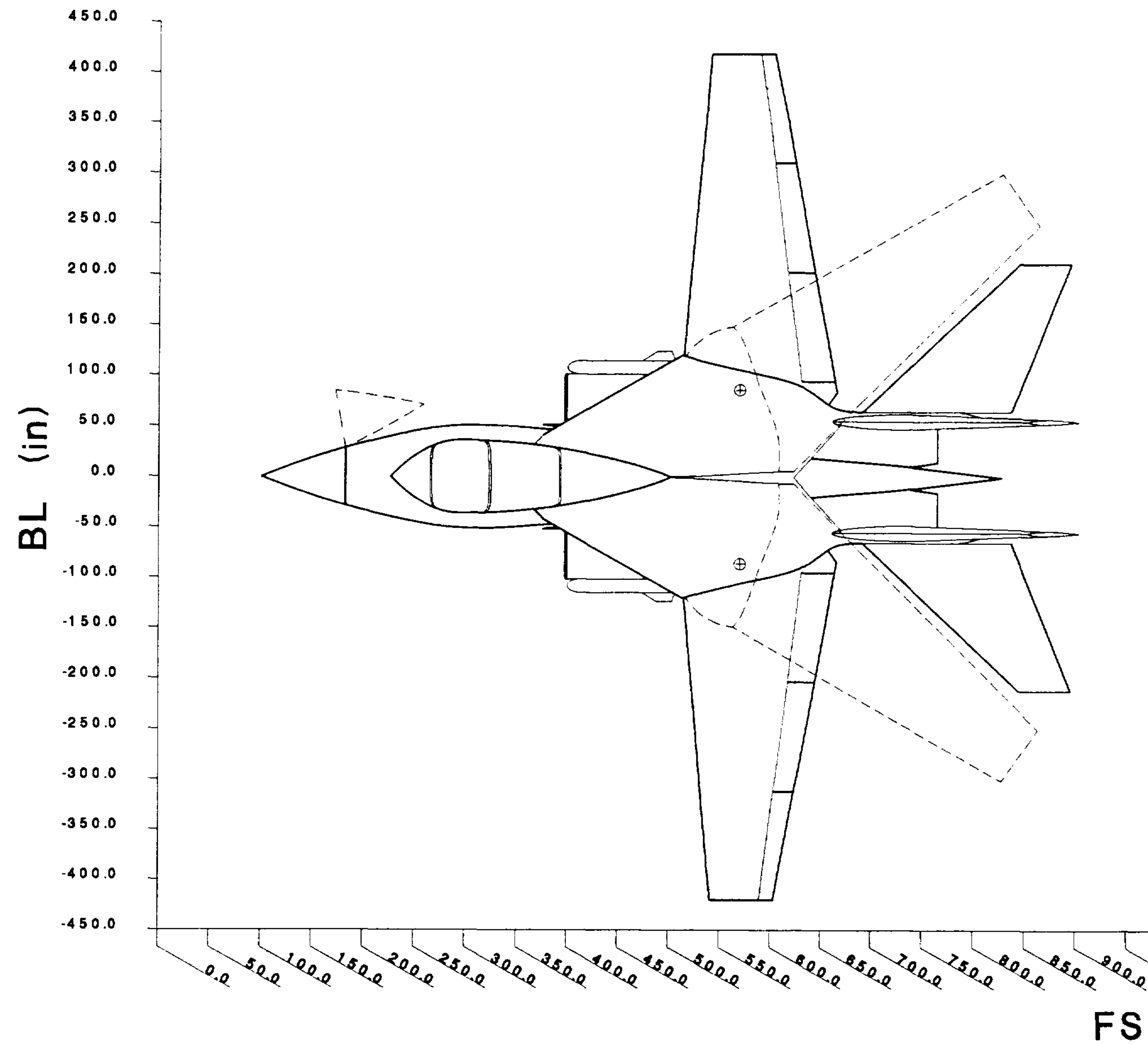
A variable swept wing is chosen by the design team to meet the loiter and dash requirements specifically listed in the RFP. The configuration consists of a high wing with a reference area of 700 square feet. For high lift-to-drag ratios in loiter and improved low speed performance at takeoff and landing, the wings have a leading edge sweep of 5 degrees. To meet the 500 kt at sea level dash requirement, the wings are swept aft to 60 degrees. The aft swept wings are chosen to improve the ride quality of the airplane at high dynamic pressure.

To meet the operating requirements for the TARAWA class ships, the aircraft must be able to takeoff and land within 500 ft. Conventional methods of high lift systems would require a high thrust-to-weight ratio for takeoff. The design team concluded that the large engines needed for takeoff would be inefficient at loiter as compared to smaller engines operating close to their peak specific fuel consumption levels. Therefore, to incorporate smaller engines into the design, an active high lift system is adopted. The high lift system consists of a single slotted fowler jet flap. Bleed air is ducted from the two APU's and is used to accelerate the flow passing over the flaps.

The empennage consists of twin vertical tails and a variable incidence horizontal stabilizer. Two vertical stabilizers are chosen to meet directional stability and height requirements. A single vertical tail required for directional stability would be too large for the underdeck storage on the TARAWA class ships. The horizontal stabilizer is placed behind and under the wing to "catch" the downwash and dynamic pressure created by the jet blown flaps. This placement aids in trimming the large pitching moment produced by the blown flaps.

The physical attributes of the *Condor* are displayed in the form of a three-view and an inboard profile in Figures 3.1.1 and 3.1.2 respectively. A table of the geometric parameters for the aircraft can be found in Figure 3.1.1.

FOLDOUT FRAME 1



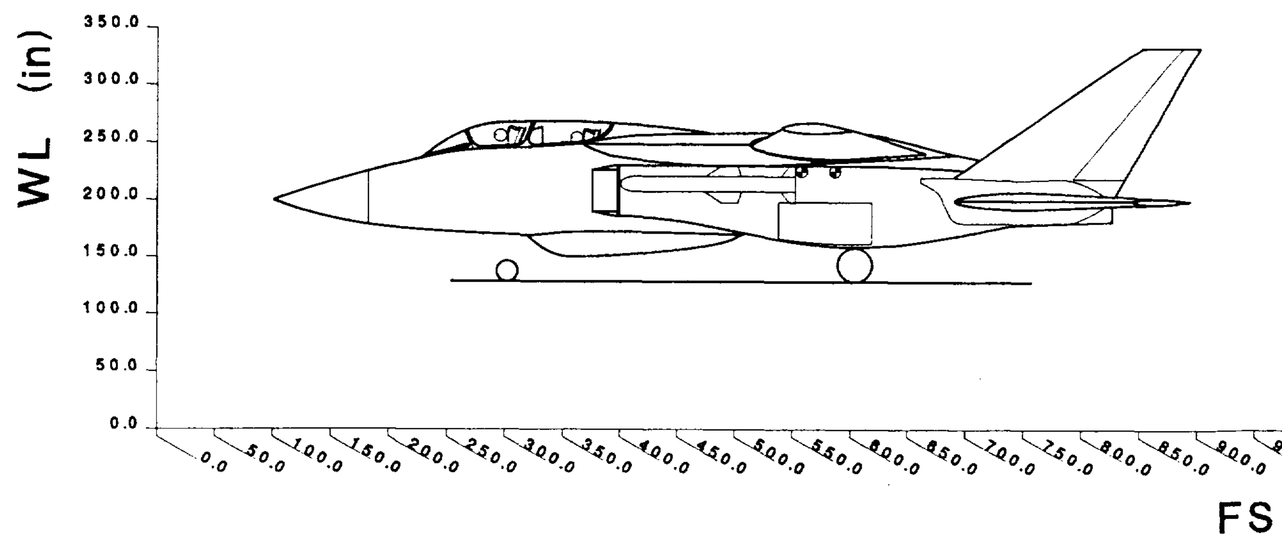
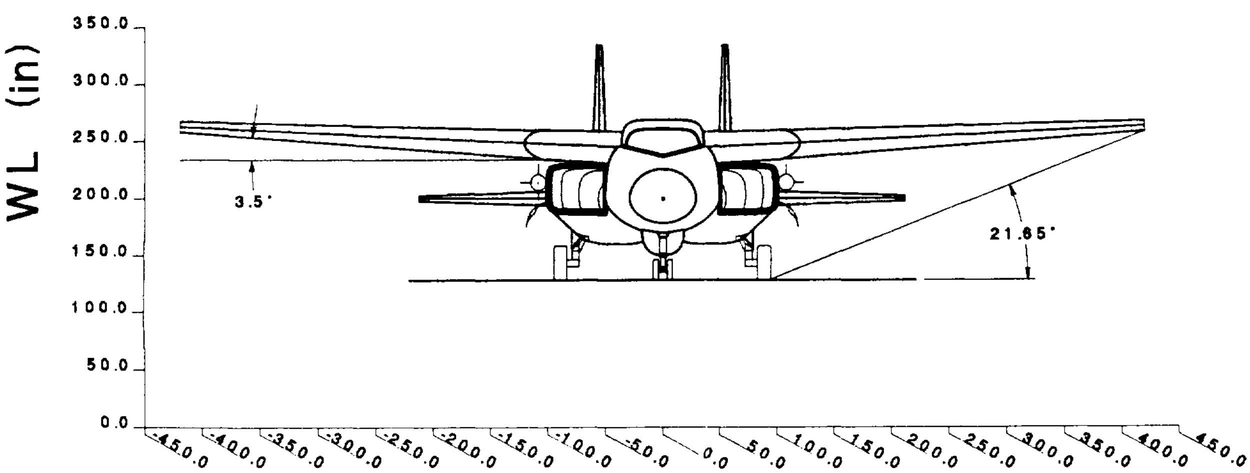


Figure 3.1.1: Three-view c

Condor: Geometric Characteristics	Wing	Horizontal Tail	Vertical Tail
Area	700 ft²	240 ft²	200 ft²
Span	70 ft	27.7 ft (from the fuselage side)	11.18 ft
MGC	10.8 ft	9.42 ft	9.81 ft
Aspect Ratio	7	3.19	1.25
Sweep Angle	5° - 60° (L.E.)	46° (L.E.)	46° (L.E.)
Taper Ratio	.35	0.32	0.30
Thickness Ratio	0.15	0.10	0.10
Airfoil	NACA 65A415	NACA 65A010	NACA 65A010
Dihedral Angle	3.5°	0.0°	0.0°
Incidence Angle	2.0°	Variable	0.0°
Aileron Chord Ratio	0.30		Rudder: 0.30
Aileron Span Ratio	0.23 - 0.99		Rudder: 0.18 - 0.99
Flap Chord Ratio	0.30		
Flap Span Ratio	0.23 - 0.99		

	Fuselage	Cabin Interior	Overall
Length	59.6 ft	12.5 ft	66.7 ft
Maximum Height	9.5 ft	5 ft	17.5 ft
Maximum Width	16.6 ft	5.8 ft	70 ft

(in)



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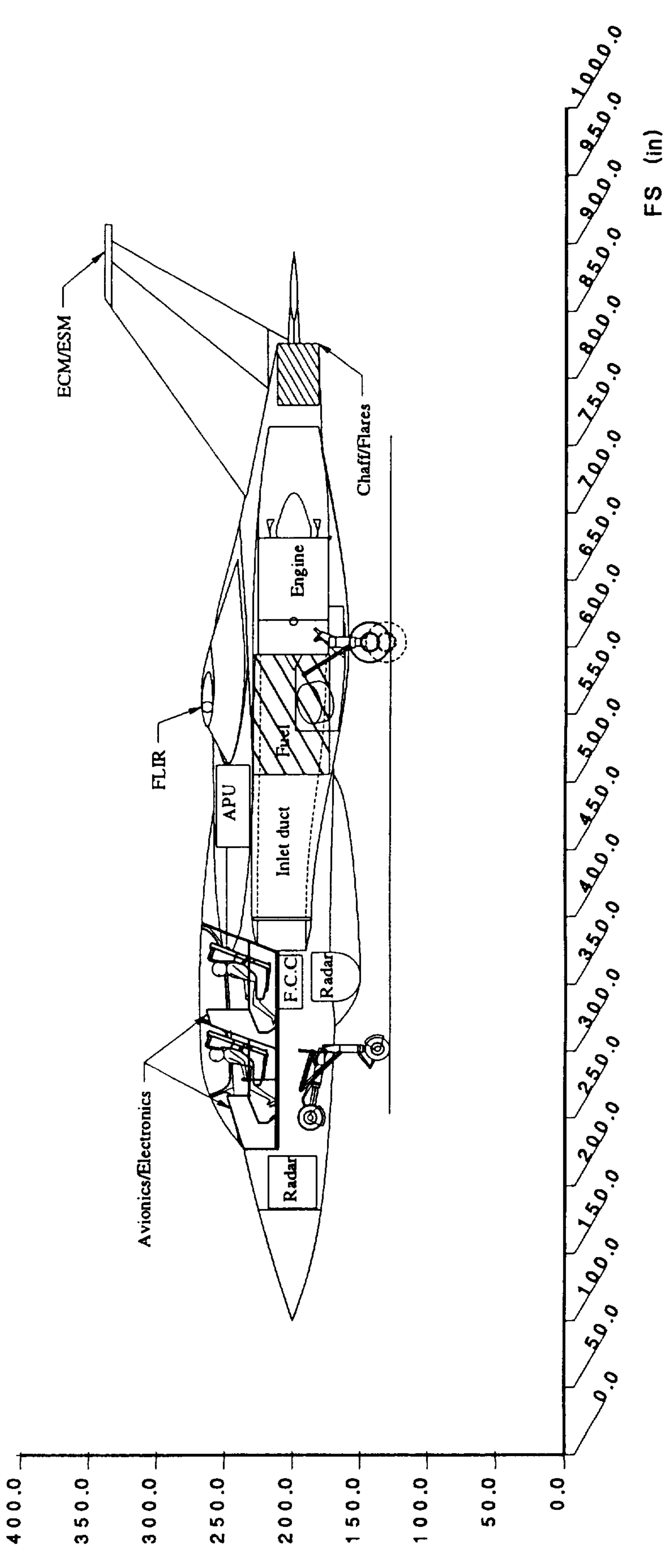


Figure 3.1.2: Inboard Profile of the MPS-2000 Condor

3.2 Systems

Once the outside lines of the aircraft are established, internal and external systems must be incorporated into the design. Conflicts between systems and the configuration must be overcome through an iterative process that may entail reconfiguration of the aircraft's outside lines. The final system designs for the *Condor* are presented in the following sub-sections.

3.2.1. Landing Gear Layout

The landing gear of the *Condor* is a retractable, tricycle configuration. It is designed to the US Navy carrier based aircraft specifications with a vertical touchdown speed of 22 fps. and a landing speed of 82 kts. Retracted wheel volume, tire size, wheel retraction kinematics, longitudinal and lateral tip-over clearance, Foreign Object Damage (FOD) and shock strut size are a few parameters driving the landing gear layout.

To support the *Condor* during ground maneuvers and landing, the assumption is made that the nose gear and main gear will support a maximum of 10% and 98% of the takeoff weight through the range of c.g. locations. The loads on the struts and tires are calculated by using methods in Reference 5. From the calculation, it is determined that the *Condor* would require a layout of 2 noses tire and 2 main tires.

Forward retraction into the fuselage is selected for nose and main gear of the *Condor*. For the main gear, retraction is possible through a tilted pivot as shown in Fig 3.2.1.1. The final position of retraction of the nose gear and main gear are also shown in Fig. 3.2.1.1.

All of the shock-strut loads were multiplied by a load factor of 1.25 to account for future growth of the *Condor*. The maximum static loads for the nose and main struts are as follows:

- Nose Gear: 1 strut at 9,080 lbs
- Main Gear: 1 strut at 14,000 lbs each

The shock-struts for the nose and main gear are of the Bendix Oleo-Pneumatic dual chamber type. The shock-strut dimensions are shown in Table 3.2.1.1.

Table 3.2.1.1 *Landing Gear Shock-Strut Dimensions*

	Strut Length (in.)	Shock Absorber Diameter(in.)
Nose Gear	32	4
Main Gear	29	6

Designing for the highest strut loads for the c.g. range and considering the size and pressure of tires available, the tire selections and layout were arrived at following Table 3.2.1.2:

Table 3.2.1.2 *The Tire Sections and Layout*

	Nose Gear	Main Gear
Size (dia. x width) (in)	18 x 4.4	30 x 11.5
Maximum Loading per tire (lbs)	4,350	30,000
Rim Size (in)	10	14.5
Manufacture	Goodrich	Goodrich

The *Condor* is designed with the main gear well behind the aft c.g. and with a large wheel base to satisfy the minimum longitudinal and maximum lateral tip-over criteria for US Navy of 15 and 54 degrees respectively. The *Condor* has longitudinal and lateral tip-over angles of 25 and 53 degrees respectively as shown in Figure 3.2.1.2a and 3.2.1.2.b respectively.

The search radar for the *Condor* is placed on the bottom of the fuselage. To prevent an FOD problem with the radar, a splash guide is attached behind the nose wheel shown in Fig 3.2.1.1.

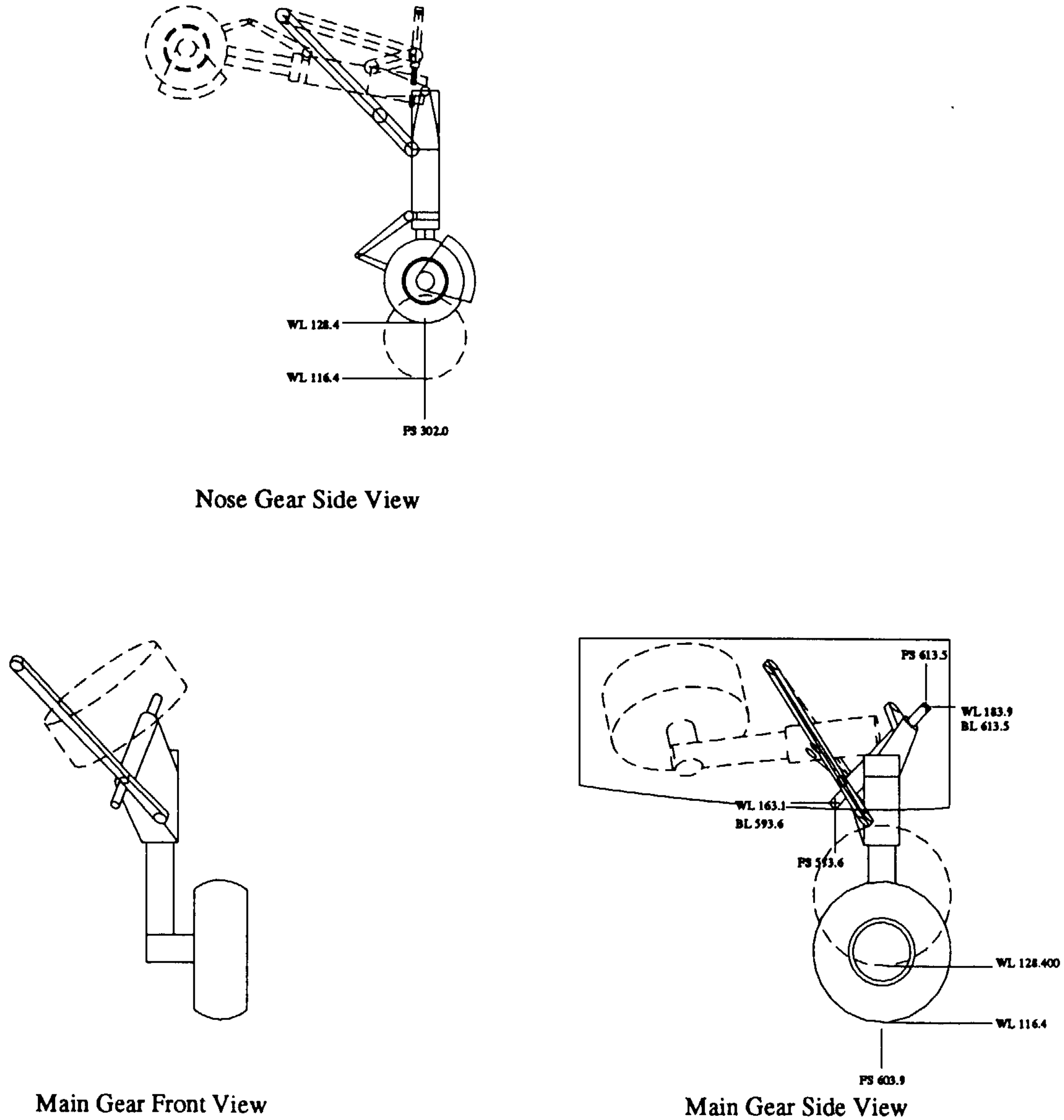


Figure 3.2.1.1: Landing Gear Kinematics for the Condor

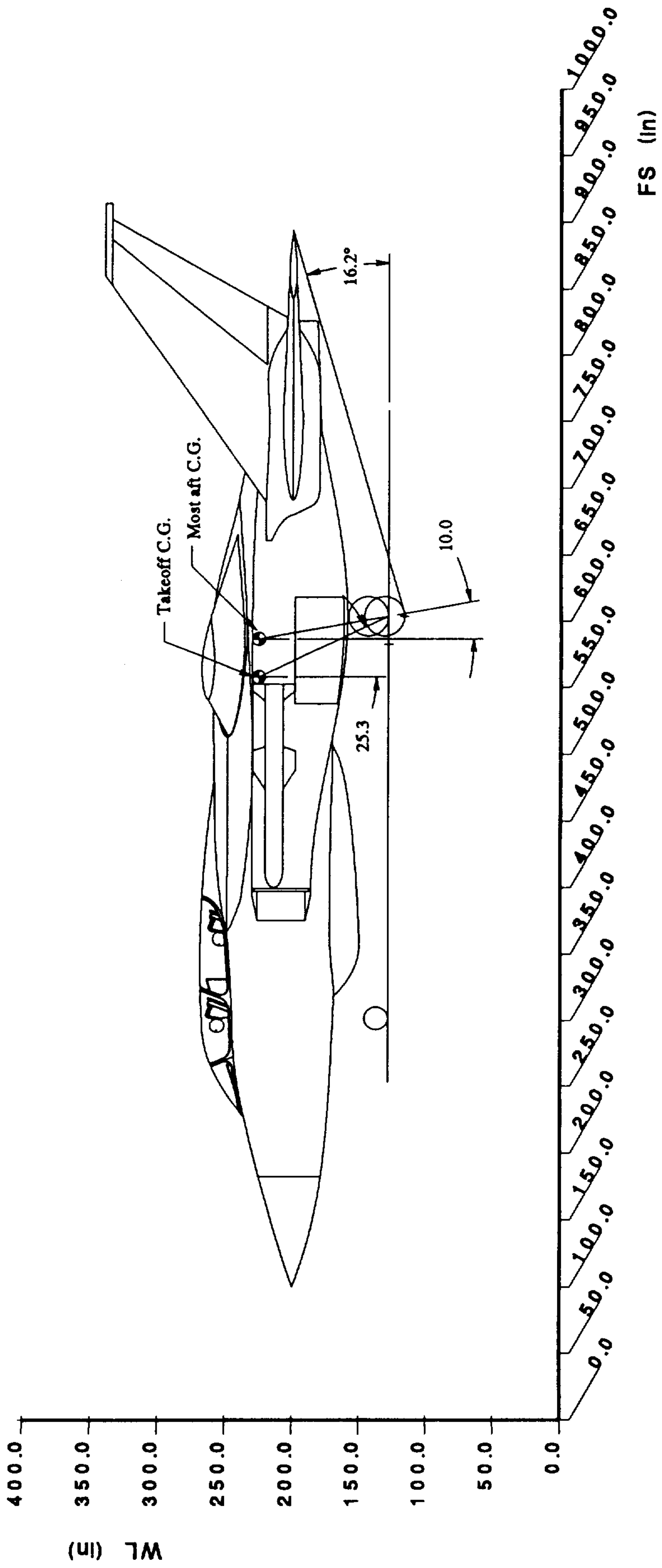


Figure 3.2.1.2.a: Longitudinal Tipover Angles

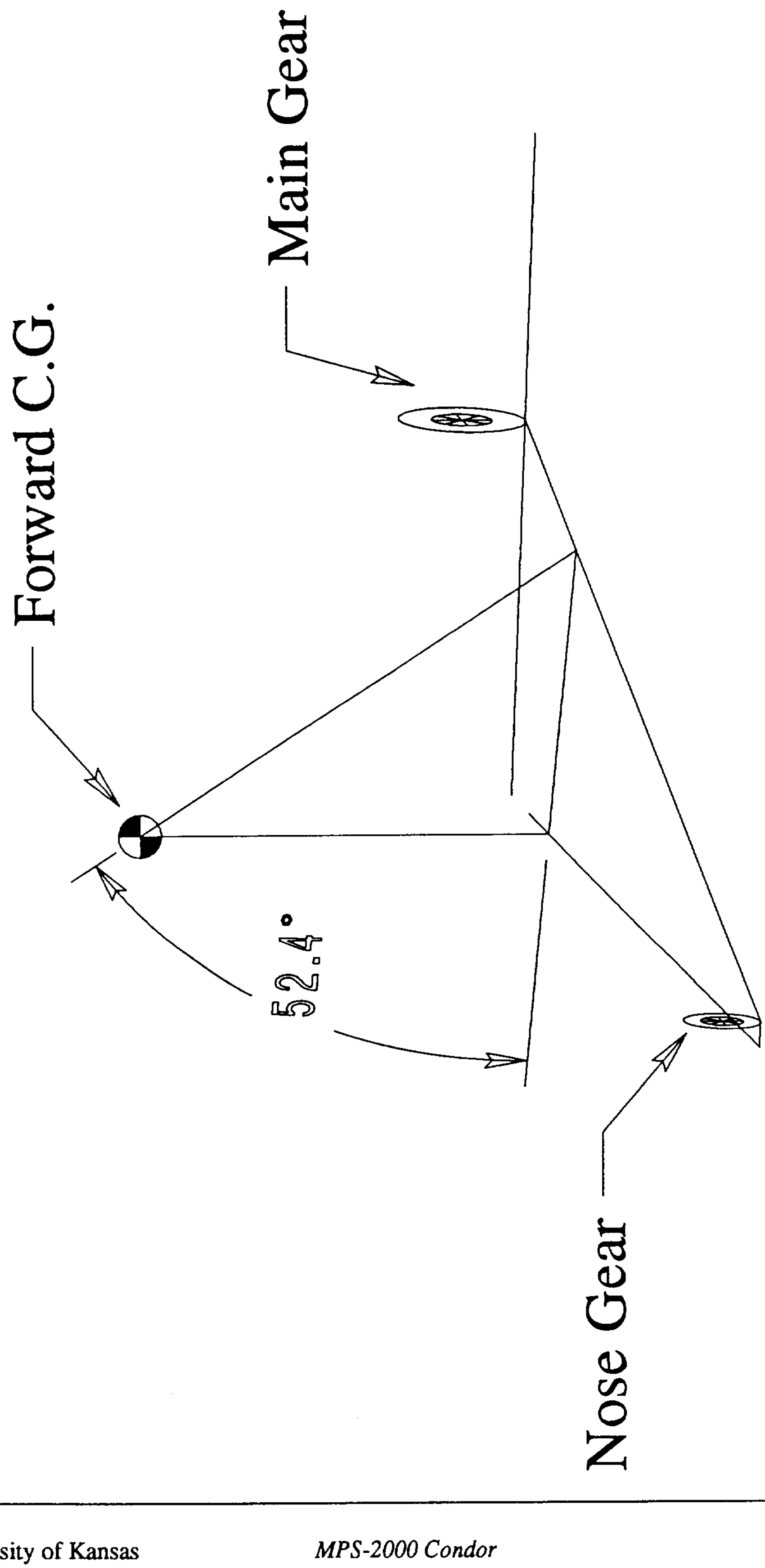


Figure 3.2.1.2.b: Lateral Tipover Angle

3.2.2. Cockpit

The cockpit for the *Condor* is designed for four crew members: pilot, copilot, navigator and radar operating officer. The following three considerations are presented and discussed:

- General arrangement of the cockpit
- Cockpit instrumentation
- Ejection seat

The three view of the cockpit in Fig. 3.2.2.1 shows the two-by-two layout. Two standard male pilots (179 lbs and 69 in) are shown in the figure. The pilot and copilot are seated in the front left and front right seats, and the radar operating officer and navigator are seated in the rear left and rear right seats. Since the rear occupants do not have a forward visibility requirement, the rear crew members seats are located directly behind the front seats. This reduces the frontal area of the canopy. The pilot and copilot both have a standard 15° below-the-horizon visibility. The flight controls are assumed to be fly-by-wire with stick controls. The rudder pedals travels 3.25 inches, and can be adjusted 8 inches.

The cockpit instrumentation of front and rear consoles can be seen in Fig 3.2.2.2. The console is designed for simplicity by using multifunction displays. The description of each multifunction display is shown in Fig (3.C). Note the large rear console: it is 36 inches high and runs the entire width of the cockpit. This is to accommodate the wide array of electronics equipment (at least 1.5 inches space between the equipment) which may be installed into the cockpit. A sliding key pad desk was also installed for the navigator's convenience. This sliding desk will be automatically retracted into the console when ejection is required. The weapon control panel is installed at the left side of the front cockpit for the pilot and at the right side of the front cockpit for the copilot. Two radar control displays are installed at the rear-left side.

Each crew member sits in a standard zero-zero ejection seat. Each seat has an ejection clearance of 34 in. (longitudinal) x 31 in. (lateral). The sequence of ejecting the crew members from the cockpit is: navigator, radar operating officer, copilot and pilot.

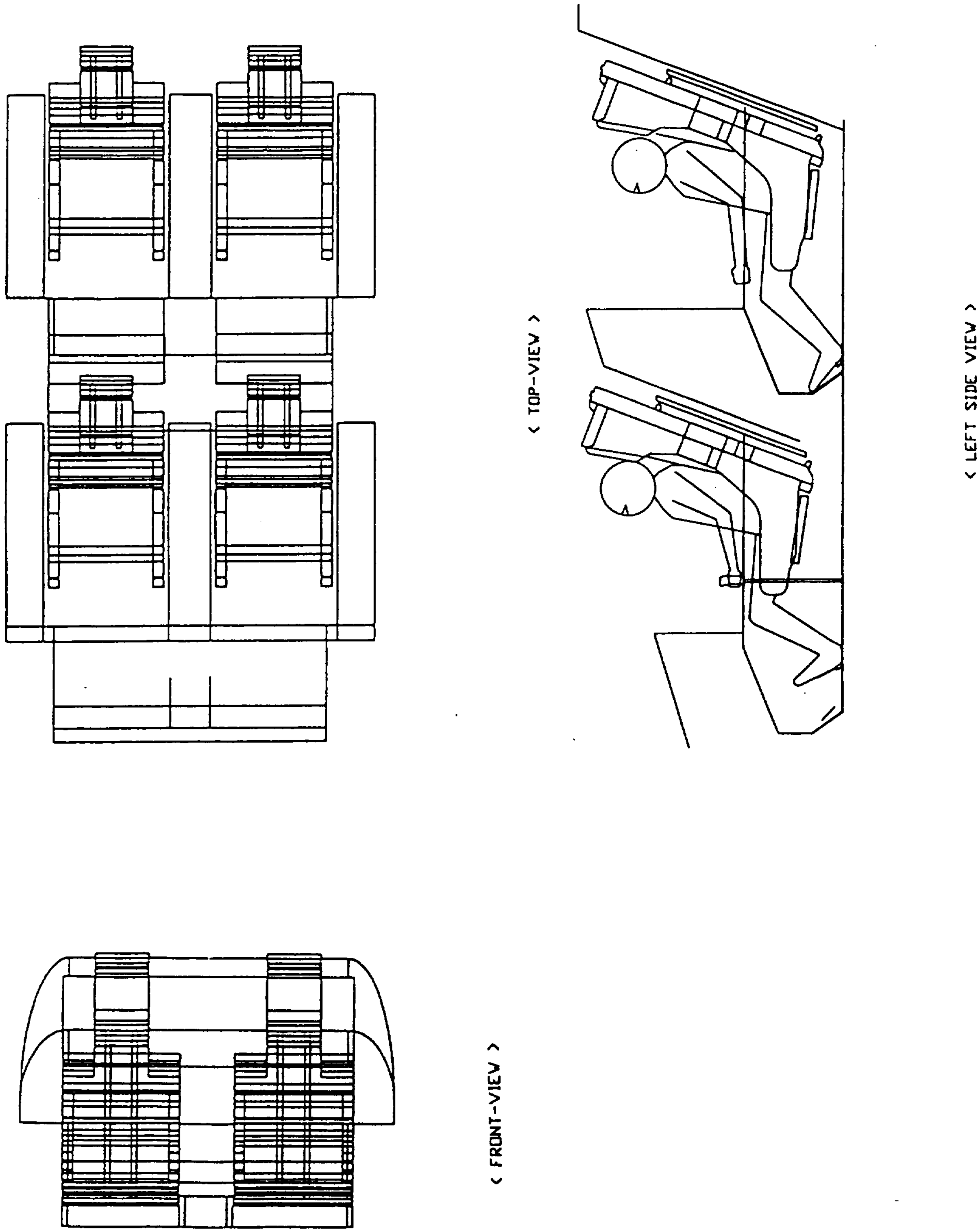
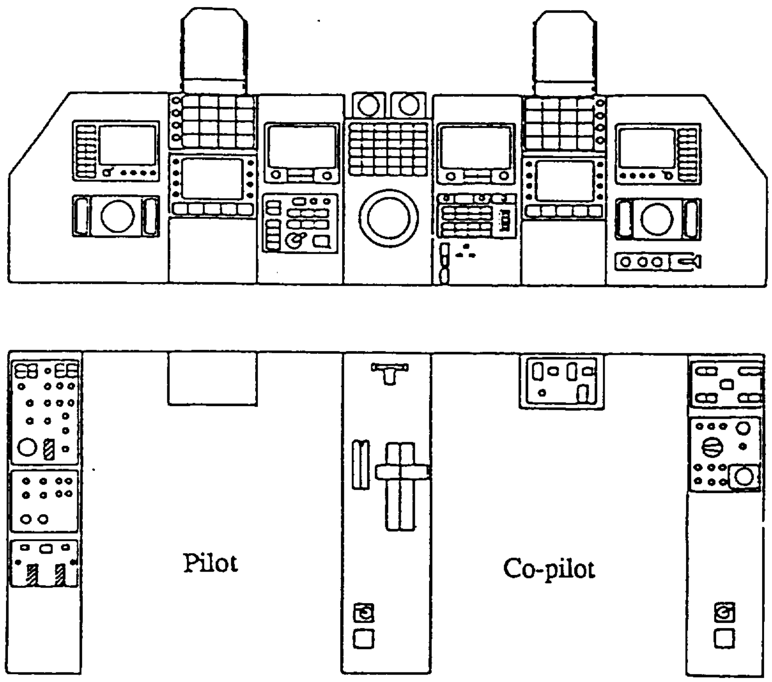
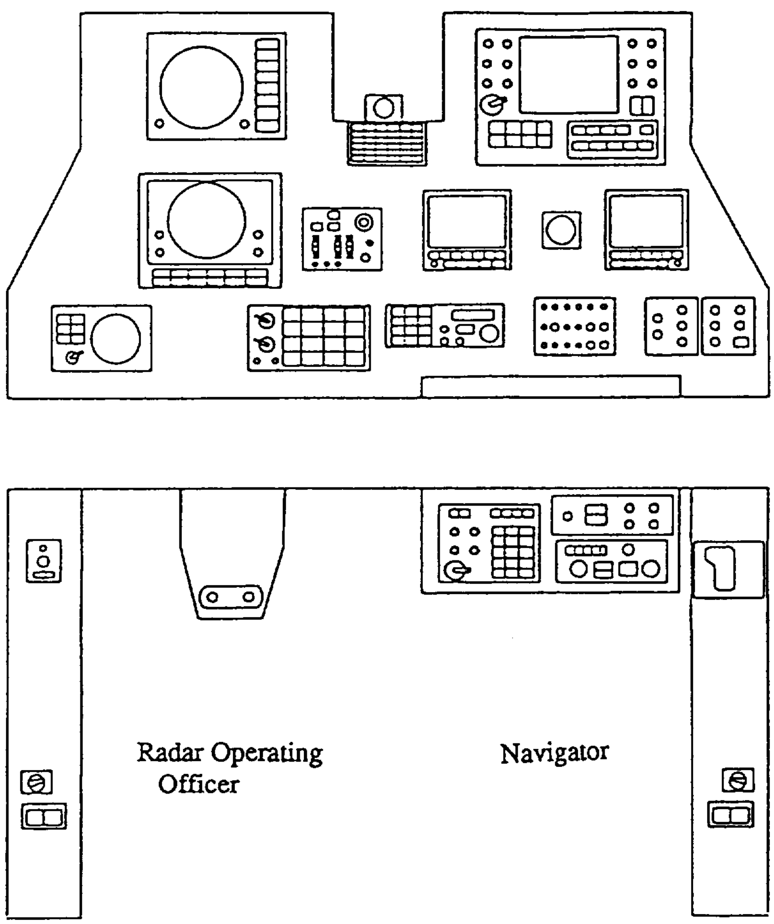


Figure 3.2.2.1: Thre-View of the Cockpit of the Condor

FOLDOUT FRAME /

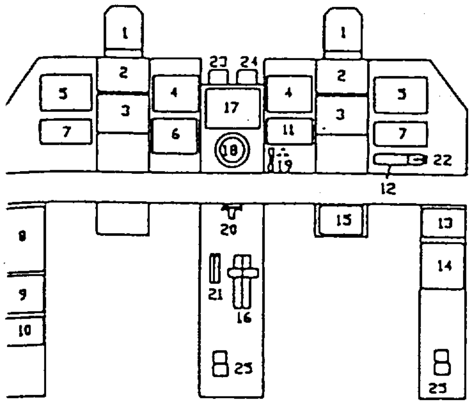


Front Console

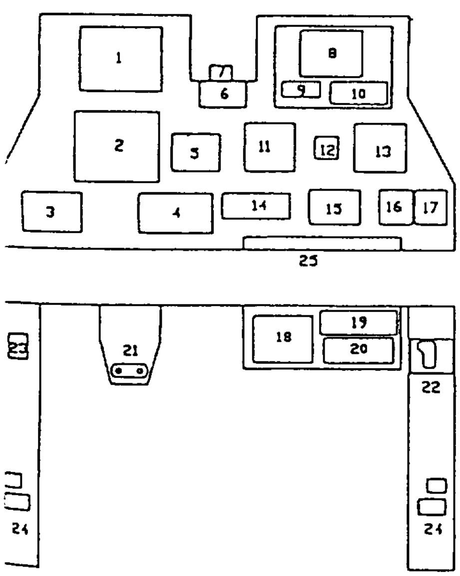


Rear Console

Figure 3.2.2.2: Cockpit Instrumer



1. HEAD UP DISPLAY
2. HEAD UP DISPLAY CONTROL PANEL
3. MULTIFUNCTION DISPLAY 1 (Repeater Projected Map
4. MULTIFUNCTION DISPLAY 2 (E-Scope Radar Repeater)
5. MULTIFUNCTION DISPLAY 3 (Weapon Control)
6. AUTOPILOT AND FLIGHT DIRECTOR CONTROL PANEL
7. ALTITUDE DIRECTOR INDICATOR
8. FUEL CONTROL PANEL
9. ENGINE CONTROL PANEL
10. AIR INTAKE RAMPS CONTROL PANEL
11. COMMAND AND STABILITY AUGMENTATION SYSTEM CONTROL PANEL
12. HYDRAULIC CONTROL PANEL
13. WEAPON CONTROL PANEL
14. ENVIRONMENT CONTROL PANEL
15. BOMB RELEASE SAFETY LOCK CONTROL PANEL
16. THROTTLES
17. WARNING PANEL
18. REPEATER PROJECTED MAP DISPLAY
19. LANDING GEAR OVERRIDE LEVER
20. LANDING GEAR EMERGENCY RELEASE LEVER
21. WING SWEEP LEVER
22. EMERGENCY POWER SUPPLY
23. STANDBY COMPASS
24. O'CLOCK
25. OXYGEN CONNECTOR AND CONTROL PANEL



1. ATTACK RADAR MULTIFUNCTION DISPLAY
2. SEARCH RADAR MULTIFUNCTION DISPLAY
3. ADVANCED SELF PROTECTION JAMMER CONTROL PANEL
4. AVIATOR NIGHT VISION SYSTEM CONTROL PANEL
5. MAPPING RADAR CONTROL PANEL
6. CENTRAL WARNING PANEL
7. O'CLOCK
8. MULTIFUNCTION DISPLAY
9. RADAR AND PROJECTED MAP DISPLAY CONTROL PANEL
10. NAVIGATOR MODE CONTROL PANEL
11. LEFT TV/TAB DISPLAY
12. ARTIFICIAL HORIZON
13. RIGHT TV/TAB DISPLAY
14. V/UHF CONTROL PANEL
15. COMMUNICATION CONTROL SYSTEM CONTROL PANEL
16. INTERNAL LIGHTS PANEL
17. EXTERNAL LIGHTS PANEL
18. INERTIAL NAVIGATOR CONTROL PANEL
19. MAIN COMPUTER CONTROL PANEL
20. SECONDARY ALTITUDE AND HEADING REFERENCE CONTROL PANEL
21. NAVIGATOR NIGHT VISION SYSTEM
22. NAVIGATOR'S HANDCONTROLLER
23. COCKPIT VOICE RECORDED CONTROL PANEL
24. OXYGEN CONNECTOR AND CONTROL PANEL
25. SLIDING DESK

tation of Front and Rear Consoles

3.2.4 Engine Integration

For the propulsion system for the *Condor* consists of two BMW-12-12 BR 710 engines. The engine selection is based on the new technology of the powerplant the thrust requirements of the *Condor* and its relatively low specific fuel consumption. The engine is a turbofan with a bypass ratio of 4 and a takeoff thrust of 14,700 lbs. For military application, the BMW-12-12 BR 710 was strengthened structurally. As mentioned in Section 2.5, the military engine weight differs from the commercial engine weight by 20% based on an assumption recommended by Dr. Roskam from the University of Kansas.

The engines are placed side-by-side in the aft section of the fuselage between the tail booms. Engine access is achieved through ventral panels as described in Section 3.4. The inlets for the engines are sized with a cross sectional area of 12.25 sq. ft. The inlet sizing is a result of mass flow requirements studied in each flight condition. The loiter condition was found to be the critical condition for sizing the inlets.

3.2.4 Flap Blowing System

The ability of the *Condor* to operate from a TARAWA class ship is due to the high lift capability of flap blowing system described aerodynamically in Section 2.4.2. Physically, the air necessary for the blowing of the flaps is provided by two APU's. In the case of an APU failure, the other APU is capable of providing the mass flow required for the *Condor* to complete a successful takeoff. The air is ducted from the APU's, through the pivot mechanism to the leading edge of the wing-root section and then to the trailing edge duct (See Figure 3.2.4.1). The full span slot in the duct is 0.2 inches thick through which the air is accelerated to 472 ft/s. As can be seen from Figure 3.2.4.2, as the air exits the slot. it passes over the flaps.

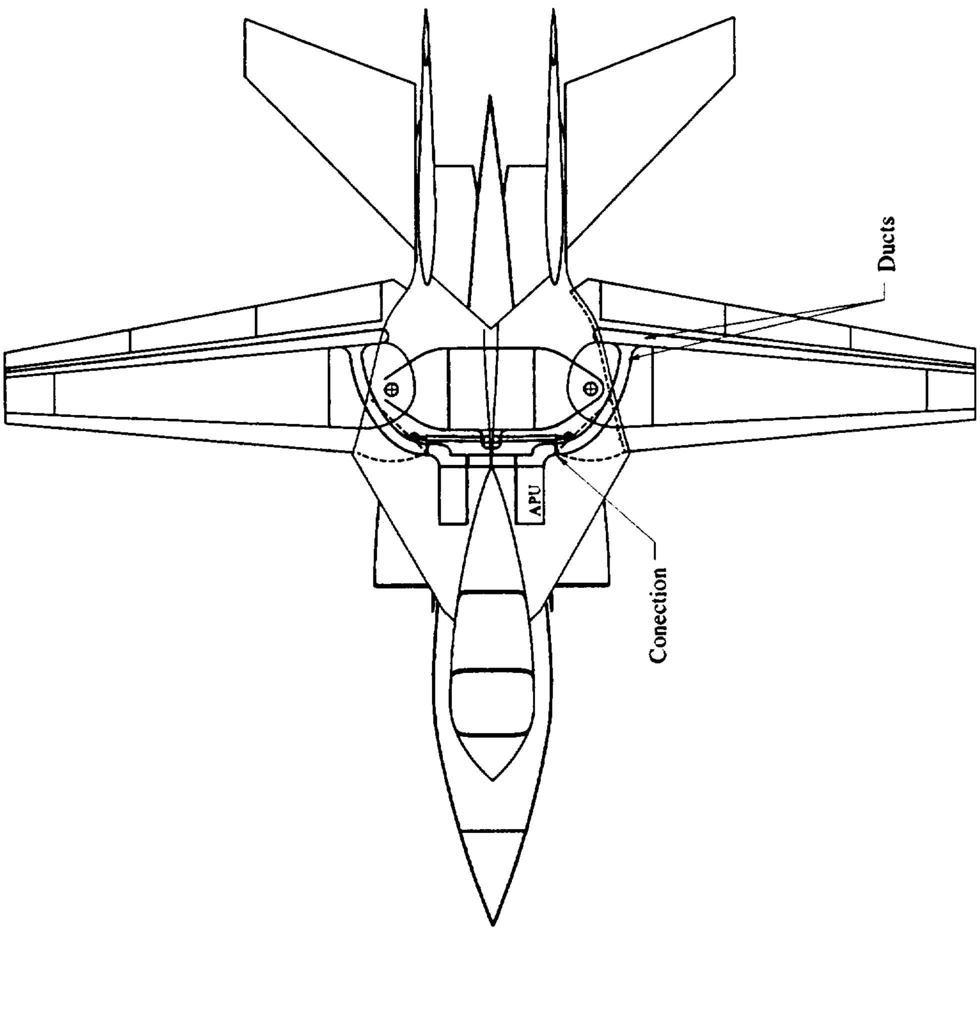


Figure 3.2.4.1 : Ducting for Flap Blowing System

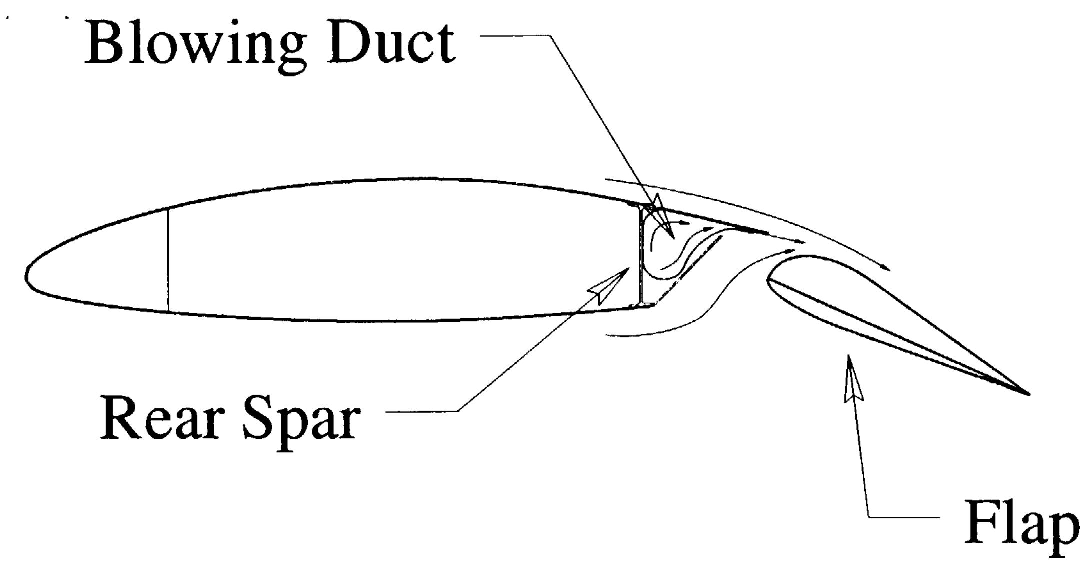


Figure 3.2.4.2: Flap Blowing Mechanism

3.2.5 Flight Control System

The design of the *Condor* includes the use of an irreversible flight control system. The system can be categorized into a primary and a secondary systems. The primary flight control system for the *Condor* includes:

- Longitudinal Control: Stabilizer
- Lateral Control: Ailerons, Differential Stabilizer
- Directional Control: Rudder

Each primary control surface is moved with double redundancy hydraulic actuators signaled with fly-by-wire with a mechanical back-up on the longitudinal controls. The following controls are considered members of the secondary flight control system:

- **Trim Controls:** Longitudinal, Lateral, Directional
- **Thrust:** Engine Fuel Control
- **High Lift:** Flap, Circulation Control

The secondary controls are singular redundancy. The primary and secondary flight control systems for the *Condor* are displayed in Figures 3.2.5.1.

In an attempt to reduce production and maintenance costs for the *Condor*, the flight control surfaces are split into smaller, separate surfaces. For example, the flaperons are split into three separate surfaces instead of one large moving control surface. This configuration selection not only allows wing elasticity, but it also allows the surfaces to be sized so that the same actuators can be used as those found on the rudders. In this fashion, large numbers of these actuators can be produced which reduces the unit cost as suggested by Reference 5.

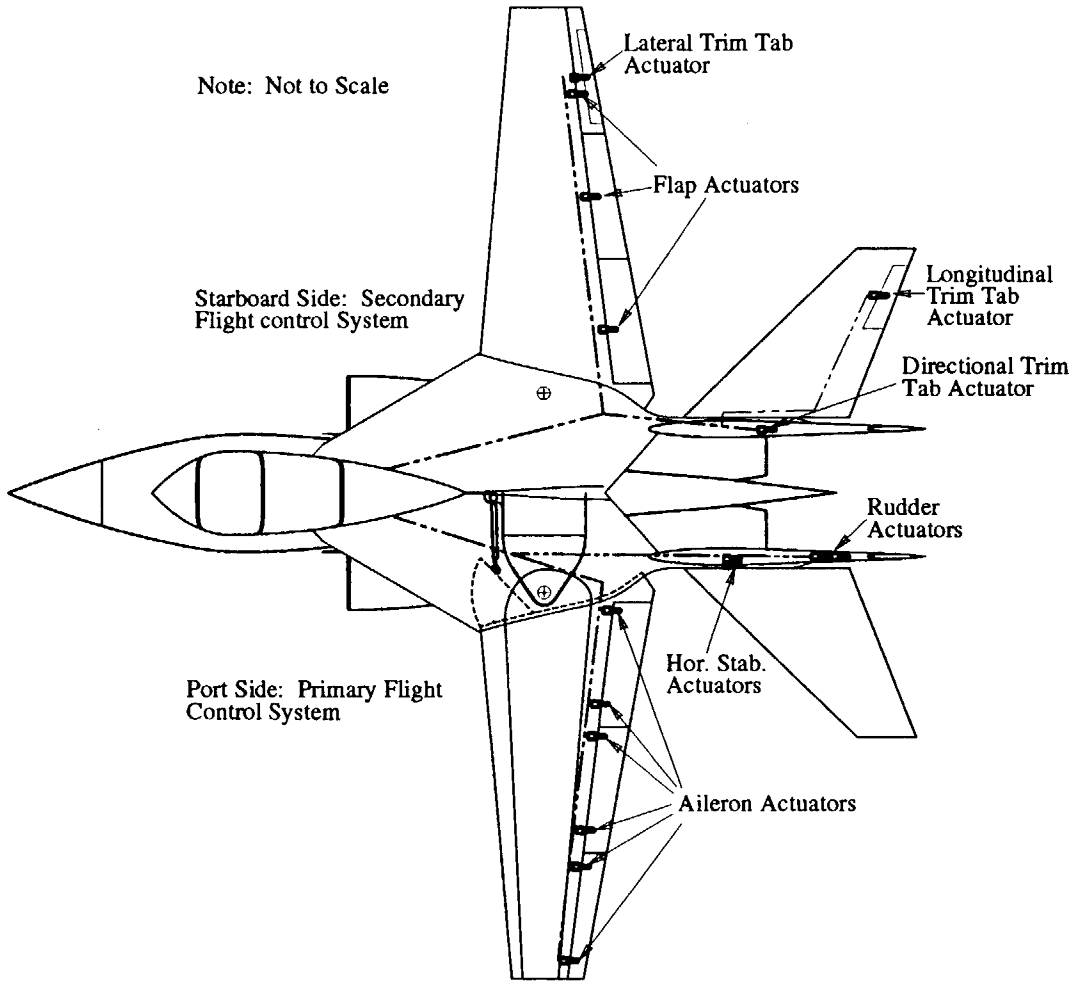


Figure 3.2.5.1: Primary and Secondary Flight Control System for the Condor

3.2.6 Hydraulic System

Power for the various systems described in this chapter is primarily supplied by the hydraulic system. This system consists of a series of pumps, reservoirs and valves to supply and deliver the needed power. The following systems are dependent upon the Hydraulic system:

- Primary Flight Control (See Section 3.2.5)
- Secondary Flight Control (See Section 3.2.5)
- Landing Gear Mechanisms (See Section 3.2.1)
- Variable Wing Sweep (See Section 3.2.8)

3.2.7 Electrical System

The electrical system for the *Condor* supplies additional power to those systems described in Section 3.2.6 and other systems independent of hydraulic power. The systems on the *Condor* requiring electrical power are:

- Internal and External Lighting
- Flight instruments and Avionics
- Engine Starting
- Primary and Secondary Flight control Systems

Primary electric power is produced on the *Condor* with the use of engine driven generators. These DC generators can also be reversed and used as starter motors. The secondary electrical power systems includes a battery system and two Auxiliary Power Units (APU).

The electrical power production system must be sized to carry the electrical loads of all the systems required for the designated mission. The RFP has dictated that the power generation capacity of the *Condor* is 450KVA. The system required for this power generation also includes an ECS cooling capacity of 225 KVA and 250KVA for the radar.

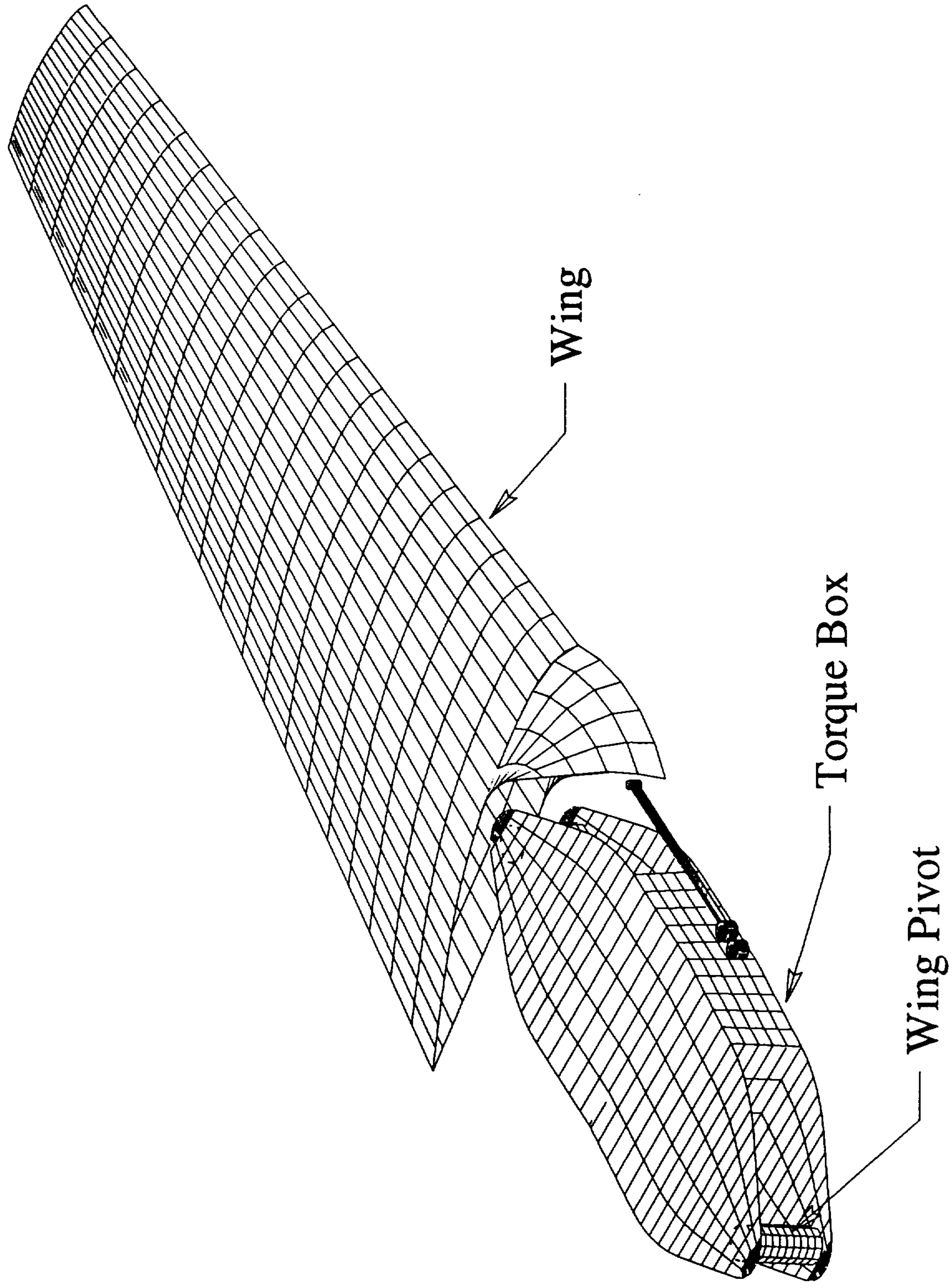


Figure 3.2.8.1: Wing Pivot System

3.2.8 Variable Wing Sweep System

The relatively simple design of the wing sweep system can found in Figure 3.2.8.1. As can be seen from the drawing, the sweep system consists of a pivot and an actuator. The wing is swept through a range of leading edge angles from +5 deg. to +60 deg. The hydraulic actuator is located in front of the pivot structure.

3.2.9 Fuel System

The fuel needed for the *Condor* to complete the required mission specification, as listed in the RFP, is stored in the torque box of the wing and in a fuselage tank under the wing pivot. The total fuel carrying capability of the aircraft is 15,600 lbs as described in Section 2.5. Each wing contains 123 cubic feet of fuel storage between the front and aft spars. The fuselage integrated tank has a volume of 73 cubic feet. The overflow tanks are located in each wing outboard of the primary tanks to relieve excessive pressure. The general orientation and relative size of the fuel tanks can be found in Figure 3.2.9.1.

The tanks can be refueled through the use of a single point refueling port on the top surface of the port wing. All three of the primary fuel tanks are connected with flexible fuel lines through the wing pivot mechanism. Fuel dumping capabilities also exist in the case of an emergency landing shortly after takeoff.

3.2.10 Avionics

As dictated in the RFP, the avionics components for the *Condor* include ESM/ECM, attack and search radars and infrared (IR). The ESM/ECM pods are located at the two vertical tail tips. The IR pods are located at the tip of each wing. This location allows for 360 deg. coverage in azimuth and elevation with the wings swept forward and aft. The attack radar is located in the nose cone of the fuselage. For 360 deg. coverage, the surveillance radar is located in a ventral pod on the fuselage. The locations of the various components can be found in the three view and inboard profile in Section 3.1.

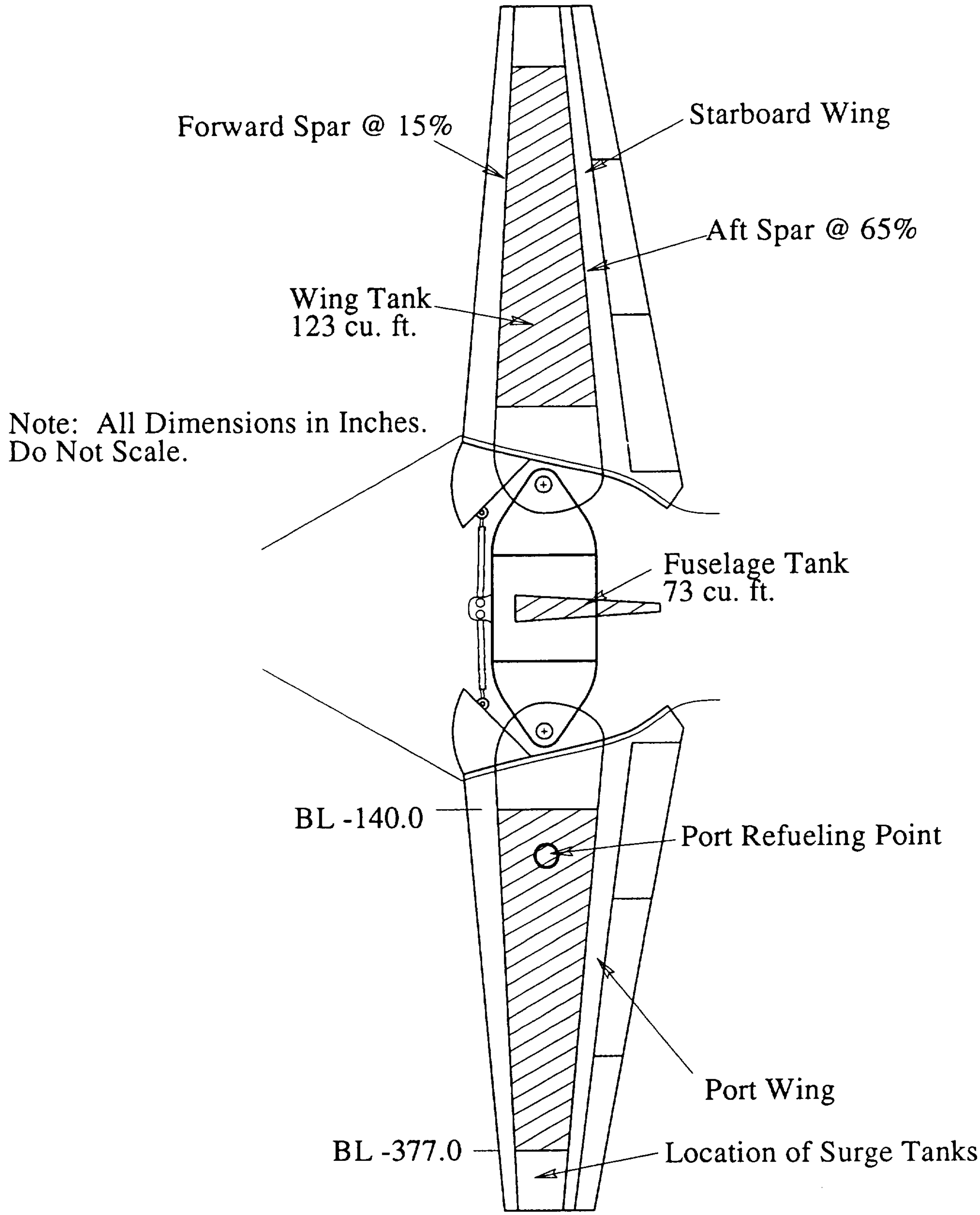


Figure 3.2.9.1: Fuel Tanks for the Condor

3.2.11 Weapons

For a mission designed for anti-surface attack, the *Condor* is required by the RFP to carry the following weapons:

- Two McDonnell Douglas RGM-84 Harpoon Missiles
- Eight Flares or Markers

An anti-air payload may include the following:

- Two Raytheon & Hughes AIM-120A AMRAAM Missiles
- Two Raytheon & Ford Instrument AIM-9 Sidewinder Missiles
- Four Flares
- Four Chaff Dispensers

The Harpoon, AMRAAM and Sidewinder missiles are located on the side of the engine inlets. Flare and chaff dispensers are located in the aft fuselage. The weapons are displayed in the tree view and inboard profile in Section 3.1.

3.3 Structural Layout

In this section of the report, a description is given of the primary structural composition for the *MPS-2000 Condor*. To meet the requirements in the RFP for a low cost aircraft, the *Condor* is constructed primarily with conventional methods and materials. In the day and age of environmental awareness, most of the materials selected for the airframe construction can be recycled and reused.

A more detailed description of the primary structural components of the *Condor* can be found in Section 3.3.1. The material composition for the structure is described in Section 3.3.2

3.3.1 Structural Components

The primary structure for the fighter is composed of a series of frames and longerons supporting a load bearing skin. The sizing and placement for the structural members has been based on methods found in Reference 4.

The fuselage structure is designed to accommodate attachment points for the wing, engines, empennage, cabin and landing gear. According to recommendations found in Reference 4, the fuselage must be capable of withstanding the following loads without failure or fatigue damage:

- Empennage loads due to trim, maneuvering, turbulence and gusts
- Cabin pressurization
- Landing gear loads experienced during impact and taxiing
- Propulsion loads
- Wing loads

To support the loads required of it, the fuselage is comprised of 2 inch frames spaced at 20 inch intervals. This design consideration is consistent with other trainers and fighters as recommended in Reference 4. Additional or heavier frames can be found to support the structure and loads for the wing pivot, empennage connection and landing gear attachment. The frames are tied together with longerons at 10 inch intervals. A detailed drawing of the fuselage structural arrangement can be found in Figure 3.3.1.

The structural composition for the wing, such as the one found on the *Condor*, must include in the following considerations:

- Pivot location, integration, and mechanism
- Fuel placement
- Control surface attachment
- Circulation control system allowances

To account for all of the loads and considerations required of the *Condor*, the wing structure is comprised primarily of spars, ribs and stiffeners. A forward and an aft spar comprise the torque box of the wing. The forward spar is located at 15% of the local chord and the aft spar is placed at 65%. This spar placement allows for a 30% flaperon with 5% clearance for the blowing systems required for the flaps. Both spars are connected at the pivot structure and extend spanwise to the wing tips. The ribs are spaced at intervals of 24 inches and support the skin and stiffeners. The structural arrangement for the wing can be found in Figure 3.2.2 The empennage structure is similar to that found on the wing minus the pivot structure.

The wing pivot consists of a pin of 14.5 inches in diameter. The structure supporting this pin must carry all of the lift, drag and bending moment produced by the wing. The structural configuration for the pivot system is sized to those found on other variable swept wing fighters. Figure 3.2.8.1 in Section 3.2.8 displays in detail the pivoting structure for the *Condor*.

3.3.2 Materials Selection

The primary structure for the fuselage for the *Condor* is comprised of conventional aluminum alloys. Although a weight gain may be realized with aluminum as opposed to using composite materials, this decision is based on environmental concerns (described in Section 7.3). Proven methods exist today for the recycling and reuse of aluminum. To this design team's knowledge, no acceptable methods are present today to adequately dispose of composite materials. Due to the extreme heat produced by the two engines, the supporting structure for the powerplants is titanium.

Due to the added complexity and weight associated with a variable swept wing, the design team, of the *Condor* accepted the environmental consequences and chose composite materials for the wing and empennage. As can be seen from Figure 3.3.3, the torque boxes and trailing edge control boxes of the wing and empennage are constructed of carbon composite materials. Not only is a weight savings realized with the non-metallic materials, but composite components usually contain fewer parts than similar metallic components[15]. A lower parts count leads to fewer mechanical fasteners. The leading edges of the wing and empennage are assumed to be aluminum. The metal leading edges protect the composite materials from erosion experienced from sand, saltwater, etc. The pivot structure for the variable swept wing is aluminum. The pivot mechanism is stainless steel

Not shown in Figure 3.3.3 are the avionics pods described in Section 3.2.10. The pods are all constructed of plastic or composite materials. Special care must be taken in the design of the radomes such that all are electrically grounded. According to Dr. Roskam at the University of Kansas, plastic or composite radomes have a tendency to build up electrostatic from the avionics equipment known as "P" static. The static build up eventually produces a spark that causes electromagnetic interference with navigation equipment.

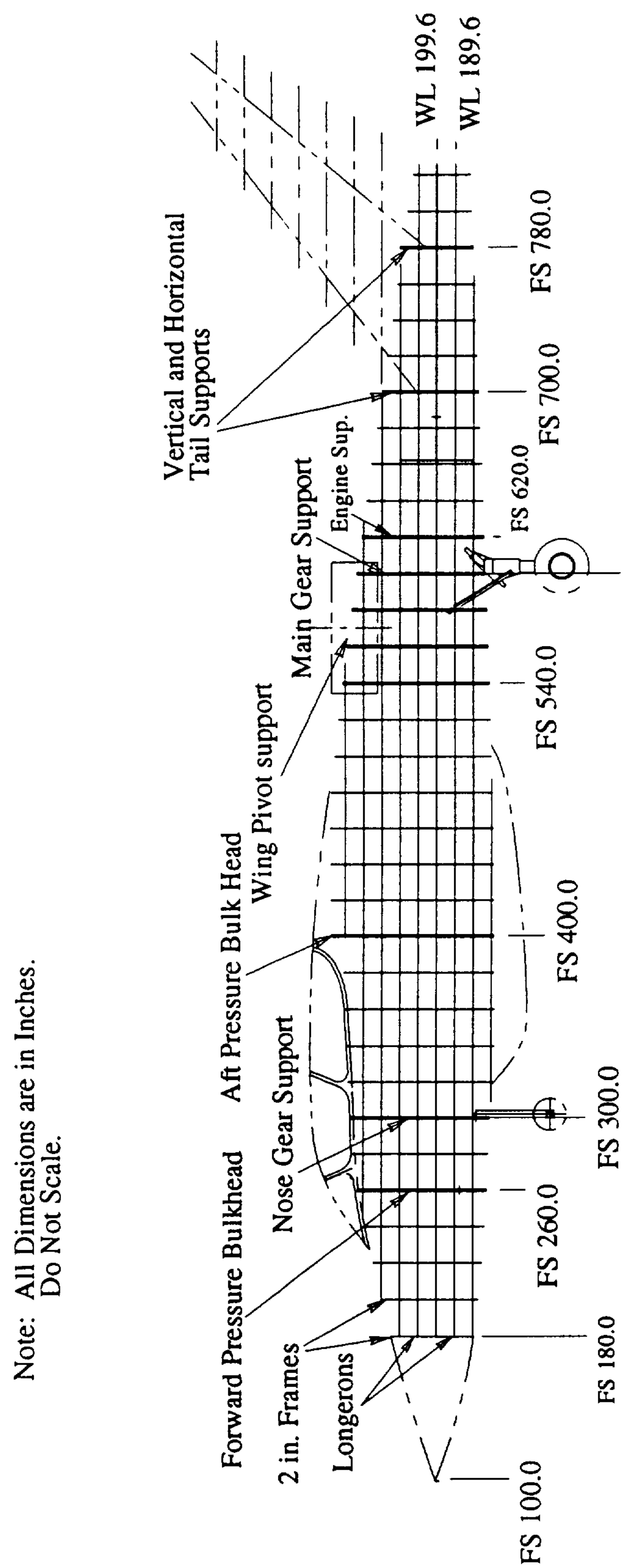


Figure 3.3.1 Structural Arrangement for the Fuselage

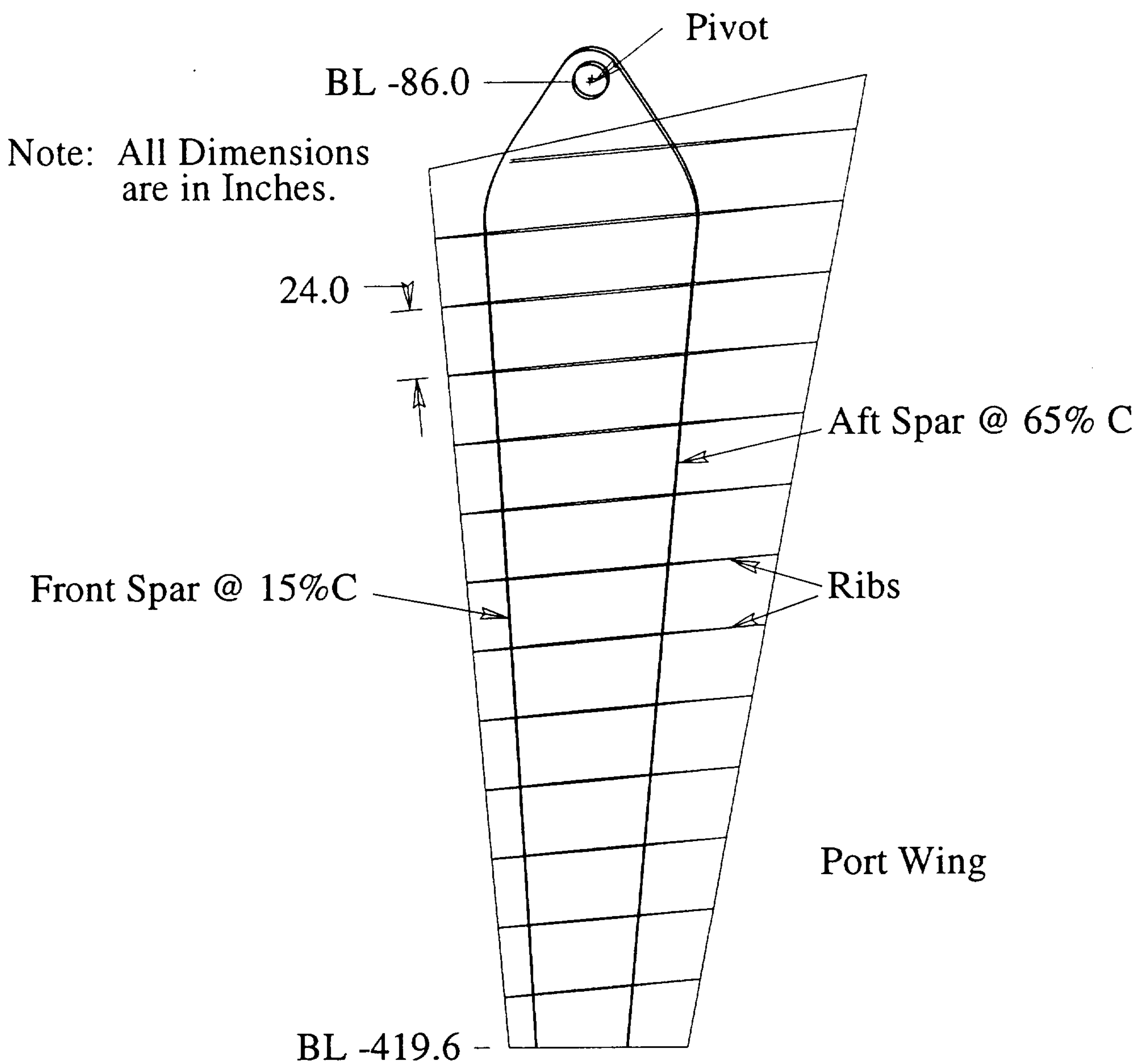


Figure 3.3.2 Structural Arrangement for the Wing

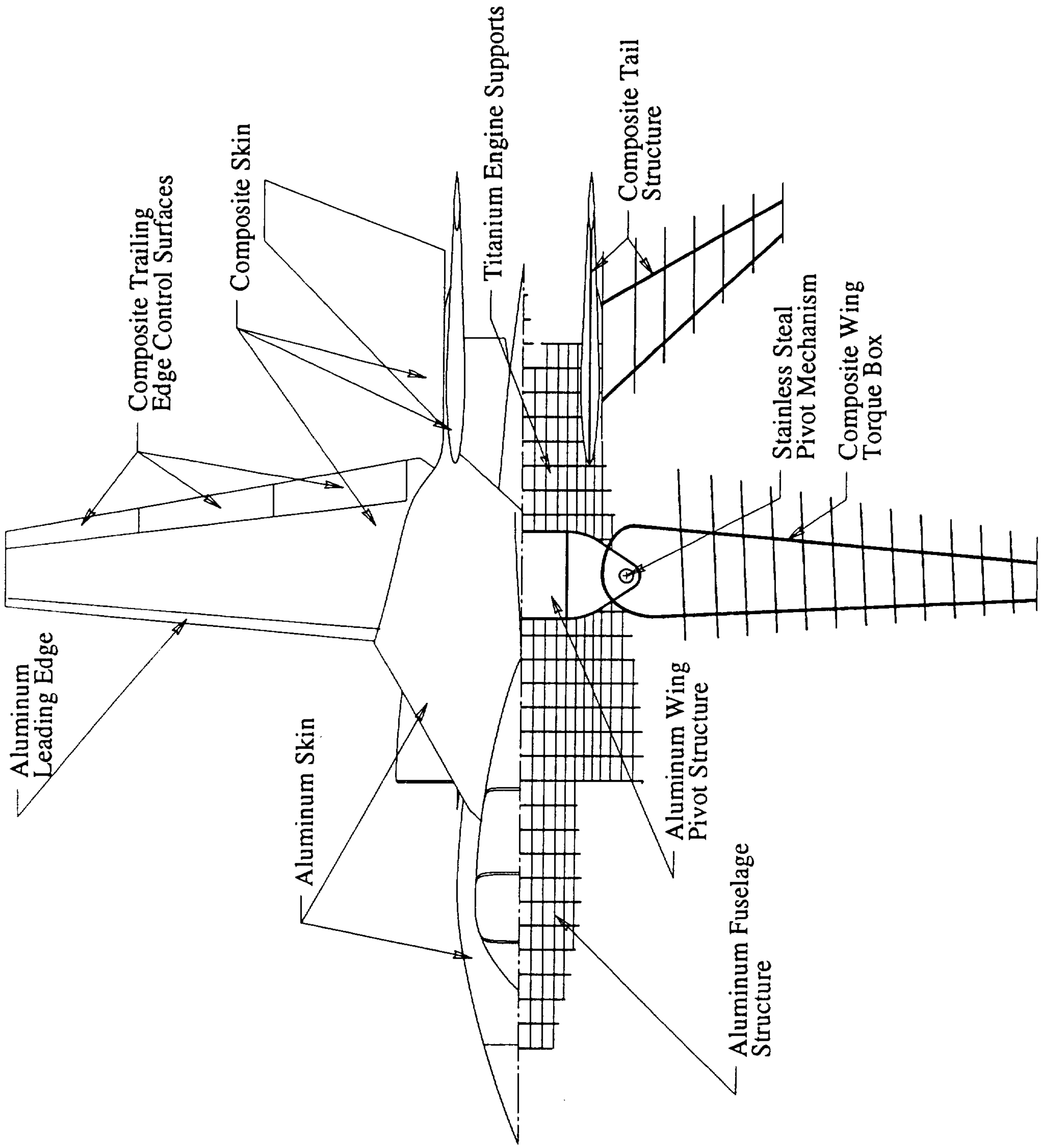


Figure 3.3.3 Materials Breakdown for the Condor

3.4 Maintainability and Accessibility

To minimize the operating cost for the *Condor*, the aircraft design must provide easy access to the major systems of the aircraft for routine maintenance. Ideally for the maintenance crews, the entire skin of the aircraft would be composed of a multitude of access panels exposing each system. However, realistic and air worthy structural considerations prevent the designers of the *Condor* from catering solely to the maintenance crews. Therefore, a compromise is reached to allow relatively easy access to the major components of the aircraft without sacrificing the structural integrity of the airframe.

A common problem with today's conventional military fighters is engine access. Unlike commercial aircraft with podded engines mounted on the fuselage or under the wing, most engines in fighter aircraft are buried within the fuselage. Instead of merely opening a nacelle on a commercial aircraft, a fighter with a buried engine must allow special access within the fuselage. This added complication could add weight to the aircraft and time for the maintenance crews. The more complicated the processes to access an engine (i.e. removing a wing, empennage, etc.), the more expensive the design.

The basic configuration design of the *Condor* is similar to most other fighters with regard to the engines buried within the fuselage. With this in mind, the designers of the *Condor* have devised a means to access and remove the engines without removing any of the surrounding structure or components. Given the relative positions of the engines, tail booms, and fuselage, no primary structural members are located directly under the engines. This structural arrangement allows for large ventral access panels to the engine compartment. The engines can be lowered directly from the airframe through the panel openings and onto a cart. It is the estimation of the design team that both engines on the *Condor* can be easily removed and replaced within a fraction of the time needed for other fighters requiring partial dismantling of the fuselage for the same operation.

Another concern for the designers of the *Condor* is cockpit access. The canopy for the aircraft is hinged on one side as opposed to an aft hinge or two separate canopies. In other words, flight crews or pilots will not be required to "duck" under the canopy for access to the cockpit. With this unconstrained accessibility, the ejection seats and consoles can be lifted directly out of the cockpit without removing the canopy. The current canopy mechanism does limit pilot ingress and egress to one side only. Pilots or crew members seated adjacent to the canopy hinge must first climb over the empty seats opposite of the hinge.

Finally, the wings of the *Condor* are designed to be completely removable from the fuselage structure. Due to the variable sweep characteristics of the *Condor* wings, The entire load bearing structure of the wings are centered at the pivot. Similar to the pivots found on the Grumman F-14, the wing pivots on the *Condor* can be dismantled. The wings can be removed separately for individual repair or replacement.

3.5 Exceptions to RFP

During the design process of the *Condor*, it became clear to the design team that certain requirements of the RFP needed to be clarified or slightly modified to more closely tailor to the emerging configuration of the *Condor*. While all rules and regulations of the RFP were closely studied and applied, the few exceptions collectively conceived by the design team can be rightly justified and it is assumed that an acceptable solution can be successfully presented to the examiners of this design proposal.

An early discovery of a possible configuration problem entailed the search radar. The RFP requested a radar with an array aperture of 250" x 40" elliptical with 120 deg. coverage and 360 deg. total coverage. Given the preliminary sizing of the *Condor*, it was discovered that the radar area, if it is assumed that the physical size of the radar matches its aperture, is a large percentage of the total planform area of the aircraft. In addition, it is further stated in the RFP that the entire radar system only weighs 300 lbs. It became apparent to the design group that a radar with physical dimensions equal to this aperture of this size would be inappropriate for a maritime patrol strike aircraft operating from a Tarawa class ship. In addition, the weight supplied of 300 lbs for seemed to the designers to be contradictory to the large physical size of the radar. During the research phase of the initial sizing and radar system design of the *Condor*, numerous telephone conversations were conducted with Mr. Patrick Gouhin of AIAA in an attempt to clarify the radar requirement. However, Mr. Gouhin was unable to provide the design team any additional information concerning the radar and instructed the team to consider the possibility of an error in the RFP and to make an assumption for the radar system. The final solution to the radar size dilemma is an abandonment of the RFP requirement and the adoption of a commercial radar system, a Heracles II surveillance radar.

Another suggested modification to the RFP is in regard to the takeoff and landing platform suggested in the RFP. Although the *Condor* meets the field length requirements as suggested in the RFP for a Tarawa class ship (see Section 2.6), the operation from a carrier of the WASP or TARAWA type ship presents restrictions for an airplane of the size of the *Condor*. Three possible modifications to the carrier flight deck are proposed (See Figure 3.5.1). It is realized that this will increase the acquisition cost of the aircraft, but the operational advantage gained is considerable.

The preferred options is the bow-starboard angled deck. This allows one airplane to takeoff while the next airplane is next to the head of the runway and the other airplanes are in line. The second advantage is that this type of angled flight deck presents the least amount of modification of the three flight decks proposed. The second most preferred is the full-starboard angled deck which gives the longest runway but requires the biggest modifications. The third type (stern-port) is not considered acceptable because of runway-elevator interference. in addition, waiting airplanes must be parked at the bow requiring to taxi back to the stern for takeoff.

The final exception to the RFP is based on the suggestion for the use of lift fans. The large takeoff weight required for the rigorous mission specification combined with the relatively short field length almost certainly requires aircraft high lift capabilities beyond the scope of present day wing-flap technology. Therefore, it has been suggested by the RFP to possibly incorporate lift fans in the configuration of the *Condor* to artificially increase lift and to aid in shortening the field length of the aircraft. However, based on preliminary research conducted by the design team, it was decided by the group that the structural and aerodynamic penalty suffered by incorporating large holes within the airframe for the fans out weighs the gain in takeoff and landing performance. Therefore, a possible alternative to the RFP for the generation of high lift is a flap blowing system. Bleed air from the engines is ducted to blow over the flaps for increased lifting capabilities of the wing and flaps. Further details of this system can be found in Section 2.4.

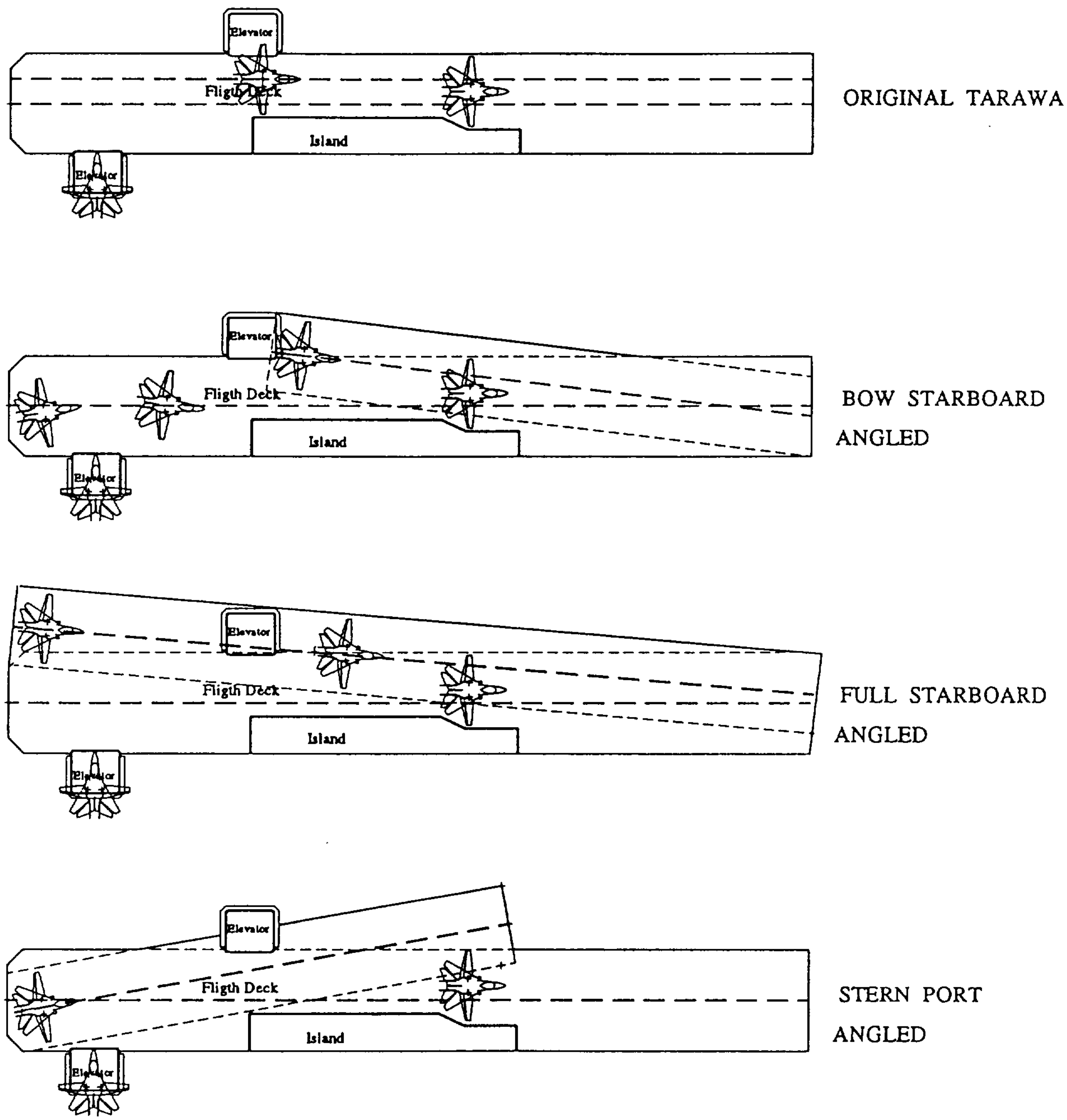


Figure 3.5.1: Proposed Aircraft Carrier Modifications

4. Management Organization

For an aircraft production program to become truly successful, not only must there be a promising design, but there must also exist appropriate management to ensure proper program procedures. The entire aircraft development program from initial sizing and configuration layout to final assembly and support must be well thought out and organized. The management "design" for the *Condor* is based on already established and proven aircraft development programs. The design team of the *Condor* has utilized the "field" experience of Dr. Roskam from the University of Kansas to help shape the management organization of the aircraft program. The proposed management staff for the program is described in Section 4.1 and procedures of assuring schedules and quality and cost control can be found in Section 4.2.

4.1 Biography of Proposed Management Staff

The management staff for the development of the *Condor* program is primary comprised of one manager in charge of the program and the various levels of development managers working collectively under his/her supervision. At the top of the management tree for the *Condor* is the Executive Vice President. The person appointed to this position should have previous work experience in both aircraft manufacturing and design. This position is responsible for overseeing the entire project from the design to the manufacturing. Particularly at the beginning of a new project, the Executive Vice President must encourage the design and production planning departments to work together. This will allow for the establishment of general data, estimates of component weights, and sequences of assembly and jigs[16]. Although various departments are encouraged to work together, any final decision making is the responsibility of the Executive Vice President.

Under the direct supervision of the executive vice president are the Director of Design Engineering and the Director of Manufacturing. The two directors work together in what is referred to as "total engineering". All aspects of design and manufacturing are decided collectively. For example, in engineering, a design of a given component must be a collective decision between the engineers in aerodynamics, structures, stability and control, etc. Colleagues in manufacturing must also be involved in the decisions to supply meaningful insight in production feasibility. Ideally, all decisions are collective between all members in the team involved in the aircraft production. However, any unsettled matters are resolved by the executive vice president.

A complication that may become present in an aircraft program the size of the *Condor* is international involvement. Aircraft programs have become too complicated and costly for one company alone to design, develop, and finance production of a complete aircraft[16]. A popular solution to this problem is international sharing of investment risks. The U.S. company awarded the contract to produce the *Condor* may split up the development of the program among one or more countries. As expected, the management organization becomes more complicated especially if the countries involved in the aircraft development are geographically far apart. A reliable communication system will have to be utilized to assure proper management of the program from one country to the next.

4.2 Management Procedures

The management organization outlined in Section 4.1 must set up a strict procedure to maintain project schedules and quality and cost control. The design and manufacturing groups must decide on a management procedure and how best to enforce it. The executive vice president must call frequent project meetings with the groups to discuss problem areas in the procedure.

To maintain and meet the schedule outlined for the project, the collective groups must breakdown the task sequentially within the calendar time. A master plan and charts must be produced to be used as a reference for the groups. For each task, the groups must decide on an average, conservative, and a pessimistic time estimate. With this information, total time estimates and costs can be determined with the use of a computer. The managers and program directors must also have the insight to predict learning curves in the design and manufacturing processes. As a procedure is conducted repeatedly, whether it is in the design process or in manufacturing, a progressive reduction in time can be achieved. These learning curves can be used to predict individual and group performance[16]. With a process labeled Program Evaluation and Revue Technique, management must actively seek continuous improvements. Problems or program overruns are highlighted and attract the attention of the management. The management then assigns extra man power (i.e.

engineers, production workers, etc.) to resolve the problem. The entire program from start to final production is under constant scrutiny from the management organization.

5. Manufacturing

Once the management organization has been established and the aircraft program is ready for full scale production, a plan of attack must be adopted for the actual manufacturing of the aircraft. Careful consideration must be used when selecting or constructing facilities for the production, as described in Section 5.1. In the case in which more than one facility may be utilized, a component breakdown for the aircraft must be developed as suggested in Section 5.2. The manufacturing process as it applies to the *Condor* is based primarily on recommendations made by Dr. Jan Roskam from the University of Kansas.

5.1 Facilities Required

A detailed analysis of the total spectrum of facilities required for the construction of the *Condor* is beyond the scope of this proposal. Total floor area, tooling and manpower needed for the program production is dependent on the forecasted maximum output rate. The predicted output rate for the *Condor* is indicated in Section 7.2. As mentioned in Section 4.1, international involvement in the program also plays a key role in the facilities required and the transportation of the components. The final assembly line must be sized for the size of the aircraft, the tooling equipment needed and number of people operating on the line.

Specifically for the *Condor*, facilities are required for the production of both metallic and composite components. The aluminum components (i.e. fuselage frames, etc.) can be constructed with conventional methods to reduce developmental costs. Additional methods and tools will have to be developed for the construction of composite components. Due to the attraction of lower parts counts and weight savings, several components in the wing and empennage are composite as described in Section 3.3. An automated system must be developed for the dispensing and laying of tape with pre-determined orientation and ply sequencing[15]. Autoclaves will also have to be utilized for curing the composites.

5.2 Component Breakdown

Since the likely hood is great that the *Condor* will be produced in numerous facilities, a preliminary component breakdown is necessary for the facility planning. The major components of the airframe are divided for manufacturing as shown in Figure 5.2.1. Each component can then be produced in separate manufacturing plants and then transferred to the final assembly plant. Government Furnished Equipment (GFE) such as engines, avionics, radars, etc. will be delivered by the government and thus require no floor space for production in the *Condor* line. Storage will be supplied for the GFE components will have to be allocated.

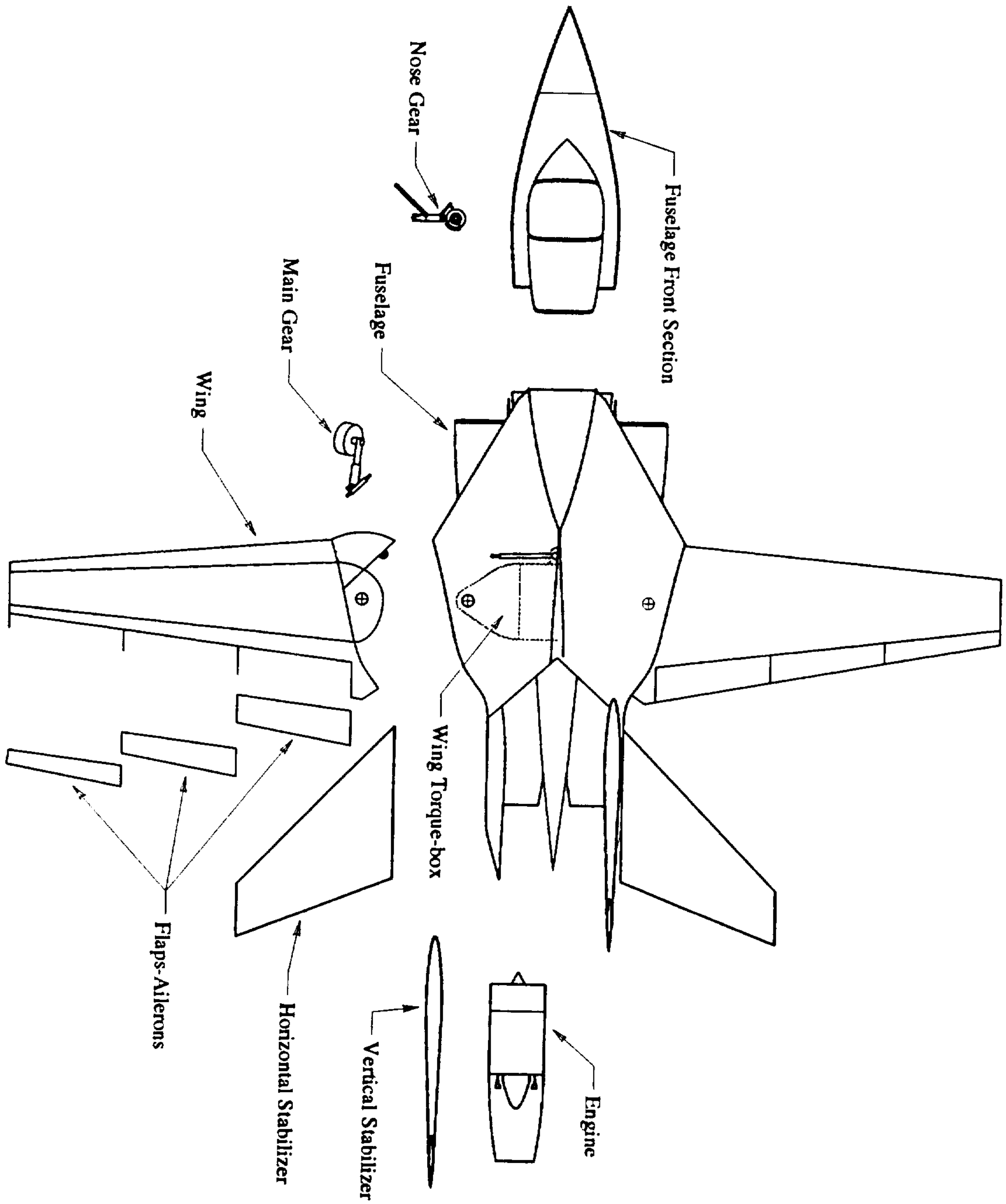


Figure 5.2.1: Manufacturing Breakdown

6. Final Design Trade Studies

The development of the preliminary design for the *Condor* requires numerous trade studies The trade studies discussed in this proposal are:

- Takeoff Wing Loading vs. Thrust
- Engine Performance vs. Altitude
- Flap Blowing vs. Field Length

6.1 Takeoff Wing Loading vs. Thrust

The purpose of this section is to present the results of the performance trade studies performed to determine the design point values for the takeoff wing loading and thrust required. Figure 2.6.3 shows that the takeoff condition is the design driver for the *Condor*. Trade studies were performed to find a suitable combination of lift coefficient, thrust-to-weight ratio, and takeoff field length. Figures 6.1a and 6.1b show the sensitivity of the wing loading and thrust-to-weight to changes in the lift coefficient and takeoff field length. While it is intuitively obvious that a short field run and a high lift coefficient are desirable, these graphs show the magnitude and serve to define upper and lower limits on possible values for the study. The results of this study are:

- $S_{TOG} = 470$ feet
- $C_L = 5.4$
- $W/S_{TO} = 66$ psf
- $T/W_{TO} = 0.61$

6.2 Engine Performance vs. Altitude

The purpose of this section is to present the results of the performance trade studies performed to determine the engine performance variations with altitude driving the engine selection process. Figure 6.2a shows the engine performance with altitude curves for the BMW 710-15 turbofan engine. Similar curves were constructed for each engine studied to verify the satisfaction of the performance requirements. The deciding factors were then size, weight, and, most importantly, fuel consumption. Figure 6.2b shows another performance curve for the GE F110 afterburning turbofan engine. The differences between the BMW 710-15 and the GE F110 are a drastic savings in weight with the BMW and a large savings in specific fuel consumption. The magnitudes of these can been seen by comparing the engine information listed on each graph. The BMW 710 was chosen because it possessed the "best" combination of each of the desired qualities and met the thrust requirements for each flight phase.

6.3 Flap Blowing Parameters

The takeoff studies covered in Section 6.1 defined the performance parameters that must be created by the blown flaps, specifically the lift coefficient. Using the methods described in Reference 14, this translates into required values for these blown flap parameters:

- Blowing Mass Flow Rate, m_dot
- Momentum Coefficient, $C\mu$
- Blowing Velocity, Vj
- Effective Blown Flap Area, S'

The cap on the amount of mass flow that can be delivered was set by the engine or APU chosen. The higher the mass flow rate the higher the lift coefficient. Also, as the momentum coefficient, $C\mu$, increases above 3.0, enhancements can be taken in attainable angles of attack for the aircraft on the order of 7 degrees increase. This aids in increasing the lift coefficient for a given $C\mu$. The blowing velocity is primarily affected by the size of the blowing orifice. The larger the opening, the lower the velocity. However, as the opening size increases, so does the mass flow rate for the given lift coefficient. Trade-offs must be made. In this case it is more critical to keep the mass flow manageable because the flow velocity is not near sonic speeds. To attain the desired lift coefficients at lower $C\mu$'s it was decided to use full span flaps and blow over the entire span. This produces an effective blowing area, S', of 500 square feet. The pertinent results are listed below.

- $C\mu$ = 4.8
- Vj = 472 fps
- AOA = 14 deg
- m_dot = 50 lbm/sec
- $C_{L\alpha}$ = 11.0 1/rad
- Flap AOD = 24 deg

Figures 6.3a, 6.3b and 6.3c show some trade study curves for the lift coefficient, mass flow rate, and blowing velocity respectively.

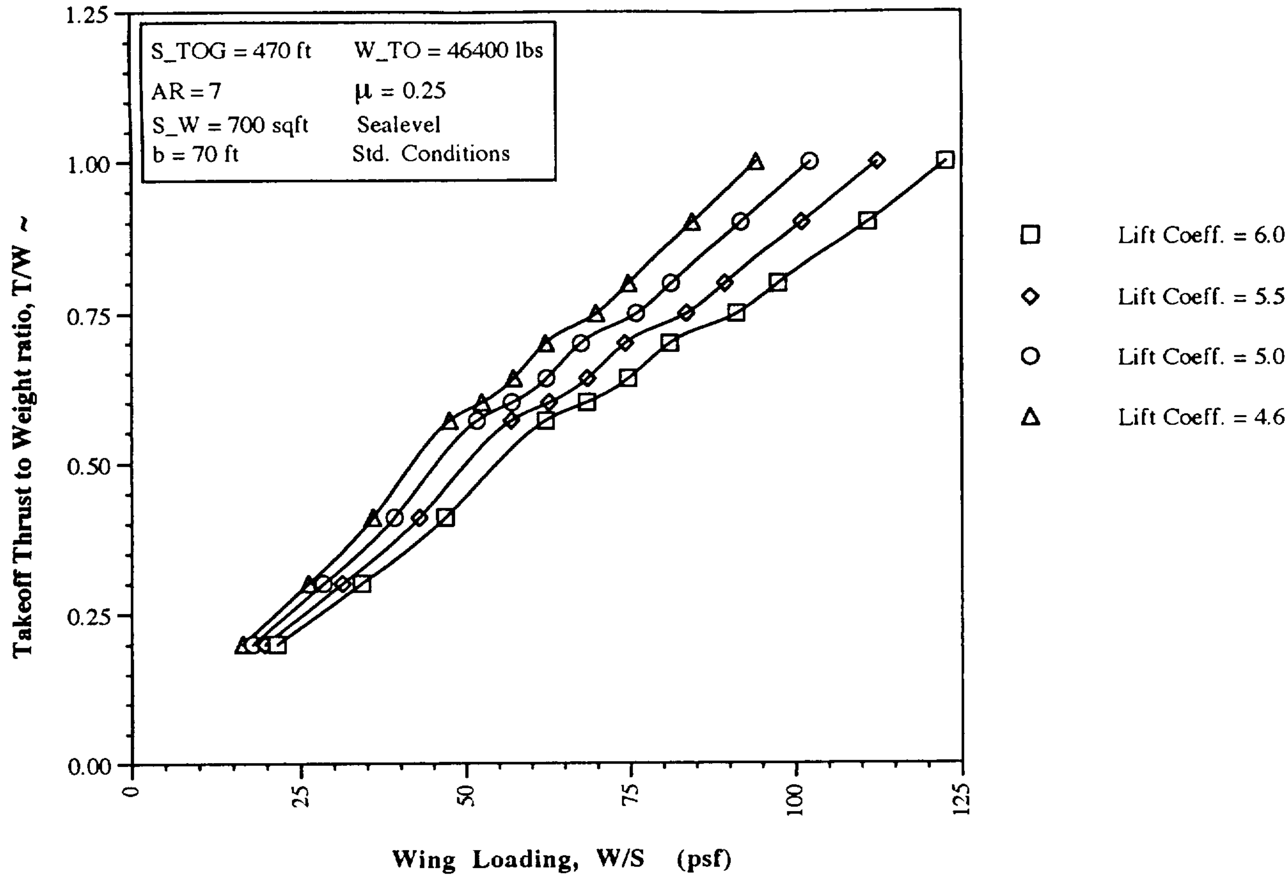


Figure 6.1a: Takeoff Wing Loading Variation with Thrust to Weight Ratio for the Condor

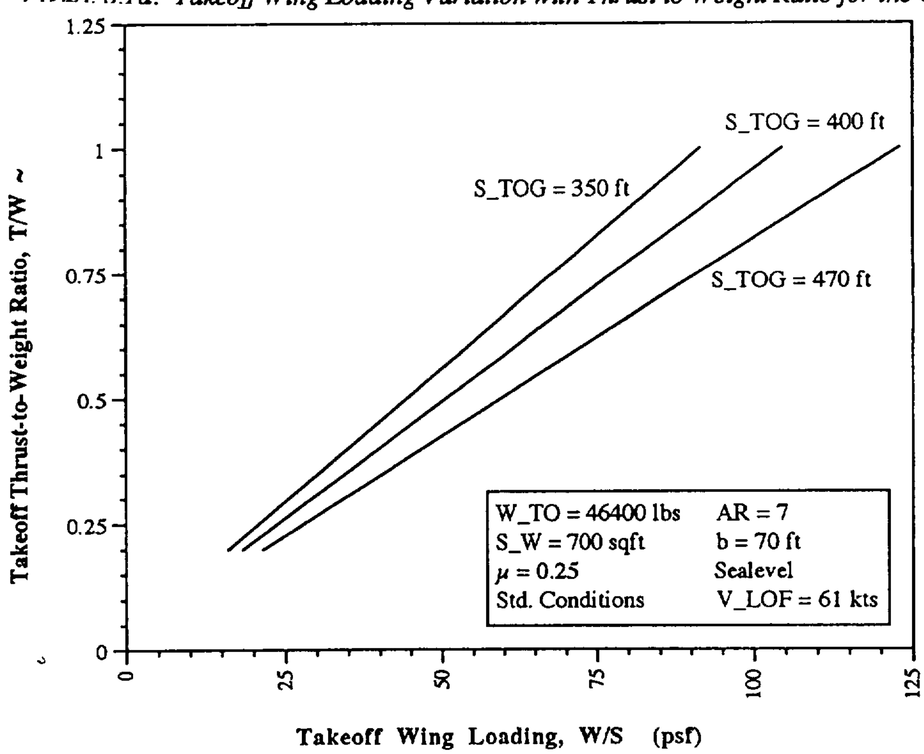


Figure 6.1b: Variation of Takeoff Wing Loading and Thrust to Weight Ratio with Takeoff Field Length

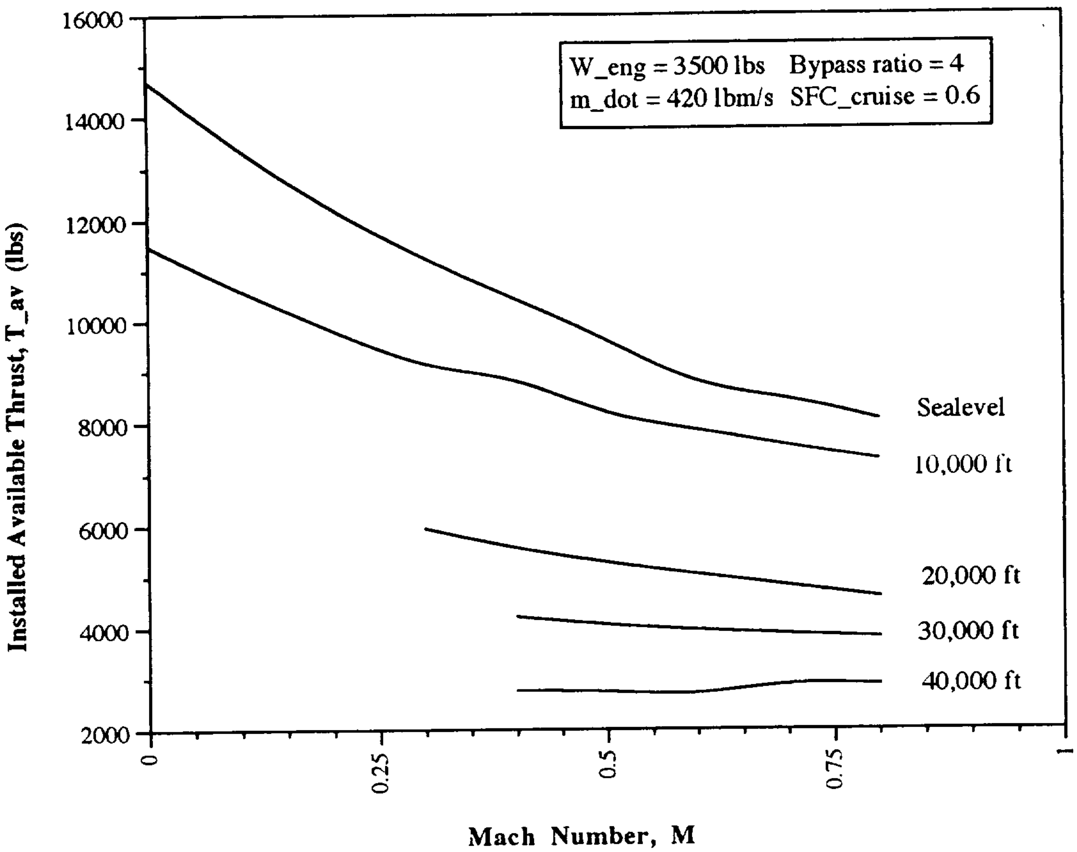


Figure 6.2a: Installed BMW 710-15 Engine Available Thrust Variation with Altitude and Mach Number

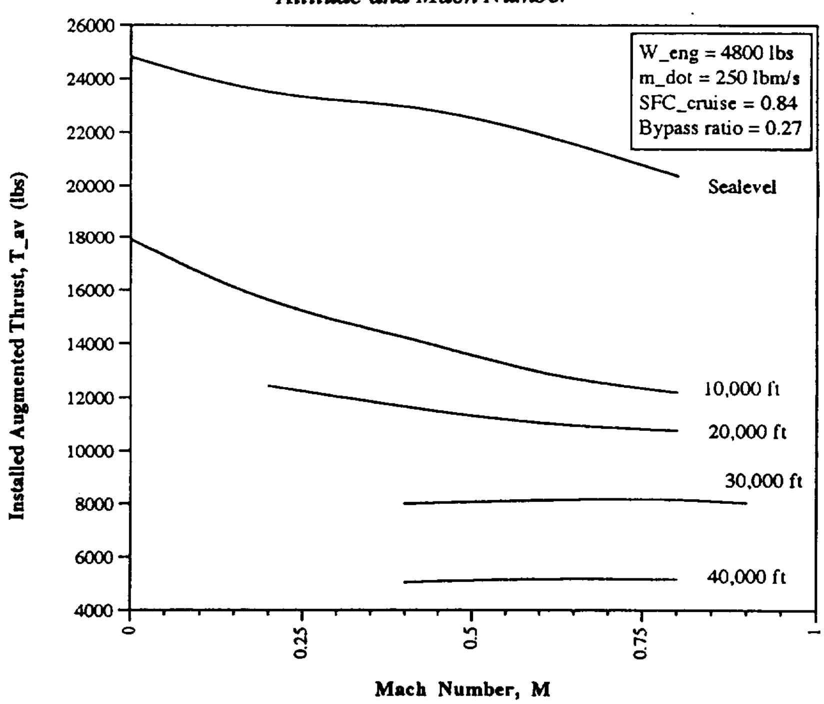


Figure 6.2b: Installed GE F110 Engine Augmented Thrust Variation with Altitude and Mach Number

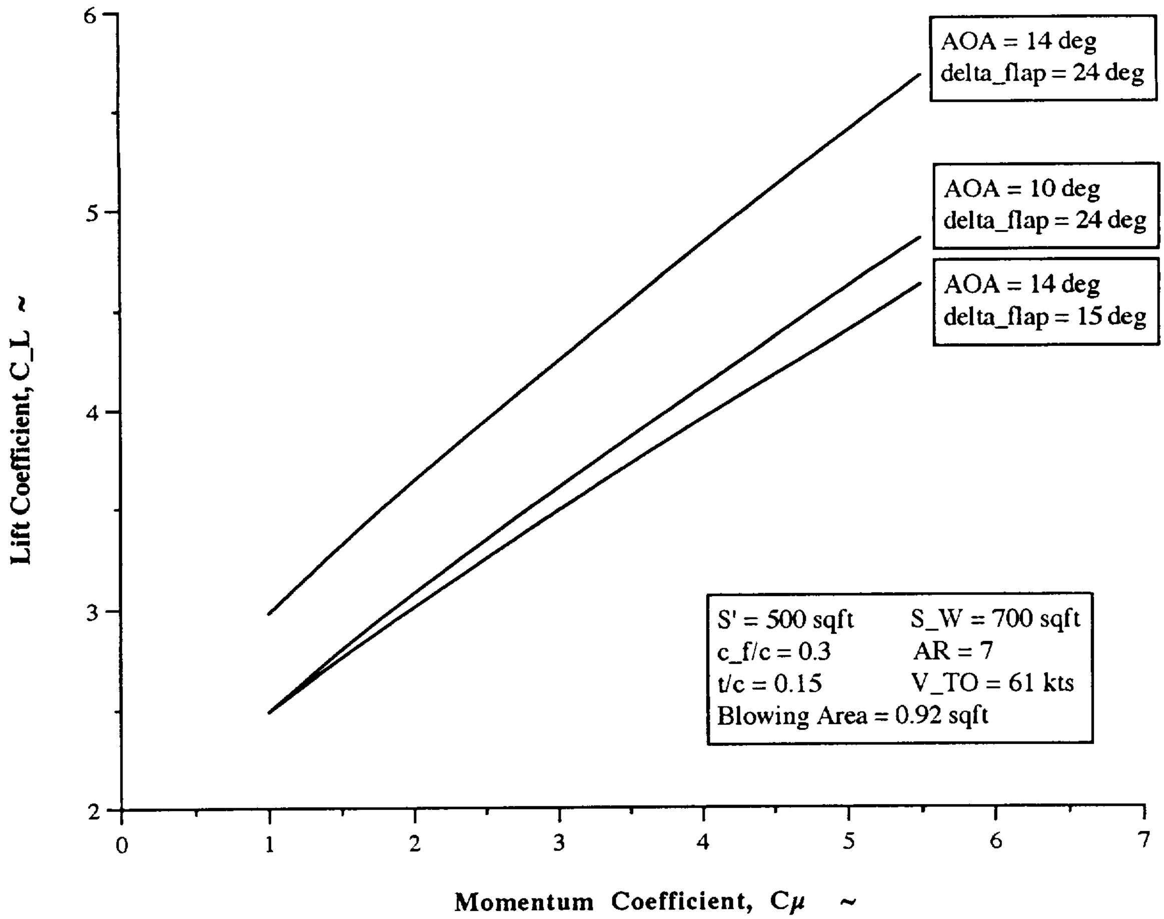


Figure 6.3a: Sensitivity of the Lift Coefficient due to Blowing Upon Changes in Angle of Attack and Flap Deflection Angle

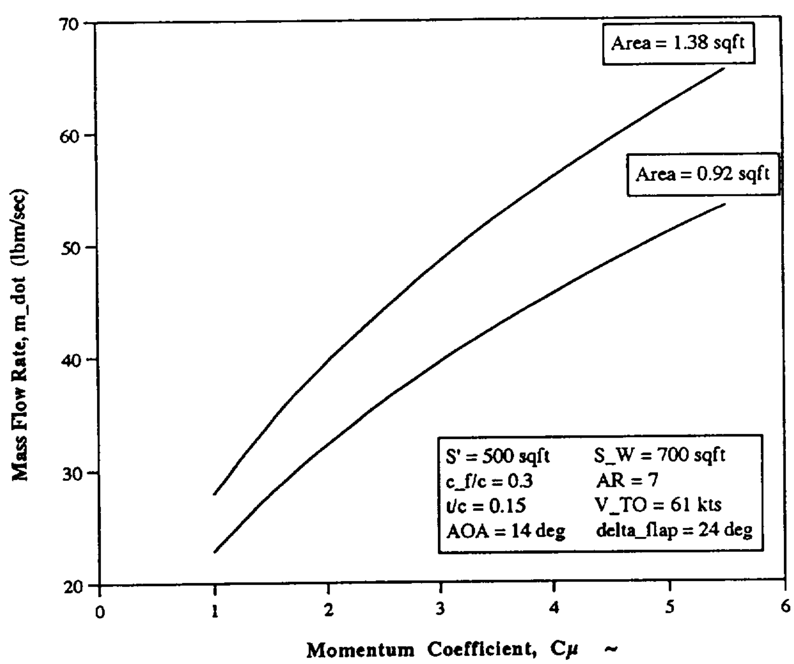


Figure 6.3b: Sensitivity of the Blowing Mass Flow Rate to Changes in the Exit Blowing Area

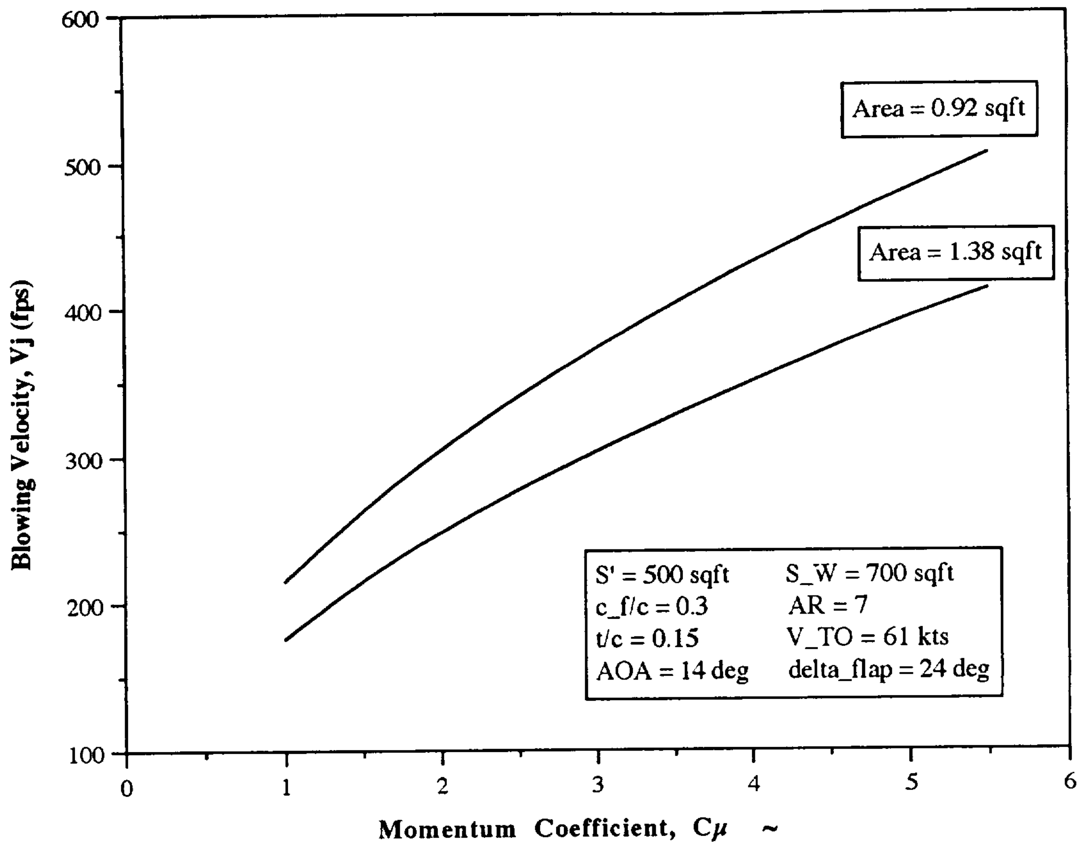


Figure 6.3c: Sensitivity of Blowing Velocity to Changes in Exit Blowing Area

7. Implementation Plan

The design of the *Condor* and the feasibility of production must be realized in practical applications of the aircraft. Consumers of the aircraft program will also be interested in the cost and the life-time environmental impact of the program. These issues are described as they relate to the *Condor* in the following sections.

7.1 Program Applications

As dictated in the RFP, the *Condor* is capable of searching, monitoring and securing large expanses of water and performing coastal and overland surveillance. The surveillance equipment and weapons on board the aircraft allow the *Condor* to intercept small, fast surface craft and other small, armed aircraft. These capabilities allow the military *Condor* to serve as support for military forces, as a defense against Joint Tactical Ballistic Missiles and drug interdiction. In a commercial variant, the radar system could allow the *Condor* to be used in search and rescue missions.

7.2 Cost Analysis

In this section, the cost estimation for the *MPS-2000 Condor* is presented. The Advanced Aircraft Analysis (AAA) program was utilized to perform the cost analysis. The cost was computed by assuming production runs of 150, 300, 500, 1000, 1500 and 2000 airplanes.

The cost study for the *Condor* involved varying the following parameters:

- Total Airplane Production
- Number of Airplanes for RDTE
- Annual Flight Hours
- Materials
- Fuel Price

The change of Life Cycle Cost (LCC) and the airplane estimated price (AEP) varying with production runs are shown in Fig 7.2.1. This parameter has the most significant effect on the LCC and AEP.

The LCC is linearly increased with increasing production runs , but the AEP is rapidly decreased with increasing production runs. The total number of airplanes for production was selected to be 300.

Fig 7.2.2 shows the effect of annual mission flight hours on LCC and AEP. The AEP had little variation with the annual mission flight hours, but the LCC had linearly increased. The number of annual mission flight hours was selected to be 120 hrs for 7 hrs for each mission. The number of missions per year was calculated to be 171.

Variation in fuel cost per gallon has an important influence on the life cycle cost. Fig. 7.2.3 shows the effect of fuel price on the LCC. The effect of material and number of airplanes for RDTE on LCC and AEP can be seen in Fig 7.2.4 and Fig. 7.2.5. As can be seen, material and number of airplanes have a negligible influence on the LCC, but the AEP was significantly affected by those two parameters. Eight airplanes are assumed to be needed for RDTE and the conventional aluminum alloys for the airframe were selected for the cost estimation.

7.2.1 RDTE Cost

The Research, Development, Testing, and Evaluation (RDTE) Cost is presented in this section. The total number of airplanes for the RDTE was chosen to be eight, and two airplane was assumed to be produce for the static tests. The RDTE cost breakdown can be seen in Table 7.2.1 and Fig 7.2.6

Table 7.2.1: *The Breakdown for RDTE Cost of Condor*

Component	Cost $\times 10^6$ (USD)
Airframe engineering and design cost	166
Development support and testing cost	66
Flight test airplane cost	692
Flight test operation cost	31
Test and simulation facilities cost	318
RDTE profit	159
Cost to finance the RDTE phases	159
Total estimated RDTE cost	**1,592**

FOLDOUT FRAME 1.

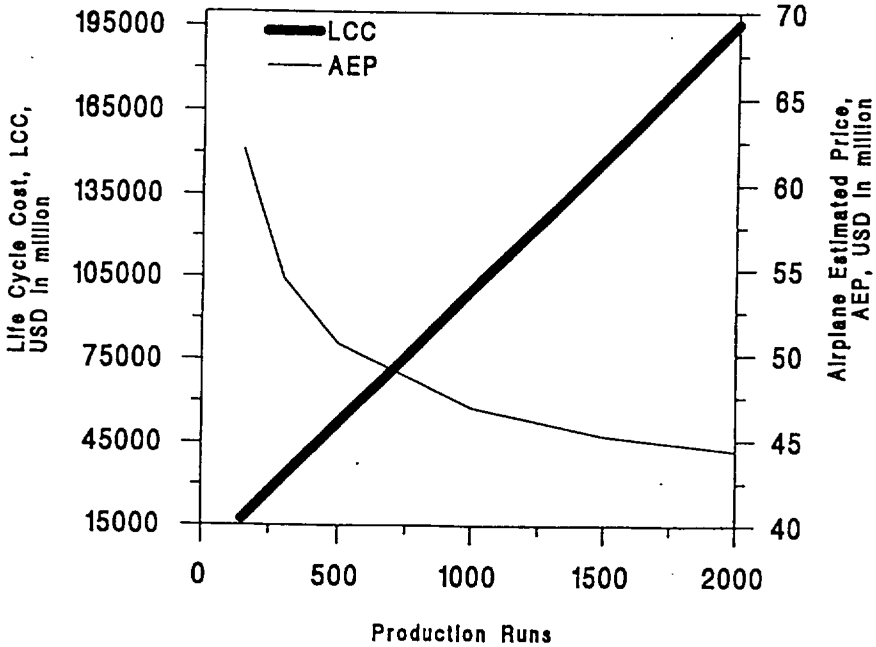


Fig 7.2.1 The Effect of Production Runs on LCC and AEP

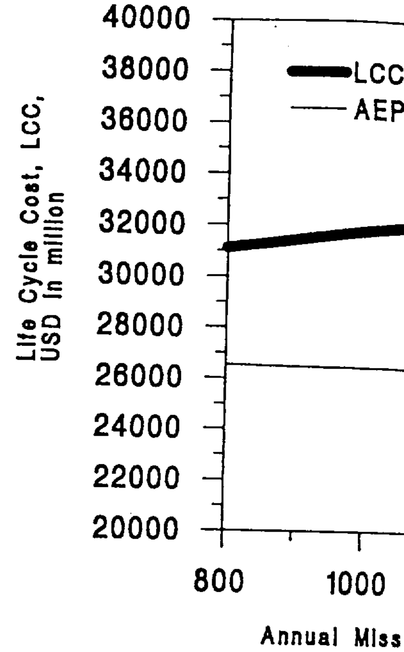


Fig 7.2.2 The Effect of

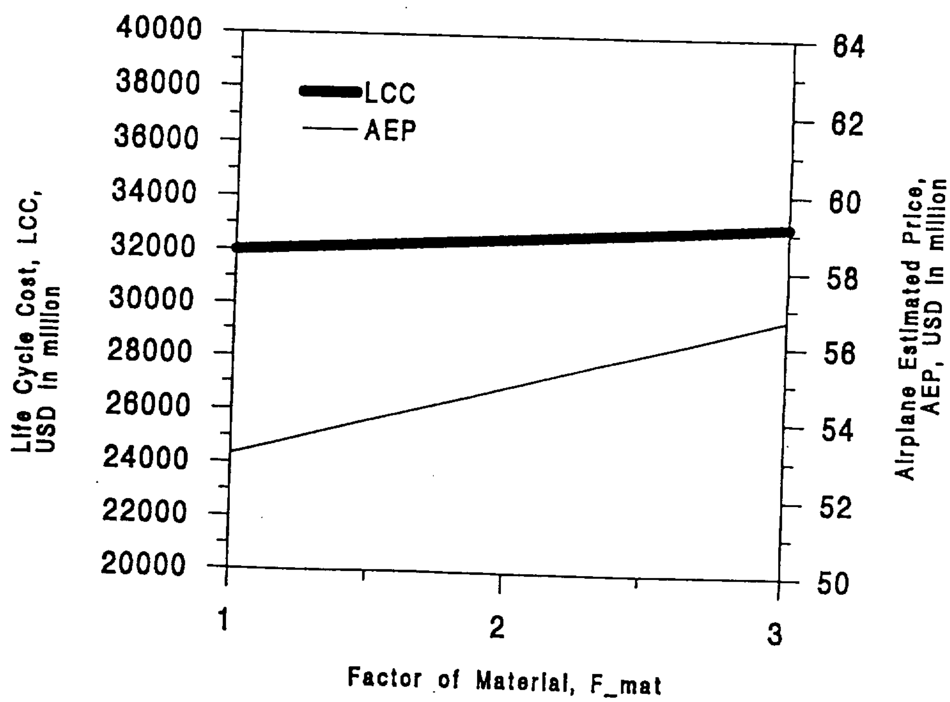
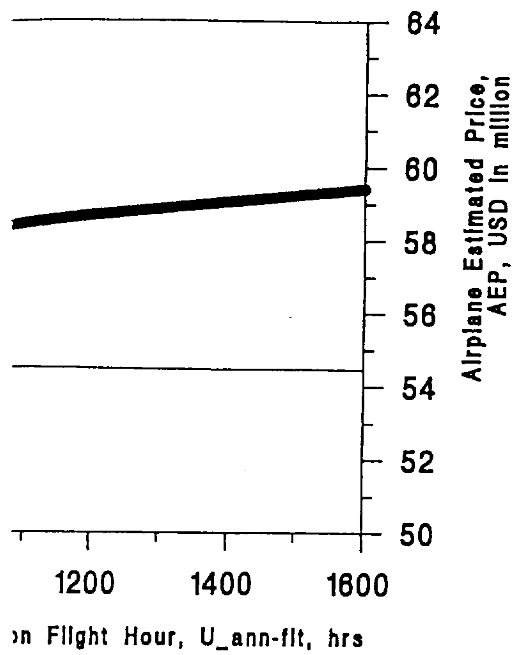


Fig 7.2.4 The Effect of Material on LCC and AEP



nnual Flight Hour on LCC and AEP

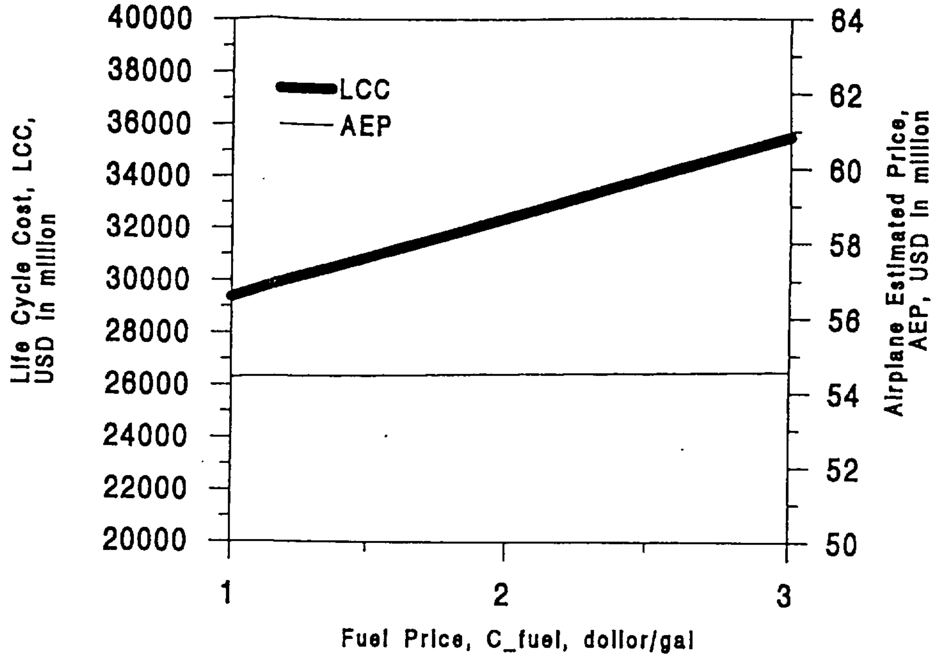


Fig 7.2.3. The Effect of Fuel Price on LCC and AEP

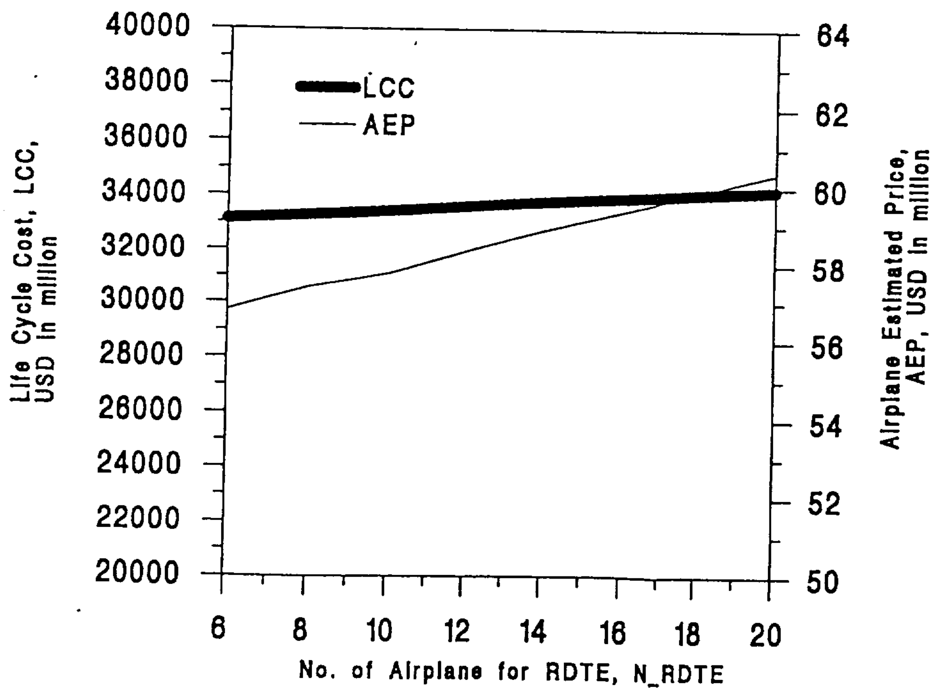


Fig 7.2.5 The Effect of No. of Airplane for RDTE on LCC and AEP

7.2.2 Acquisition Cost

The total number of airplanes for production was selected to be 300 from the trade study in Section 7.2.1. The total test flight hours was also selected as 20 hrs, the manufacturing profit and the manufacturing finance rate were 10 %, the cost of engines was 3.3 million USD and the cost of avionics equipment per airplane was estimated to be 25.8 million USD (included 12.4 million USD for two radar systems). The Acquisition cost for the *Condor* is shown in Table 7.2.2 and Fig 7.2.6

Table 7.2.2: *The Acquisition Cost Breakdown for the Condor*

Component	Cost x10^6 (USD)
Airframe engineering and design cost	160
Airplane production cost	11,816
Production flight test operations cost	96
Cost of financing the manufacturing program	1,341
Total Acquisition Cost	**14,755**

7.2.3 Operating Cost

The purpose of this section is to present calculating of the airplane program costs associated with operation. The price of fuel was assumed to be \$2.0/gallon. The number of annual mission flight hours was selected to be 1200 hrs for 7 hrs for each mission. The maintenance man hours per flight hour was assumed to be 8 hr/hr because of the modern technology improvement on the maintenance. This assumption was based on Fig. 7.2.6.

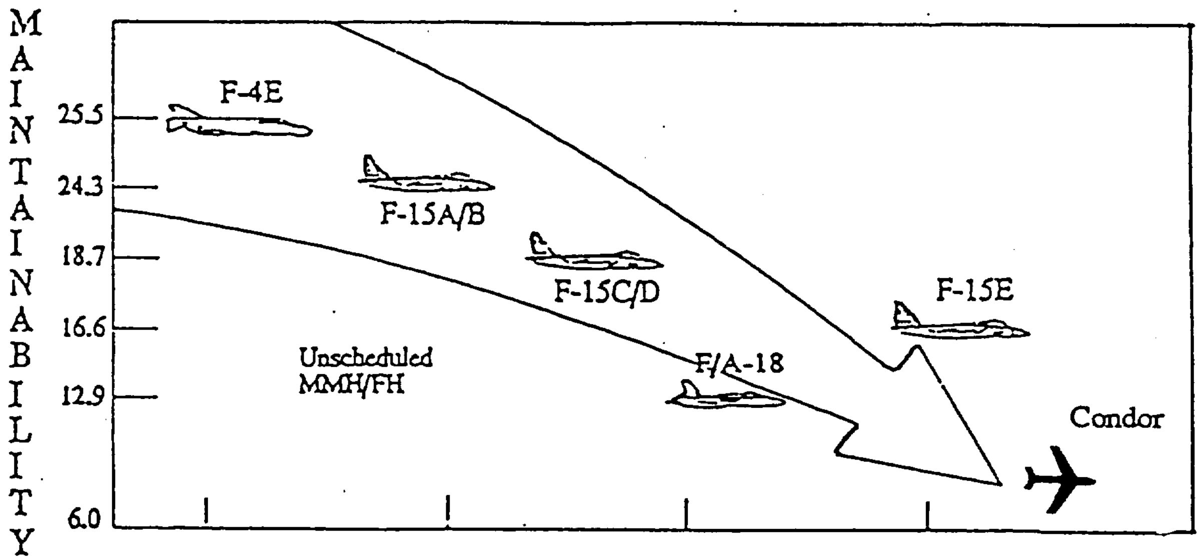


Figure 7.2.6: Maintainability Improvement
(Copied from Ref. 17)

The total operating cost for *Condor* is shown in Table 7.2.3 and Fig 7.2.6.

Table 7.2.3 *The Total Operating Cost for Condor*

Component	Cost x10^6 (USD)
Fuel, oil and lubricant cost	2,827
Direct personnel cost	4,361
Consum. material cost	215
Program cost of indirect personnel	2,835
Program cost of spares	2,363
Program cost of depot	2,205
Program cost of miscellaneous items	945
Total Operating Cost	**15,752**

7.2.4 Life Cycle Cost

The life cycle cost (LCC) is the summation of the RDTE, the acquisition, the operating and the disposal cost. The first three costs were presented in the preceding sections. The disposal cost was assumed to be 324 million USD. The LCC is shown in Table 7.2.4 and Fig 7.2.7.

Table 7.2.4: *The Life Cycle Cost for Condor*

Component	Cost $x10^6$ (USD)
Research, development, test and evaluation cost	1,592
Acquisition cost	14,755
Operating	15,752
Disposal cost	324
Life cycle cost	**32,423**

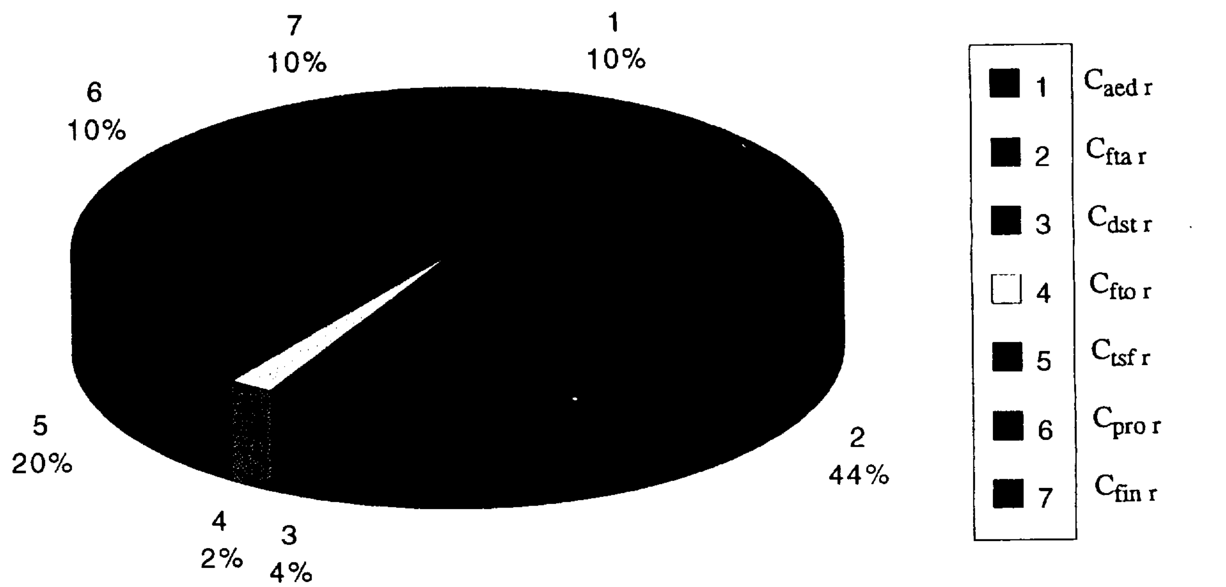
7.2.5 Cost breakdown With Varying Numbers of Airplanes

The effect of unit price per airplane with production runs was presented in the preceding Section 7.2.1. Table 7.2.5 shows the cost breakdown for the AEP calculation varying with production runs.

Table 7.2.5: *The Cost Breakdown for the AEP*

Nm	150	300	500	1000	1500	2000
N_{RDTE}	6	8	10	12	13	14
$C_{RDTE}x10^6$	$1,363	$1,592	$1,812	$2,013	$2,113	$2,212
LCC $x10^6$	$17,363	$32,423	$52,101	$100,408	$148,180	$195,833
AEP $x10^6$	$62.1	$54.5	$50.7	$46.9	$45.3	$44.4

The RDTE Cost Brakdown



The Operating Cost Breakdown

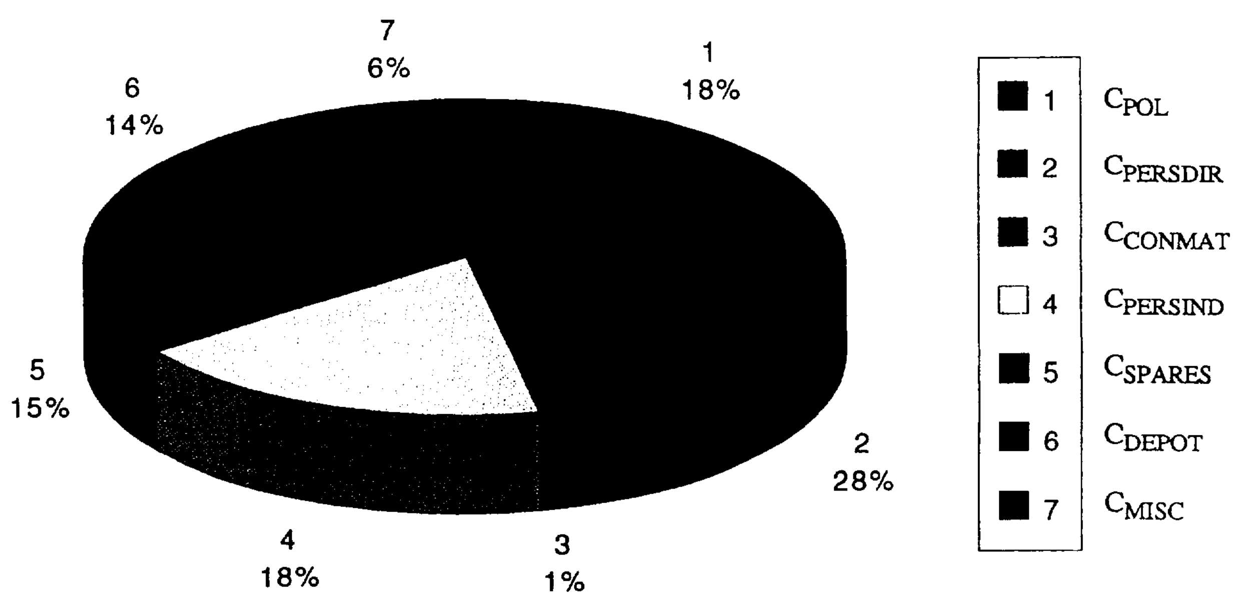
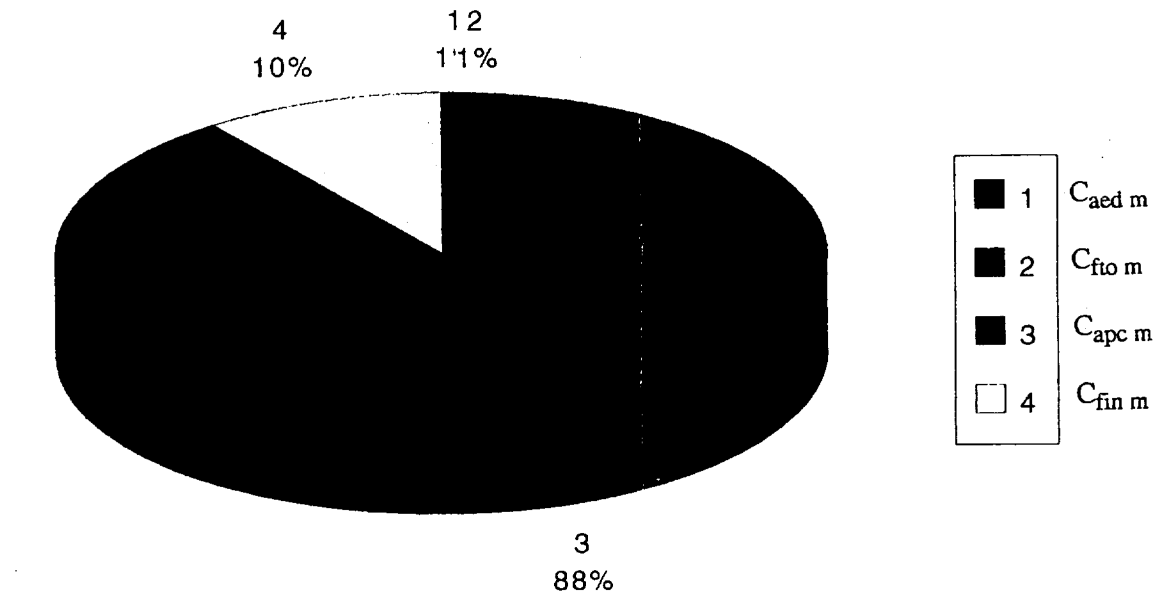
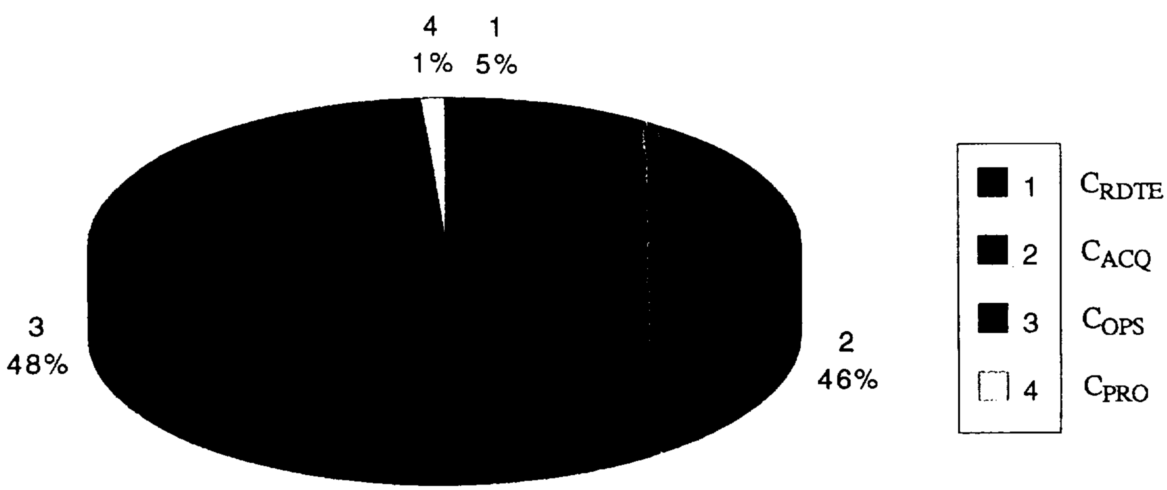


Figure 7.2.7: Cost Brea

The Acquisition Cost Breakdown



The Life Cycle Cost Breakdown



:down for the Condor

7.2.6 The Cost Comparison

The unit price of the *Condor* was estimated to be 58.3 million USD and the trend of increasing cost of tactical aircraft is shown in Fig 7.2.8. As can be seen in the figure, the cost of the *Condor* is relatively lower than the cost of other airplanes. This low cost of the *Condor* will help the customer make a decision to put this airplane in the production.

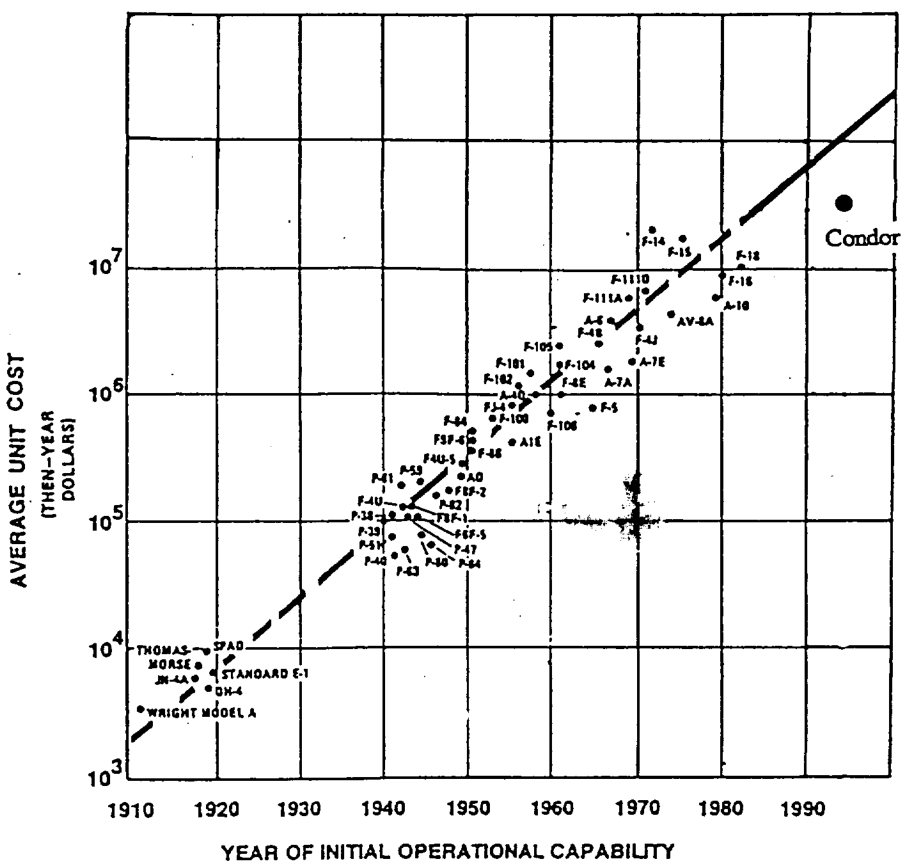


Figure 7.2.8: Trend of Increasing Cost of Tactical Aircraft
(Copied form Ref. 18)

7.3 Environmental Impact

Further design and production of the *Condor* must take into consideration recent public awareness of the environment. Careful planning must be involved in the development of this program to use materials, construction and maintenance techniques in compliance with emerging environmental and hazmat regulations. In addition, some foresight is required in the eventual disposal of the aircraft. All of these considerations must be studied in further detail and applied to the *Condor* program while observing their cost effectiveness.

As mentioned in Section 5.1, most of the airframe construction of the *Condor* is consistent with today's conventional manufacturing methods. Most of the fuselage and tail boom structure is comprised of aluminum. Although this design decision to use aluminum as opposed to composite materials may have weight penalties, the aluminum structure is 100% recyclable. Unlike the fuselage, the wing and empennage structures do contain composite materials for reasons describe in Section 3.3. Unfortunately, to this design group's collective knowledge, no environmentally acceptable methods are available today to adequately dispose of these composite materials. Further research will have to be conducted in the development of such a process.

Maintenance items included in the operation of the *Condor*, such as fuel, oil, hydraulic fluids, de-icing fluids, etc., must also be handled in an environmentally friendly manor. Biodegradable materials should be used, if available. In the case involving non biodegradable or hazardous materials, appropriate collection and disposal methods must be adopted. Unfortunately, the design team for the *Condor* has incorporated fuel dumping capabilities in an attempt to save the aircraft and crew. The fuel will be environmentally damaging when dumped.

8. Description of Automated Design Tools

The design and analysis of the *MPS-2000 Condor* as it is described in this report is largely accomplished with the aid of two automated design tools:

- *Advanced Aircraft Analysis (AAA)*
- *Aircraft Computer Aided Design (ACAD)*

These tools are described in further detail in Sections 8.1 and 8.2 respectively. A third tool supplied by AIAA, *Engine Maker*, was not utilized by this design team for the *Condor*.

8.1 Advanced Aircraft Analysis (AAA)

The Advanced Aircraft Analysis is a user-friendly program operating from the UNIX domain to be used by engineers and students to rapidly develop a preliminary aircraft configuration for early weight sizing through open-loop and closed loop dynamic stability and sensitivity analysis. The complete aircraft analysis can be conducted within regulatory and cost constraints built into the software package for civil, military and commercial fixed-wing aircraft. The program contains the following modules for preliminary aircraft design and development:

- Weight Sizing
- Performance Sizing
- Geometry
- High Lift
- Drag Polar
- Stability and Control
- Weight and Balance
- Installed Thrust
- Performance Analysis
- Stability and Control Derivatives
- Dynamics
- Control
- Cost Analysis

The AAA program is based on methods found in Dr. Roskam's Airplane Design, Parts I to VIII as found in Section 9. the AAA Program is licensed to the University of Kansas through Design, Analysis and Research Corporation in Lawrence, Kansas.

8.2 Aircraft Computer Aided Design (ACAD)

ACAD is a computer aided drafting tool developed by General Dynamics, Fort Worth Division. The program is capable of producing scaled and dimensional drawings with special features included for aircraft design. Three dimensional models are also possible with ACAD program for wetted area computation, projected area computations, and volume computations. The program is licensed to the University of Kansas.

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