Eagle RTS:
A Design for a Regional Transport Aircraft

A design project by students in the Department of Aerospace Engineering at Auburn University, Auburn, Alabama, under the sponsorship of NASA/USRA Advanced Design Program

Auburn University
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Abstract

The Eagle RTS (Regional Transport System) is a 66 passenger, twin turboprop aircraft with a range of 836 nautical miles. It will operate with a crew of two pilots and two flight attendants. This aircraft will employ the use of aluminum alloys and composite materials to reduce the aircraft weight and increase aerodynamic efficiency. The Eagle RTS will use narrow body aerodynamics with a canard configuration to improve performance. Leading edge technology will be used in the cockpit to improve flight handling and safety.

The Eagle RTS propulsion system will consist of two turboprop engines with a total thrust of approximately 6300 pounds, 3150 pounds thrust per engine, for the cruise configuration. The engines will be mounted on the aft section of the aircraft to increase passenger safety in the event of a propeller failure. Aft mounted engines will also increase the overall efficiency of the aircraft by reducing the aircraft’s drag.

The Eagle RTS is projected to have a takeoff distance of approximately 4700 feet and a landing distance of 6100 feet. These distances will allow the Eagle RTS to land at the relatively short runways of regional airports.
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<td>ft</td>
</tr>
<tr>
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</tr>
<tr>
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<td>Moment due to angle of attack</td>
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</tr>
<tr>
<td>CMαf</td>
<td>Moment due to rate of change of angle of attack</td>
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<tr>
<td>CMq</td>
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<tr>
<td>CL</td>
<td>Coefficient of lift</td>
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<td>CLu</td>
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<tr>
<td>CLα</td>
<td>Coefficient of lift due to angle of attack</td>
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</tr>
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<td>CLat</td>
<td>Coefficient of lift due to angle of attack of tail</td>
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<td>CLt</td>
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</tr>
<tr>
<td>CLβ</td>
<td>Coefficient of lift due to sideslip angle</td>
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</tr>
<tr>
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</tr>
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<td>hrs</td>
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</tr>
<tr>
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<td>degrees</td>
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<td>degrees</td>
</tr>
<tr>
<td>i₁</td>
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<td>degrees</td>
</tr>
<tr>
<td>Iₚ</td>
<td>Moment of inertia about y-axis</td>
<td>lb-s/ft⁴</td>
</tr>
<tr>
<td>L</td>
<td>Length of the wings relative to centerline</td>
<td>ft</td>
</tr>
<tr>
<td>Mₙ,Mₚ,Mₚₚ</td>
<td>Mₚₙ,Mₚₚₚ,Mₚₙₚₚ</td>
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</tr>
<tr>
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</tr>
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</tr>
<tr>
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</tr>
<tr>
<td>ω</td>
<td>Frequency</td>
<td>rad/s</td>
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<tr>
<td>φ</td>
<td>Turn angle</td>
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</tr>
<tr>
<td>R</td>
<td>Range</td>
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</tr>
<tr>
<td>R</td>
<td>Radius of turn</td>
<td>ft</td>
</tr>
<tr>
<td>RC</td>
<td>Rate of climb</td>
<td>ft/min</td>
</tr>
<tr>
<td>RD</td>
<td>Rate of descent</td>
<td>ft/min</td>
</tr>
<tr>
<td>ρ</td>
<td>Atmospheric density</td>
<td>lb-s²/ft⁴</td>
</tr>
<tr>
<td>q</td>
<td>Dynamic pressure</td>
<td>lb/ft²</td>
</tr>
<tr>
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<td>Units</td>
</tr>
<tr>
<td>----------</td>
<td>-----------------------------------------------------------</td>
<td>---------</td>
</tr>
<tr>
<td>S</td>
<td>Reference area</td>
<td>ft²</td>
</tr>
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<td>S50</td>
<td>Distance from approach to contact</td>
<td>ft</td>
</tr>
<tr>
<td>SG</td>
<td>Distance of ground roll</td>
<td>ft</td>
</tr>
<tr>
<td>ST</td>
<td>Total distance</td>
<td>ft</td>
</tr>
<tr>
<td>Ψ</td>
<td>Turn rate</td>
<td>deg/sec</td>
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<td>Ts</td>
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<td>THPreq</td>
<td>Thrust horsepower required</td>
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<tr>
<td>THPav</td>
<td>Thrust horsepower available</td>
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</tr>
<tr>
<td>V</td>
<td>Velocity</td>
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<td>Velocity at point of contact</td>
<td>ft/s</td>
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<tr>
<td>VB</td>
<td>Velocity at point of braking</td>
<td>ft/s</td>
</tr>
<tr>
<td>VCL</td>
<td>Velocity at climb</td>
<td>ft/s</td>
</tr>
<tr>
<td>V50</td>
<td>Velocity at h=50 ft obstacle</td>
<td>ft/s</td>
</tr>
<tr>
<td>VTO</td>
<td>Velocity at takeoff</td>
<td>ft/s</td>
</tr>
<tr>
<td>VH</td>
<td>Horizontal tail volume</td>
<td>ft³</td>
</tr>
<tr>
<td>W</td>
<td>Weight</td>
<td>lb</td>
</tr>
<tr>
<td>Xcg</td>
<td>Location of center of gravity</td>
<td>ft</td>
</tr>
<tr>
<td>Xac</td>
<td>Location of aerodynamic center</td>
<td>ft</td>
</tr>
<tr>
<td>Xnp</td>
<td>Location of neutral point</td>
<td>ft</td>
</tr>
<tr>
<td>Xu, Xw, Xw, Xq</td>
<td>X-Force coefficients</td>
<td>-</td>
</tr>
<tr>
<td>Zu, Zw, Zw, Zq</td>
<td>Z-Force coefficients</td>
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Introduction

The Eagle RTS (Regional Transport System) is a 66 passenger aircraft designed to satisfy the need for accessible and economic travel. The primary function of this aircraft is to provide small and medium sized cities with a quality air transportation service. The need for this regional aircraft stems from the hub airport congestion. This service will allow a passenger to travel from one spoke city to another spoke city without entering the congested hub city airport. It also allows those people traveling longer routes to begin the flight at home instead of traveling by automobile to a hub airport.

The number one design objective for the Eagle RTS will be safety. This aircraft will be safer because it avoids the hub air traffic congestion. Another safety consideration involved in the design is the anti-stall characteristic of the aircraft due to tailoring of the canard. The location of the propulsion system is also a factor in safety. The propulsion system on the Eagle RTS is placed on the aft section of the aircraft so that in the event that a blade is shed, it will not affect the passenger compartments or the major control surfaces. These safety considerations will make this aircraft a safer flight vehicle than most aircraft today.

In trying to provide the most economical and commercial flight system available, the Eagle RTS design team plans to employ the use of existing technology which will lower production and maintenance costs. This practice will reduce labor and crew costs by decreasing the amount of new training required. In selecting the propulsion system, the effects of the environment were also considered. Two advantages of turbo-prop engines are the high fuel efficiency and low noise levels produced by this type of engine.

In order for the Eagle RTS to fly spoke-to-spoke, it must be capable of landing on shorter runways. It also must have speeds comparable to that of the larger aircraft to make its service beneficial to the airlines. The Eagle RTS will cruise at 260 knots at an altitude of 25,000 feet. The aforementioned factors of safety, speed, comfort, and airport flexibility will make the Eagle RTS economically competitive in the commercial aircraft market.
I. Aerodynamics

The body shape is an elongated “teardrop” shape with pusher engines located behind the sweptback wings. This configuration will allow for minimum body drag while allowing for maximum flexibility in designing the interior arrangement. Figure 1.1 provides a three-view and Figures 1.2-1.4 provide the side, top and front views of the Eagle RTS.

The airfoil selected for the Eagle RTS is the NACA 632-615 series airfoil. This airfoil was selected because it had the most efficient cruise characteristics. The NACA 632-615 airfoil has a high stall angle of attack. Also, as this airfoil approaches the stall angle it goes into a ‘soft’ stall as opposed to an abrupt stall. According to Daniel P. Raymer the recommended wing thickness ratio for twin turbo prop aircraft is 0.14 (1).

The Eagle RTS uses a compound wing design shown in Figure 1.5. The sweep angles for this wing are 9° and 60°. These angles were chosen to provide a wing area which produced a maximum lift coefficient and a minimum wing loading while also providing excess fuel tank storage. The wing loading is calculated to be 70 lb/ft². Figure 1.6 provides a plot of the lift coefficient versus the angle of attack. (2)

The drag polar was calculated using Roskam’s Methods for Estimating Drag Polars of Subsonic Airplanes (3) which was done through the use of a FORTRAN language program as seen in Appendix A.

\[ C_D^{\text{cruise}} = 0.0615 \]

Figure 1.7 provides a plot of the drag coefficient versus the lift coefficient.

The Eagle RTS will employ the use of a canard. The purpose of this canard is to prevent stall characteristics such as spin and uncontrolled roll. The canard airfoil selected for the Eagle RTS is the NACA 0009 series. A detailed dimensional layout of this canard may be seen in Figure 1.8. The main wing will cruise at an angle of attack of 1°. Also the main wing has a zero lift angle of attack of -5° and a stall angle of attack of 12°. To choose the proper canard to prevent aircraft
Figure 1.1 Three View of the Eagle RTS
Figure 1.2 Side View of the Eagle RTS
Figure 1.3 Top View of the Eagle RTS
Figure 1.4 Front and Rear View of the Eagle RTS
Figure 1.5 Compound Wing Design

Eagle RTS
Wing Reference Area
Reference Area = 986.68 sq. ft.
- only half total area shown

Edge of Body Skin
Figure 1.6 Aircraft Lift Coefficient vs. Angle of Attack
Figure 1.7 Lift Coefficient vs. Drag Coefficient
BODY EDGE - PIVOT

- only half of area shown

Platform Area = 62.734 sq. ft.

3'-11.8" Canard Detail

Eagle RTS
stall this canard must have a stall angle below that of the main wing. The canard for the Eagle RTS will cruise at an angle of attack of 2° while stalling at an angle of attack of 9° plus or minus 1°. Because the canard will stall at 9° the main wing will never reach its stall angle of attack of 12° (4).

A secondary advantage of the canard for the Eagle RTS is that the canard will eliminate the negative lift normally associated with a tailplane configuration. Also, a canard has a more rapid response time to control input than a tailplane configuration. One disadvantage of a canard is the effect of trailing vortices on the main wing aerodynamics and the engine efficiency. Although these actual effects are still being researched, according to Daniel P. Raymer the most efficient way to minimize these effects is to place both the main wing and engines as far aft and above the canard as possible; which has been done for the Eagle RTS (1).

The tail section for this aircraft uses a vertical tail configuration, NACA 0009 series, which will provide the Eagle RTS with directional stability. The design of this tail uses an area determined from Equation 1.1.

\[ S_{VT} = c_{vt}b_wS_w / L_v \] (1.1)

This equation, provided by Raymer, uses a constant \( c_{vt} \) of 0.08 for twin turboprop aircraft, wing span and wing reference area divided by the distance from the vertical tail to the mean aerodynamic center \( L_v \) to determine the vertical tail area (1). Figure 1.9 provides a detailed dimensional layout of the vertical tail.

Another important factor in the design is the efficiency of this aircraft, also known as Oswald's efficiency factor. This factor depends on the aspect ratio which is calculated using Equation 1.2,

\[ AR = b^2 / S \] (1.2)

where the aspect ratio is a function of the square of the wing span divided by the reference area. The aspect ratio for the Eagle RTS is 6.5. From this aspect ratio and a parasite drag \( C_{D_p} \) of 0.032
Eagle RTS
Vertical Tail

Figure 1.9 Vertical Tail Configuration
the airplane efficiency factor, from Figure 1.10, is found to be 0.775. According to Richard S. Shevell an efficient aircraft operates between an Oswald's efficiency factor of 0.75 and 0.9 (5).

![Figure 1.10 Airplane Efficiency Factor](image)

Figure 1.10 Airplane Efficiency Factor
II. Performance Analysis

The first and probably most important consideration in conducting the performance analysis is to determine the thrust needed. Reference material is from references 5, 9 and 11. All values used in performance calculations are found in Tables 2.1 and 2.2 and the equations used are:

\[
\text{THP}_{\text{req}} = \frac{(\text{Drag})(\text{Velocity})}{550 \text{ ft-lb/s}^2}
\]

\[
\text{THP}_{\text{av}} = \frac{(\text{Thrust})(\text{Velocity})}{550 \text{ ft-lb/s}^2}
\]

These equations calculate horsepower required and available for the aircraft.

Another important factor in aircraft performance is the rate of climb. To determine the rate of climb in feet per minute:

\[
\text{RC} = \frac{(\text{THP}_{\text{av}} - \text{THP}_{\text{req}}) \times 33000}{W} \quad (2.2)
\]

The rate-of-climb versus velocity is shown in Figure 2.1. Our rate-of-climb at cruise velocity and an altitude of 25000 ft and full passenger and fuel load is 928 ft/min which is not bad for an airplane our size. The opposite consideration to the rate-of-climb is the rate-of-descent. It follows along with the logic of the climb rate and is mainly determined by the flight path angle (\(\gamma\)):

\[
\sin \gamma = \frac{\text{Drag}}{\text{Weight}}
\]

\[
\text{RD} = V\sin\gamma = \left[\frac{(2WC_{D}^2/C_{L}^2\cos^2\gamma)/\rho S}{0.5}\right]
\]

(2.3)
Figure 2.1: Rate of Climb vs. Velocity
Table 2.1: Performance Parameters at Cruise Velocity

<table>
<thead>
<tr>
<th>Airplane Characteristics:</th>
<th></th>
<th></th>
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<tbody>
<tr>
<td>( W_{oper} = 58688 , \text{lb} )</td>
<td>( V_H = 0.00853 )</td>
<td>( c = 14.9 , \text{ft} )</td>
</tr>
<tr>
<td>( S = 986.68 , \text{ft}^2 )</td>
<td>altitude = 25000 ft</td>
<td>( V_{cruise} = 260.5 , \text{knots} )</td>
</tr>
<tr>
<td>( b = 80 , \text{ft} )</td>
<td>( \text{THP}_{cruise} = 6300 , \text{hp} )</td>
<td>( W_{indg} = 46408 , \text{lb} )</td>
</tr>
<tr>
<td>( C_D = 0.062 )</td>
<td>( C_L = 0.386 )</td>
<td></td>
</tr>
</tbody>
</table>

- \( \text{THP}_{req} = 3413.78 \, \text{hp} \)
- \( \text{RC} = 928 \, \text{ft/min} \)
- \( \text{Range} = 836.0 \, \text{nmi} \)
- \( S_{TO} = 4696.4952 \, \text{ft} \)
- \( n = 1.01223 \)
- \( \psi = 0.66 \, ^\circ/\text{s} \)
- \( V_{flight\ path} = 441.3 \, \text{ft/s} \)
- \( \text{THP}_{av} = 5355.0 \, \text{hp} \)
- \( \text{RD} = 2135.7 \, \text{ft/min} \)
- \( \text{Endurance} = 2.55 \, \text{hrs} \)
- \( S_{T_{indg}} = 6100.0 \, \text{ft} \)
- \( \phi = 8.92^\circ \)
- \( \text{Radius} = 38307 \, \text{ft} \)
Table 2.2 Performance Calculations and Constants

<table>
<thead>
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<tr>
<td>$L_{\text{max}}$=69887.2 lb</td>
<td>$D_{\text{TO}}$=10579.4 lb</td>
</tr>
<tr>
<td>$(C_{L}/C_{D})_{\text{max}}$=9.2124</td>
<td>$(C_{L}/C_{D})_{\text{max}}^{3/2}$=15.5344</td>
</tr>
<tr>
<td>$T_{\text{TO}}$=18438.04 lb</td>
<td>$W$=58688 lb</td>
</tr>
<tr>
<td>$V_{\text{TO}}$=186.05 ft/s</td>
<td>$V_{CL}=202.96$ ft/s</td>
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<tr>
<td>$V_C=194.5$ ft/s</td>
<td>$V_B=155.6$ ft/s</td>
</tr>
<tr>
<td>$\mu_b=0.3$</td>
<td>$C_{L_{\text{max}}}=3.9$</td>
</tr>
<tr>
<td>$S=986.68$ ft$^2$</td>
<td>$T_s=12617.34$ lb</td>
</tr>
<tr>
<td>$\mu=0.02$</td>
<td>$V_{50}=219.87$ ft/s</td>
</tr>
<tr>
<td>$W_{\text{Indg}}=46408$ lb</td>
<td>$V_{\text{stall}}=100.15$ knots</td>
</tr>
</tbody>
</table>

Takeoff:

$F_S=T_S - \mu W = 11443.58$ lb

$F_{\text{TO}}=T_{\text{TO}} - D_{\text{TO}} - \mu(W-L_{\text{TO}}) = 6684.88$ lb

Landing:

$F_S=T + \mu_b W = 11443.58$

$F_C=T + D_{V_c} + \mu(W-L) = 13740.302$

$F_{B1}=T + D_{V_b} + \mu(W-L)= 13387.112$

$F_{B2}=T + D_{V_b} + \mu_b(W-L_{V_b})= 25432.38$
In the initial analysis it was estimated that the range would be 1000 nmi. The range depends on the propeller efficiency, the specific fuel consumption, lift/drag, and weight. To find the maximum range, we use a maximum lift to drag ratio. The equation that is used to calculate range in nautical miles is:

\[
R = 325(\eta/c) (L/D) \ln(W_i/W_f)
\]  

(2.4)

Using an efficiency of 0.8 and an SFC of 0.547 lb/hr-HP the range turns out to be 836 nmi. This is below what was specified at the beginning of the design process. However, the range of this airplane will enable it to fly reasonably long distances and is deemed to be adequate.

Takeoff & Landing Performance

The FAR regulations governing the takeoff and landing performance of commercial aircraft is listed below:

<table>
<thead>
<tr>
<th>Velocity:</th>
<th>( V_{TO} \geq 1.1 \ V_{Stall} )</th>
<th>( V_{50} = 1.3 \ V_{Stall} )</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>( V_{CL} \geq 1.2 \ V_{Stall} )</td>
<td>( V_C = 1.15 \ V_{Stall} )</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Climb:</th>
<th>Gear down 1/2% ( V_{TO} )</th>
<th>Gear up 3% ( V_{TO} )</th>
</tr>
</thead>
</table>

| Field Length: | 115\% takeoff distance over 35 ft obstacle |

Table 2.3
Using the equations:

\[ S_1 = \frac{W V_{to2}}{(2g F_s - F_{to})} \ln \left( \frac{F_s}{F_{to}} \right) \]

\[ S_2 = \frac{W}{2g} \left( \frac{V_{cl2} - V_{to2}}{T_{to} - D_{to}} \right) \]

\[ S_3 = H \sqrt{\left( \frac{W}{T_{to} - D_{to}} \right)^2 - 1} \]  

(2.5)

it was found that the total takeoff distance required was 4700 ft. See Figure 2.2 for a breakdown of takeoff and landing distances. Since this airplane is designed to operate from smaller airports as well as larger ones, the landing and takeoff distances must be reasonably short to handle the average runway. To calculate the landing distance required we use the equation below with values of forces and other parameters found in Table 2.1. First start with the distance required to clear a fifty foot obstacle on approach to the contact point on the ground. This is found by:

\[ S_{50} = \frac{(L/D)(V_{502} - V_C^2)}{2g} + 50 \]  

(2.6)

By using equation (2.6) in conjunction with this next equation which is ground roll distance measured from the contact point to the braking point and zero velocity (stopping) point:
Takeoff Parameters:
$V_{TO}=186.05$ ft/s
$S_1=3563.64$ ft
$S_{TO}=4696.592$ ft

$Landing Parameters:
$V_{SO}=219.87$ ft/s
$S_{SO}=1964.44$ ft
$S_{TL}=3684.063$ ft

Figure 2.2 Takeoff and Landing Breakdown
\[
S_G = \frac{W}{2g} \left\{ \left( V_{C}^2 - V_{B}^2 \right) / (F_{B1} - F_C) \ln(F_{B1}/F_C) \\
+ V_{B}^2 / (F_S - F_{B2}) \ln(F_S/F_{B2}) \right\} 
\]

(2.7)

Add these two together to get the total landing distance.

\[
S_T = S_50 + S_G 
\]

(2.8)

Our total landing distance is 3700 ft which is roughly two thirds our takeoff distance. Using FAR requirements for a commercial aircraft the landing distance becomes 6100 ft. With the addition of ground spoilers this landing distance will decrease. This is well within limits of the runway requirements for our airplane.
III. Stability Analysis

Longitudinal Stability:

In conducting a stability analysis, several features of this field were investigated to determine if the airplane was stable or not. This includes a comparison of pitching moment versus angle of attack, neutral point location and the stability margin. When speaking of stability, it is in reference to longitudinal stability where the airplane has a restoring moment about the center of gravity when disturbed from its equilibrium position. This is normally described by the term $C_{m\alpha}$.

For static stability the $C_{m\alpha}$ must be negative in order to bring the plane back to equilibrium after a disturbance. Each surface on the airplane has a contribution to $C_m$. The surface with the most impact is the wings. The equation for finding its total value is

$$C_{m\alpha} = C_{L\alpha w}(X_{cg/c} - X_{ac/c}) \quad (3.1)$$

In order to find this value of $C_{m\alpha}$ and subsequently the variation of it with angle of attack, the neutral point or static margin and center of gravity must be found. Table 3.1 shows the component weights and locations to find the center of gravity location for various loading conditions.

All methods in this analysis are from references 5, 9 and 11. Table 3.2 lists all the characteristics pertinent to this analysis plus any values found in other sections of this report. The first quantity needed to find is the neutral point of the aircraft. This would tell how much the c.g. could move and still keep the plane statically stable. This is important in commercial aviation because of changing configurations, passenger and baggage loadings. Movement beyond the neutral point causes the airplane to be statically unstable. We would like to solve the equation:

$$X_{np/c} = X_{ac/c} - C_{m\alpha}/C_{L\alpha w} - \frac{\eta}{H}C_{L\alpha w}/C_L \alpha_w(1 - \frac{de}{d\alpha}) \quad (3.2)$$

First of all we can neglect the $(de/d\alpha)$ term (change in downwash angle due to angle of attack).
Table 3.1: C.G. Location for Various Loadings

C.G Formulas:

\[ x = \frac{\sum m_i x_i}{\sum m_i} \quad y = \frac{\sum m_i y_i}{\sum m_i} = 0.0 \quad z = \frac{\sum m_i z_i}{\sum m_i} = 1.2092 \text{ ft} \]

Component Weights:

- \( M_1 \) = fixed equipment + fuselage = 11014.0 + 8204.0 = 19218 lb
- \( M_w \) = wing mass = 8540.0 lb
- \( M_L \) = landing gear mass = 3190.0 lb
- \( M_c \) = empennage mass (including tail mass) = 1899.0 lb
- \( M_E \) = engine mass = 6304.0 lb
- \( M_N \) = nacelles mass = 1823.0 lb

Total Component Weight = 40974.0 lb
(Table 3.1 continued):

C.G. Location for Various Loading Conditions:
Average weight per person= 1701lb x 66 passengers = 11220 lb
Baggage average weight = 3130 lb
Fuel average weight = 14280

\[ X_{CG} = 50.7509 \text{ ft from the nose} \quad \text{Static margin}= 12.39 \text{ ft} \]

0.75 passenger and baggage loading at 30 feet from the nose
0.50 fuel weight = 7140 lb

\[ X_{CG} = 51.5363 \text{ ft} \quad \text{Static margin}= 11.604 \text{ ft} \]

0.00 passenger and baggage loading
approximately zero fuel
( empty landing weight)

\[ X_{CG} = 56.865 \text{ ft} \quad \text{Static margin}= 6.28 \text{ ft} \]
Secondly, the tail efficiency term (η) can be equal to unity since it depends on the position of the tail surface in the wake of the fuselage and the wings. Since we have canard surfaces, no wake influence is seen. Lastly, Cmαf, pitching moment due to angle of attack on the fuselage, is given as an estimate since it depends on average fuselage section widths and upwash angles which cannot be determined from our basic analysis. With this in mind we find:

\[ C_{L\alpha f} = \frac{\pi AR_f}{2} = \frac{\pi b^2}{2 S_f} \]
\[ C_{L\alpha w} = \frac{\pi AR_w}{2} = \frac{\pi b^2}{2 S_w} \]  

Then the volume ratio may be determined.

\[ V_H = l_i S_i / S_c \]  

From here it is a matter of putting the values into Equation (3.2). The neutral point was found to be at \( X_{NP} = 15.892 \) ft forward from the trailing edge of the wings or 58.1 ft aft of the nose and the aerodynamic center at \( X_{ac} = 63.14 \) ft from the nose. The center of gravity is located at \( X_{cg} = 50.7509 \) ft aft from the nose for a fully loaded aircraft.

Using the newly found neutral point we can determine the static margin by using the relation:

\[ (X_{ac}/c - X_{cg}/c) = (h-hn) \]  
\[ (h-hn) = 12.39 \text{ ft} \]  

Finally the variation of Cm with angle of attack for various deflections of the canard may be found. Starting with the equation for lift of the tail section plus finding the reference pitching moment and using the already found static margin:

\[ C_{Li} = C_{L\alpha} V_H (l_i + e_o) \]  

\[ (3.7) \]
Figure 3.1 Pitching Moment versus Alpha
\[ C_{m_{\alpha}} = C_L (h-hn) - V_H C_L \tag{3.8} \]
\[ C_m = C_{m_0} + C_{m_{\alpha}} \alpha \tag{3.9} \]

This variation can be seen on Figure 3.1 for various control deflections. Mentioning the significance of the canard surface in relation to our design selection, it is free from propulsive interference and thus is better to trim the large moment produced by high lift devices such as our wings. But it also adds to a destabilizing effect on the airplane. This can be counteracted by proper positioning of the center of gravity.

**Roll Stability:**

Similar to longitudinal stability, the rolling stability can be achieved when a restoring moment is present when the wings are subjected to disturbing forces. The largest contribution to the rolling stability is the dihedral effect and is designated as $C_{lB}$. When it is less than zero, the aircraft possess static rolling stability. Relating to our aircraft, it was found that the aircraft was unstable largely due to our wings located underneath the fuselage. So by giving the wings a dihedral angle ($\Gamma = 7^\circ$) we attain better rolling stability. In addition, the wings are swept back considerably which gives rise to an increase in the aforementioned dihedral effect.

**Dynamic Stability:**

The main reason for conducting a dynamic stability analysis is to quantify the aircraft movement in flight; i.e. flight dynamics. In the commercial aviation market, passenger comfort is a prime consideration in customer satisfaction. Also, pilots prefer to fly aircraft that possess good flight characteristics so they do not have to 'fight' the airplane in flight or increase the time necessary to trim flight perturbations. Table 3.1 gives a list of all the longitudinal derivatives found plus the phugoid and short period approximations. Table 3.2 is a list of the equations used to determine the approximate solutions for aircraft dynamic motion.
Table 3.2: Dynamic Stability Derivatives

**Initial Conditions:**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$u_0$</td>
<td>440 ft/s</td>
</tr>
<tr>
<td>$W$</td>
<td>69045 lb</td>
</tr>
<tr>
<td>$S$</td>
<td>986.68 ft$^2$</td>
</tr>
<tr>
<td>$b$</td>
<td>80 ft</td>
</tr>
<tr>
<td>$c$</td>
<td>14.9 ft</td>
</tr>
<tr>
<td>$CD$</td>
<td>0.06</td>
</tr>
<tr>
<td>$CL$</td>
<td>0.386</td>
</tr>
<tr>
<td>$CD_a$</td>
<td>0.0</td>
</tr>
<tr>
<td>$CL_a$</td>
<td>0.0</td>
</tr>
<tr>
<td>$Coa$</td>
<td>0.3</td>
</tr>
<tr>
<td>$CM_a$</td>
<td>10.48</td>
</tr>
<tr>
<td>$CMq$</td>
<td>-0.33536</td>
</tr>
<tr>
<td>$I_y$</td>
<td>103.22 lb$^2$</td>
</tr>
<tr>
<td>$q$</td>
<td>103.22 lb/ft$^2$</td>
</tr>
<tr>
<td>$qs$</td>
<td>101843 lb</td>
</tr>
<tr>
<td>$qsc$</td>
<td>1517460.7 lb-ft</td>
</tr>
<tr>
<td>$c/2u_0$</td>
<td>0.01693 s</td>
</tr>
<tr>
<td>$u_om$</td>
<td>27569668 lb-ft/s</td>
</tr>
<tr>
<td>$u_oly$</td>
<td>3.993e8 slug-ft$^3$/s</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$X_u$</td>
<td>$-0.000222$ s$^{-1}$</td>
</tr>
<tr>
<td>$X_w$</td>
<td>$0.000318$ s$^{-1}$</td>
</tr>
<tr>
<td>$X_q$</td>
<td>0.0</td>
</tr>
<tr>
<td>$Z_u$</td>
<td>$-0.00285$ s$^{-1}$</td>
</tr>
<tr>
<td>$Z_w$</td>
<td>$0.00154$ s$^{-1}$</td>
</tr>
<tr>
<td>$Z_q$</td>
<td>0.0</td>
</tr>
<tr>
<td>$M_u$</td>
<td>0.0</td>
</tr>
<tr>
<td>$M_w$</td>
<td>$-0.001274$ ft$^{-1}$s$^{-1}$</td>
</tr>
<tr>
<td>$M_q$</td>
<td>$-0.000288$ s$^{-1}$</td>
</tr>
</tbody>
</table>

**Phugoid Approximations:**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\omega_{np}$</td>
<td>0.01516 rad/s</td>
</tr>
<tr>
<td>$\lambda_{1,2}$</td>
<td>$-0.000111 \pm 0.015159i$</td>
</tr>
<tr>
<td>Period</td>
<td>414.46 s = 6.91 min</td>
</tr>
<tr>
<td>$t_{1/2}$</td>
<td>6216.22 s = 103 min</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\xi_p$</td>
<td>0.007322</td>
</tr>
</tbody>
</table>

**Short-Period Approximations:**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$Z_\alpha$</td>
<td>$-0.6145$</td>
</tr>
<tr>
<td>$M_\alpha$</td>
<td>$-0.5083$</td>
</tr>
<tr>
<td>$\omega_{nsp}$</td>
<td>0.71295 rad/s</td>
</tr>
<tr>
<td>$\lambda_{1,2sp}$</td>
<td>$-0.00476 \pm 0.71293i$</td>
</tr>
<tr>
<td>Period</td>
<td>8.132 s = 0.147 min</td>
</tr>
<tr>
<td>$t_{1/2}$</td>
<td>144.96 s = 2.42 min</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\xi_{nsp}$</td>
<td>$-0.006682$</td>
</tr>
</tbody>
</table>
Table 3.3: Stability Derivative Approximations

<table>
<thead>
<tr>
<th>Phugoid:</th>
<th>Short-Period:</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\omega_{np} = (-Z_u g/u_o)^{1/2}$</td>
<td>$\omega_{nsp} = (Z_a M_q/u_o - M_a)^{1/2}$</td>
</tr>
<tr>
<td>$\lambda_{1,2} = -\xi_p \omega_{np} \pm i \omega_n (1 - \xi^2)$</td>
<td>$\lambda_{1,2} = -\xi_p \omega_{nsp} \pm i \omega_n (1 - \xi^2)$</td>
</tr>
<tr>
<td>$t_{1/2} = 0.69/\eta$</td>
<td>$t_{1/2} = 0.69/\eta$</td>
</tr>
<tr>
<td>$\xi_p = -X_u/2\omega_{np}$</td>
<td>$\xi_{sp} = (M_q + M_\alpha + Z_\alpha/u_o)/2 \omega_{nsp}$</td>
</tr>
<tr>
<td>Period = $2\pi/\omega_n$</td>
<td>Period = $2\pi/\omega_{nsp}$</td>
</tr>
</tbody>
</table>

The phugoid or long period mode is characterized by changes in pitch, altitude and velocity. In this analysis, if the aircraft was disturbed from its equilibrium, the period of the phugoid motion was found to be 6.91 minutes and the time to half amplitude at 103 minutes. The short-period characteristics were more tolerable with a period at 0.147 minutes and a time to half amplitude equal to 2.42 minutes. To correct the large phugoid oscillations, the lift to drag ratio would have to be reduced, thus decreasing the range. The other alternative is to control the motion by our automatic stabilization computer which would in this case be the preferable choice. As for the short-period characteristics, they are very important as the performance of the airplane is directly related to the frequency of the short-period motion. If the motion is not sufficiently damped or the frequency is too low, the aircraft may become uncontrollable. Although this aircraft is balanced and stable, according to calculations the frequency and damping is indeed too low and must be augmented by the automatic stabilization system.
IV. Structures and Materials

When designing the interior of the fuselage, the number of passengers is an important factor because it directly influences the exterior dimensions, the cabin dimensions, the airline profit and feasibility, and future applications of the aircraft. Therefore, the seating arrangement should be chosen to reflect the needs of the passengers. Passengers want to fly the most economical, comfortable, and safe aircraft available. Based on these factors, the Eagle RTS will accommodate 66 passengers, with a seating configuration of four seats abreast in two rows (See Figure 4.1 and 4.2). With this seating configuration, a sufficient amount of space exists between the aisles and seats to allow for maximum passenger safety and comfort.

Another important factor in the design of an aircraft is the preliminary weight estimation. Weight directly determines the general configuration, aerodynamic characteristics, and cost of the aircraft. Therefore an accurate estimation of the weight is the first priority in the design of the aircraft. In order to calculate this initial weight, the mission profile of the aircraft must be determined. The Eagle RTS is a regional twin turboprop aircraft with a maximum range of 836 nautical miles capable of transporting 66 passengers. With this type aircraft, the mission profile includes eight phases: Start-up, Taxi, Takeoff, Climb, Cruise, Loiter, Descent, Landing, and Shutdown. By referencing other aircraft with similar mission profiles, an estimated gross take-off weight for the Eagle RTS is determined to be 70,000 lbs\(^{(11)}\). Gross take-off weight is the weight of the aircraft fully loaded and therefore must include the empty weight, payload weight, crew weight, and the weight of the fuel. This equation can be written,

\[
W_{\text{GTO}} = W_E + W_{\text{PAY}} + W_{\text{CREW}} + W_F
\]  

Based on industry studies, passengers on a commercial aircraft have an average weight of 175 lbs. per person and an average baggage weight of 30 lbs per person for short to medium distance flights\(^{(13)}\). For the Eagle RTS, the payload and crew weight was determined to be 14,350 lbs.

In order to calculate the weight of the fuel, the fuel-fraction method will be used. This
- The seats are spaced 41' front to front.
- The windows are 1' wide and 18' high.
Eagle RTS
Cross Section of Fuselage

Overhead Storage
4'-6"

6'-1"

Luggage Compartment
Two Inches Between Outer and Inner Skins - Insulated

3'-4"
1'-5"
method uses fuel ratios based on previously determined values and is a ratio of the end weight to the beginning weight for each phase of the mission (See Figure 4.3). For the Eagle RTS, the fuel-fraction (FF) was determined to be 0.689 including fuel reserves (See Table 4.1)

![Mission Profile for the Eagle RTS](image)

**Figure 4.3 Mission Profile for the Eagle RTS**

**Table 4.1 Fuel Fractions for the Eagle RTS**

<table>
<thead>
<tr>
<th>ID #</th>
<th>PHASE</th>
<th>FUEL FRACTION</th>
<th>RATIO</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Start-up</td>
<td>W1/Wto</td>
<td>0.990</td>
</tr>
<tr>
<td>2</td>
<td>Taxi</td>
<td>W2/W1</td>
<td>0.995</td>
</tr>
<tr>
<td>3</td>
<td>Take-off</td>
<td>W3/W2</td>
<td>0.995</td>
</tr>
<tr>
<td>4</td>
<td>Climb</td>
<td>W4/W3</td>
<td>0.980</td>
</tr>
<tr>
<td>5</td>
<td>Cruise</td>
<td>W5/W4</td>
<td>0.820</td>
</tr>
<tr>
<td>6</td>
<td>Loiter</td>
<td>W6/W5</td>
<td>0.967</td>
</tr>
<tr>
<td>7</td>
<td>Descent</td>
<td>W7/W6</td>
<td>0.990</td>
</tr>
<tr>
<td>8</td>
<td>Landing &amp; Taxi</td>
<td>W8/W7</td>
<td>0.992</td>
</tr>
<tr>
<td>9</td>
<td>Shutdown</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Fuel reserves must be included to adhere to the FAR guidelines and to provide for additional fuel in the case of an emergency. From the fuel-fraction method, the weight of the fuel can be calculated using the equation,

\[ W_F = (1 - FF)W_{GTO} = 14,280 \text{ lbs.} \] (4.2)

By using Equation 4.1, the empty weight of the aircraft is determined to be 41,020 lbs. Now that the empty weight is known, the weight of the components can be calculated. The component weight is beneficial because this weight can be used to incorporate the use of composite materials. Using the fraction method of component weights by referencing similar aircraft, the weight of the major component groups can be determined (See Table 4.2).

<table>
<thead>
<tr>
<th>Group</th>
<th>1st Calc</th>
<th>Adjustments</th>
<th>Class I</th>
<th>Composites</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>7910</td>
<td>630</td>
<td>8540</td>
<td>7259</td>
</tr>
<tr>
<td>Empennage</td>
<td>1750</td>
<td>149</td>
<td>1899</td>
<td>1614</td>
</tr>
<tr>
<td>Fuselage</td>
<td>7560</td>
<td>644</td>
<td>8204</td>
<td>6973</td>
</tr>
<tr>
<td>Nacelles</td>
<td>1680</td>
<td>143</td>
<td>1823</td>
<td>1549</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>2940</td>
<td>250</td>
<td>3190</td>
<td>2711</td>
</tr>
<tr>
<td>Power Plant</td>
<td>5810</td>
<td>494</td>
<td>6304</td>
<td>5358</td>
</tr>
<tr>
<td>Fixed Equipment</td>
<td>10150</td>
<td>864</td>
<td>11014</td>
<td>9362</td>
</tr>
<tr>
<td>Empty Weight</td>
<td>37800</td>
<td>3220</td>
<td>41020</td>
<td>34826</td>
</tr>
</tbody>
</table>
Since composites will be used in limited areas to maximize efficiency and minimize cost, the adjusted empty weight of the Eagle RTS is 40,415 lbs. Composites will be used in areas such as the leading and trailing edges, the inboard and outboard flaps, rudders, elevators, and landing gear doors. Using Equation 4.1, an accurate estimation of the final gross take-off weight is calculated to be 69,045 lbs.

The next phase of the design is to determine the materials used to construct the Eagle RTS. The material selection is based on the maximum loads applied to the aircraft during flight. By utilizing the structural analysis computer program known as MSC/NASTRAN, the resulting stresses, displacements, and forces can be obtained. To illustrate the results processed from NASTRAN, an additional computer program, MSC/XL, can be used. The main wing was modeled using these programs as a cantilever beam with a varying thickness. The wing loading was determined to be 100 lb/ft², with a 1.5 safety factor for normal cruise conditions. The structural limit for maneuvers is a load factor of 4.8 for standard rate turn. The maximum aerodynamic load factor will only be 1.013, therefore the structural limit will never be reached.

The results from MSC/NASTRAN reveal that the maximum tensile stress on the upper surface of the wing is $4.68 \times 10^2$ psi. Also, the maximum compressive stress is $1.04 \times 10^2$ psi was located on the lower surface of the wing. Also, the shear and moment diagrams reveal the maximum allowable loads on the main wing (See Figures 4.4 and 4.5). From these diagrams, main areas where the values for shear and moment are the highest are at the locations where the sweep angle increases.

The materials used for the construction of the Eagle RTS will be an integration of aluminum alloys and composites. The skin and stringers of the upper surface will be constructed of an aluminum alloy which has high tensile stresses allowances. An alloy commonly used in this area is 7075 (Al-Zn), which has an ultimate tensile stress of $72 \times 10^3$ psi and a yield stress of $64 \times 10^3$ psi. For the lower surface of the wing, the alloy 2024 (Al-Cu) will be the main material for the
Figure 4.4: Shear Distribution for the Upper Surface on the Main Wing
Figure 4.5 Moment Distribution for the Upper Surface on the Main Wing
construction in this area. This alloy exhibits a compressive yield stress of $37 \times 10^3$ psi. Based on the values determined from the NASTRAN results, the materials used are sufficient to withstand the loads applied during flight.

In order for the Eagle RTS to operate at its maximum efficiency, the weight of the aircraft must be minimized. From the empty weight calculations, the total empty weight was reduced proportionally to the amount of composite materials utilized compared to the aluminum alloys. The purpose for an integration of aluminum alloys and composites is that at the present time, composite materials are too expensive to construct an entire aircraft of this magnitude. Also, most aircraft manufacturers are tooled toward metal. Another reason for this integration is that composite materials have not yet demonstrated sufficient strengths with the type of loadings experienced during flight for the commercial industry to totally justify the use of composites in all areas of the construction.

Composites do possess some useful advantages over aluminum alloys. Composite structures demonstrate a weight savings of approximately 25% compared to metals. Another advantage is that composites can be made into any shape and maintain their physical properties. Also, composites have some useful aerodynamic characteristics. They tend to provide a smoother surface for airflow due to the lower surface roughness than metals. Originally, the composites used for the construction were to be Aramid, known commercially as Kevlar. After reviewing the ultimate stresses for both Aramid and graphite/epoxy, it was noted that Aramid had a significantly lower compressive stress than that of graphite. Therefore a combination Aramid will be the primary composite used in areas of high tension, while graphite/epoxy will be the used in areas of high compression. A fiberglass composite will be used on the nose cone and the main deck floor panels due to its lower cost and other composites.

As previously stated, aluminum alloys will be the dominant material used on the Eagle RTS. This is mainly because of its “good” strength characteristics, high corrosion resistance, availability, and low cost. While aluminum alloys weigh more than that of composites, aluminum
is the more conservative material. This fact must be considered with a commercial aircraft carrying passengers. Thus, aluminum alloys are more accepted in the commercial aircraft market.

Although based on the NASTRAN results, the stresses in the wing are significantly lower than that of the allowable stresses for the materials used on the wing. Another aluminum alloy, such as aluminum lithium, may be used. Aluminum lithium demonstrates high strength characteristics and weighs less than that of the other aluminum alloys previously stated. Another advantage of this alloy is that it can be manufactured using standard metal techniques, although the raw material cost is greater than the aluminum alloys of copper and zinc.
V. Propulsion

The propulsion system for the Eagle RTS was selected under the consideration that the aircraft is to have a cruise at 25000 feet at a speed of 260 knots (440 feet per second; Mach 0.4). At this speed a turbojet engine would be very inefficient therefore other types of propulsion systems had to be considered. The types of engines that were viable options at speeds near Mach 0.4 were turboprop and turbofan type engines. A.A. Blythe and P. Smith wrote in their American Institute of Aeronautics and Astronautics (AIAA) paper, “Block fuel savings of 25% to 27% are predicted for a Mach 0.7 derivative open rotor aircraft relative to a comparable baseline turbofan powered aircraft. Design range can be increased 45% for equal fuel capacity.” (1) Based on this paper and similar statements in other material a turboprop engine was selected for use on the Eagle RTS.

Once the engine type was selected for the Eagle RTS the engine size needed to be determined. The thrust required for the aircraft was determined from the drag acting on the aircraft in level, unaccelerated flight at the cruise altitude of 25,000 feet. The methods for determining this drag is outlined in the aerodynamics section. The drag acting on the aircraft in cruise configuration is 6300 pounds force. In level, unaccelerated flight the thrust required is equal to the drag on the aircraft. The horsepower required to produce this thrust is calculated from Equation 5.1,

\[ Hp = \frac{T \cdot V}{550 \cdot \eta_p} \]  

(5.1)

where \( T \) is thrust in pounds force, \( V \) is aircraft velocity in feet per second, and \( \eta_p \) is the propeller efficiency factor. A typical propeller efficiency of 0.8 is assumed for this design. At cruise velocity of 260 knots, 440 feet per second, the horsepower required of the propulsion system is 6300 hp. For a twin turboprop configuration this means 3150 hp. per engine. The highest rated engine currently on the market is the PW 126 produced by Pratt and Whitney, Canada (P&WC). It is cruise rated at 2192 ehp (effective horsepower) at 1200 rpm (5). However P&WC is currently testing engines with effective horsepower in the range of 3000 ehp (5). The dimensions and weight of the engines for the Eagle RTS can be calculated using the scaling equations from Raymer’s
Aircraft Design: A Conceptual Approach (10) which are stated below in Equations 5.2, 5.3 and 5.4.

\[
L = L_{\text{actual}} (SF)^{0.4} \tag{5.2}
\]

\[
D = D_{\text{actual}} (SF)^{0.5} \tag{5.3}
\]

\[
W_t = W_{t\text{actual}} (SF)^{1.1} \tag{5.4}
\]

where \( SF \) is the scaling factor characterized by the equation,

\[
SF = \frac{T_{\text{required}}}{T_{\text{actual}}} \tag{5.5}
\]

Using the P&WC PW 126 as a base engine the Eagle RTS engine is calculated to have the dimensions as outlined in Table 5.1.

<table>
<thead>
<tr>
<th>Dimension</th>
<th>P&amp;WC PW 126</th>
<th>Eagle RTS Engine</th>
</tr>
</thead>
<tbody>
<tr>
<td>Length:</td>
<td>84 inches</td>
<td>97.1 inches</td>
</tr>
<tr>
<td>Width:</td>
<td>26 inches</td>
<td>31.2 inches</td>
</tr>
<tr>
<td>Height:</td>
<td>31 inches</td>
<td>37.2 inches</td>
</tr>
<tr>
<td>Weight:</td>
<td>1060 pounds</td>
<td>1675.6 pounds</td>
</tr>
</tbody>
</table>

Another consideration in the design of the Eagle RTS propulsion plant is the size of the propellers. In his book Daniel Raymer says that when noise is a consideration, as it is in the Eagle RTS, the helical tip speed of the propeller blades should be kept at or below 700 feet per second (10). The propeller disk diameter is calculated using Equations 5.6 and 5.7,

\[
(V_{\text{tip}})_{\text{static}} = \pi \, n \, d \tag{5.6}
\]

\[
V_{\text{helical}} = (V_{\text{tip}}^2 + V^2)^{0.5} \tag{5.7}
\]

where \( n \) is the rotational rate of the engine in revolutions per second, \( V \) is the aircraft speed in feet per second, and \( d \) is the propeller disk diameter. Using a rotational rate of 1200 rpm (20 rev/sec),
typical for the P&WC PW 100 family of engines, and a cruise velocity of 440 feet per second the
propeller disk diameter is calculated to be eight feet eight inches.

The placement of the engines can be seen in Figures 1.2, 1.3 and 1.4. The engine
dimensions and engine pylon dimensions can be seen in Figure 5.1. The engines are placed in the
position shown due to several reasons. First, the propeller blades require a minimum of nine
inches clearance (1). Secondly, in the unlikely event of a propeller blade being shedded, the blade
will not rip through the passenger compartment nor the cockpit. Also, the possibility of damaging
the aircraft controls and control surfaces will be remote during such an incident. The final reason
behind the engine placement deals with the interaction of the engine air inlet and the canard tip
trailing vortices. Despite outward appearances the heart of a turboprop engine is a jet engine and a
jet engine requires a smooth, steady flow of air to function. The canards will shed vorticies
downstream. To minimize the effect of the vortices on the flow into the engines, the engines have
been placed as far aft as possible and as high as possible on the aircraft.

A pusher configuration was chosen for the Eagle RTS for several reasons including the fact
that the propeller slipstream will not impinge on the aircraft body, therefore reducing induced
aircraft drag. Another reason for a pusher configuration is that “the inflow to the propeller keeps
the air flow attached to the tailcone” thereby reducing drag further (10). Finally, the cabin noise is
reduced by placing the engines and propellers behind the passenger compartment.
Eagle RTS
Engine Dimensions

- length: 97.1 inches
- width: 31.2 inches
- height: 37.2 inches
- weight: 1675.6 pounds
VI. Cost Analysis

The determination of total costs are based on calculations from Raymer’s Aircraft Design. Where the following variables are:

- \( W_E \) = empty weight = 40,415 lbs
- \( V \) = maximum velocity = 290 knots
- \( Q \) = quantity produced = 500
- \( FTA \) = number of flight test articles = 2
- \( C_{ENG} \) = engine cost = $186,000
- \( C_{av} \) = avionics cost = $1,000,000

The engineering hours include the design and analysis of the aircraft. Most of the time is performed during the research and development phase of the operation. The amount of engineering hours is calculated using

\[
\text{Eng hours} = 4.86 \times W_E^{0.777} V^{0.894} Q^{0.163} = H_E = 8,076,733 \quad (6.1)
\]

The amount of time necessary to prepare the aircraft manufacturers plant for tooling is determined by using the following equation,

\[
\text{Tooling hours} = 5.99 \times W_E^{0.777} V^{0.894} Q^{0.263} = H_T = 18,532,044 \quad (6.2)
\]

The amount of time needed to assemble the entire quantity produced is calculated by

\[
\text{Mfg hours} = 7.37 \times W_E^{0.82} V^{0.454} Q^{0.641} = H_M = 31,102,898 \quad (6.3)
\]
The quality control hours is the amount of time needed to ensure that the product meets the design specifications and is determined using the equation,

\[
QC \text{ hours} = (0.133)H_M = HQC = 4,136,685 \tag{6.4}
\]

The development-support cost are those cost such as mockups, simulators, and various tests. These value can be calculated using the following equation,

\[
\text{Devel. Cost} = 45.42W_E^{0.63}V^{0.822} = C_D = \$3,832,305. \tag{6.5}
\]

Flight test cost cover all expenses to receive a certification from the FAA. This includes the number of flight tests necessary to demonstrate airworthiness. The flight test costs is determined by,

\[
\text{Flight Test Cost} = 1243.03W_E^{0.325}V^{0.822}FTA^{1.21} = C_F = \$9,548,494 \tag{6.6}
\]

The manufacturing materials cost is the total cost of the raw material needed to assemble the aircraft, which includes the amount of aluminum alloys and composites. This value is calculated from the following equation,

\[
\text{Mfg Material Cost} = 11.0W_E^{0.921}V^{0.621}Q^{0.799} = C_M = \$932,490,882 \tag{6.7}
\]

The total cost of the aircraft can now be estimated using the following equation,

\[
\text{RDT&E + flyaway} = H_ER_E + H_TR_T + H_MR_M + H_QR_Q + C_D + C_F + C_M + C_{ENGN} + C_{AVN} \tag{6.8}
\]
where the hourly rates for engineering, tooling, manufacturing, and quality control are estimated in U.S. dollars\(^1\). Using this equation, the total cost of the Eagle RTS project was determined to be 5.1 billion dollars. The selling price including an investment factor for the Eagle RTS will be 10.2 million dollars. At this price, the Eagle RTS will be competitive with other aircraft in the regional commercial market.

The direct operating costs (DOC) of the Eagle RTS is divided into three sections: fuel, crew salaries, and maintenance (See Figure 6.1). The fuel cost was calculated by determining the amount of fuel burned per year. Assuming that the Eagle RTS averages 4000 flight hours per year, the fuel cost is 1.5 million dollars per 1000 flight hours. The crew salaries is estimated to be 209,000 dollars of the DOC. The maintenance costs per year can be estimated by determining the maintenance hours required per flight hour. The maintenance cost per year was calculated to be 30,000 dollars. The majority of the maintenance costs are due to the type of engine selected for the Eagle RTS. The remaining cost of the DOC is the depreciation and insurance value. Therefore the direct operating cost of the aircraft per 1000 flight hours was determined to be 1.04 million dollars.

![Figure 6.1 Direct Operating Costs](image-url)
VII. Management Plan

The Eagle RTS will have an expected operational lifetime of 60,000 flight hours or approximately 15 years. The first phase of the production of this aircraft is the research and design, which will consist of two years (See Table 7.1). This phase includes time for such areas as engineering, development, and tooling time. The manufacturing timeline of the Eagle RTS will be three years because most of the major aircraft manufacturers already possess the knowledge and equipment to produce aircraft which integrate aluminum alloys and composite materials. A total of 500 aircraft will be constructed in this phase. As stated earlier, the expected lifetime is 15 years, but could be increase due to the economic and safety features represented in this aircraft.

Table 7.1 Management Timetable
Discussion

One possible question that will arise is that of the effect of canard trailing vortices on the engine and overall performance of the aircraft. In the previous section, aerodynamics, this topic was covered and stated that the most efficient way to minimize these effects was through placing the engines and main wing as far above and aft as possible. This technique is only a temporary solution to the problem. Only time and research will provide a true answer to this problem yet, for the Eagle RTS the advantages in using a canard greatly out weigh the disadvantages.

In conducting a stability analysis, a few problems arose, namely getting the center of gravity forward enough to make the aircraft stable. But moving the wings forward helped this problem as well as balancing the aircraft and this works in conjunction with the automatic stabilization systems on-board. Having wings as large as ours gives rise to a very large moment and they have to be moved forward to counteract this effect. In roll stability, we had to move the wings up a dihedral angle of seven degrees to help in this stability problem. The flow around the fuselage tends to sideslip the aircraft and this dihedral effect helps control that problem. In dynamic analysis of this aircraft the short-period frequency and damping is too low which could lead to an uncontrollable aircraft. The way in which this could be controlled is with the automatic systems where they will give enough damping to make the airplane flyable. On the performance side, our range came out significantly better than our initial assessment and the endurance is competitive to the specified needs.

In terms of the weight of the aircraft, these values represent preliminary design estimates. A more detailed weight can be obtained once all of the external dimensions and aerodynamic characteristics are precisely known. Due to time limitations and constant adjustments in the configuration to account for stability, performance, and propulsion, an estimate of the empty weight and gross take-off weight can only be determined at this time.

Another area in which the design could be improved is with the computer structural design.
The main wing was modeled on MSC/XL as a cantilever beam. A three-dimensional design would provide a more realistic model of the forces on the main wing. Although, these initial values provide preliminary information which would allow the design team to make alterations in final design of the Eagle RTS.
Summary and Conclusions

The Eagle RTS was developed to meet a specific gap in the commercial aircraft industry. It was designed to carry passengers between metropolitan areas while avoiding the congested hub airports. The aircraft is designed to maximize performance while minimizing operational costs.

As previously stated, one of the primary considerations in designing the Eagle RTS was one of cost. The Eagle RTS was designed using proven leading edge technology to allow for an advanced aircraft while holding down the development costs.

The Eagle RTS has several interesting features such as computer controlled avionics which will allow the aircraft to continuously update and adjust its trim configuration for optimal stability. Also the aircraft computer system will calculate the optimal engine fuel flow to maintain peak engine efficiency. Another interesting feature of the Eagle RTS is that the aircraft aerodynamics were developed assuming non-laminar flow due to unclean flight surfaces and the like. This assumption is made so the consumer will have reliable fuel consumption and operational cost estimates for the Eagle RTS. If the aircraft is cleaned frequently the fuel consumption and operational costs will be minimized, thus the consumer can only benefit.

One distinguishing feature of the design of the Eagle RTS was that the structural analysis was done using MSC/XL and NASTRAN. This allowed for the maximum possible precision in the stress analysis at this stage of the design process. Based on the values determined from the NASTRAN analysis, the stresses located on the wing surfaces are significantly lower than the allowable stresses for the materials used.

The analysis of the Eagle RTS design could have been enhanced by several things. The aerodynamics could have been more tightly optimized if research on the effects of canard tip vortices on lifting surfaces was available. The evaluation of the aircraft performance would have been more precise if an engine of the required power levels had been available and values for a theoretical engine did not have to be used. However, the overall design of the Eagle RTS was well researched and will fill the void that exists in the regional transport market.
List of References


This program calculated drag polars for aircraft based on the methods in Jan Roskam's "Methods for Estimating Drag Polars of Subsonic Aircraft."

Assume Subsonic Mach numbers, \( M \leq 0.6 \)

Equation: \( C_d = (C_{do})_{wb} + (C_{do})_{v} + (C_{do})_{h} + (C_{di})_{wb} + C_d \text{ misc} \)

Where:
- \( C_d \) = drag coefficient
- \( (C_{do})_{wb} \) = zero lift drag coeff of a wing body combination
- \( (C_{do})_{v} \) = zero lift drag coeff of vertical tail(s)
- \( (C_{do})_{h} \) = zero lift drag coeff of horizontal tail
- \( (C_{di})_{wb} \) = induced drag coeff of a wing body configuration
- \( C_d \text{ misc} \) = incremental drag coeff due to miscellaneous causes

INITIALIZATION

```
INTEGER N, wsec
DIMENSION mac(10), tc(10), xt(10), SSECREF(10), SWET(10)
DIMENSION CFW(10), RLS(10)
PI=3.14159265359d0
fd=0.0
db=0.0
```

```
10 FORMAT(' *******************************************************
20 FORMAT(' ** **
30 FORMAT(' ** Drag Polar Calculator **
40 FORMAT(' ** for subsonic aircraft with **
50 FORMAT(' ** Mach<0.6 **
60 FORMAT(' **
160 FORMAT(' Enter the Aircraft Fuselage Length: ')
170 FORMAT(' Enter the Maximum Width of the Aircraft Fuselage: ')
180 FORMAT(' Enter the Maximum Height of the Aircraft Fuselage: ')
190 FORMAT(' Enter the Diameter of the Fuselage Base, 0 if tapers to')
```
& a tip: ') WRITE(*,190) READ(*,*) db WRITE(*,*) 'Enter the Wing Reference Area' READ(*,*) SREF WRITE(*,60) 70 FORMAT( ' Enter the number of sections to break wing into: ') WRITE(*,70) READ(*,*) WSEC

C (Cdo)wb
80 FORMAT( ' Calculating Cdo wb') CFW(1)=0.00238D0 CFW(2)=0.00247D0 RLS(1)=1.09D0 RLS(2)=0.84D0 CFB=0.0019D0 RWB=0.975D0 DO 110 N= 1,WSEC WRITE(*,60)

90 FORMAT( ' Wing Section ',I2) WRITE(*,90) N

100 FORMAT( ' Enter Mean Aerodynamic Chord: ') WRITE(*,100) READ(*,*) mac(N)

120 FORMAT( ' Enter Thickness Ratio: ') WRITE(*,120) READ(*,*) tc(N)

150 FORMAT ( ' Enter the chordwise position of the maximum thickness: ') WRITE(*,150) READ(*,*) xt(N)

IF(xt(N).GE.(0.3*MAC(N))) THEN L=1.2D0 ELSE L=2.0D0 ENDF

130 FORMAT( ' Enter Wing Section Reference Area: ') WRITE(*,130) READ(*,*) SSECREF(N)

140 FORMAT( ' Enter Total Wing Section Wetted Area: ') WRITE(*,140) READ(*,*) SWET(N) CDPANELI=CFW(N)*(1.0D0+L*TC(N)+100.0D0*TC(N)**4)*RLS(N) &*SWET(N)/SSECREF(N) CDPANEL=CDPANEL+CDPANELI CONTINUE

CDFB=CFB*(1.0D0+(60.0D0/(1b/fd)**3.0D0)+0.0025d0*(1b/fd))*((BSWET/ &BCSAREA) CDB=0.029*(db/fd)**3/DSQRT(CDFB) CDOWB=(CDPANEL+CFB*(1.0D0+(60.0D0/(1b/fd)**3.0D0)+0.0025D0*LB/FD) &*(BSWET/SREF))*RWB+CDB*(BCSAREA/SREF) WRITE(*,80) WRITE(*,*) CDOWB *(Cdo)v

CFV=0.0075d0
RLSV=0.75DO
WRITE(*,60)
WRITE(*,*)'** Calculating for Vertical Tail **'
WRITE(*,100)
READ(*,*) macv
WRITE(*,120)
READ(*,*) TCV
WRITE(*,150)
READ(*,*) XTV
IF(XTV.GE.(0.3*MACV)) THEN
   LV=1.20DO
ELSE
   LV=2.0DO
ENDIF
210 FORMAT(' Enter the Total Wetted Area of Vertical Tail')
WRITE(*,210)
READ(*,*) SWETV
CDOV=CFV*(1.0DO+LV*TCV+100.0DO*TCV**4)*RLSV*SWETV/SREF
WRITE(*,*)'Cdo v'
WRITE(*,*) CDOV
* (Cdo)h
CFH=0.0022d0
RLSH=0.75DO
WRITE(*,*)'** Calculating for Horizontal Tail **'
WRITE(*,100)
READ(*,*) mach
WRITE(*,120)
READ(*,*) TCH
WRITE(*,150)
READ(*,*) XTH
IF(XTH.GE.(0.3*MACH)) THEN
   LH=1.20DO
ELSE
   LH=2.0DO
ENDIF
220 FORMAT(' Enter the Total Wetted Area of Horizontal Tail: ')
WRITE(*,220)
READ(*,*) SWETH
CDOH=CFH*(1.0DO+LH*TCH+100.0DO*TCH**4)*RLSH*SWETH/SREF
WRITE(*,*)'Cdo h'
WRITE(*,*) CDOH
* (Cdi) wb
N=0.00066667DO
OMEGA=5.455472559D-4
E=0.7d0
WRITE(*,*)'** Calculating Induced Wing Body Drag **'
230 FORMAT(' Enter the Coefficient of Lift for the Main Wing: ')
WRITE(*,*)'Enter the Wing Twist, positive for wash-in: '
READ(*,*) THETA
WRITE(*,*)'Enter the Wing Span: '
READ(*,*) span
ar=span**2/sref
nu=0.69d0
cdc=1.2d0
WRITE(*,*)'Enter the Body Reference Area: '
READ(*,*) Sbref
WRITE(*,*) 'Enter the Body Planform Area: '  
READ(*,*) Sp
WRITE(*,*) 'Enter the Aspect Ratio: '  
READ(*,*) ar
Sb=BCSAREA-pi*(db/2)**2
* Miscellaneous Drag Contributions
* landing gear drag
* flaps drag
* windsheild drag
CDWINDSH=0.078*BCSAREA/SREF
* nacelle drag, neglect interference drag for turboprops
dcl1=-3.0d0
WRITE(*,*) 'Enter the Engine Pylon Chord Length: '  
READ(*,*) cn
WRITE(*,*) 'Enter the Engine Nacelle Width: '  
READ(*,*) nwidth
WRITE(*,*) 'Enter the Engine Nacelle Height: '  
READ(*,*) nheight
NCSAREA=nwidth*nheight
dn=DSQRT(NCSAREA/0.7854d0)
WRITE(*,*) 'Enter the Angle of the Nacelle to Pylon Centerline: '  
READ(*,*) eta
dcl2=-0.056d0*eta
Cdnacel=0.036d0*cn*dn/SREF*(dcl1+dcl2)**2
* speed break drag, ignored
* total misc drag
Cdmisc=CDWINDSH+CDNACEL
WRITE(*,*) 'Total Miscellaneous Drag'
WRITE(*,*) CDMISC
* DRAG POLAR
OPEN(UNIT:1,FILE='DRAGPOL.DAT',STATUS='NEW')
WRITE(*,*) 'Enter the Angle of Attack of the Main Wing Relative &to the Fuselage: '  
READ(*,*) WALPHA
WRITE(*,*) 'Enter the Lowest Angle of Attack to Compute: '  
READ(*,*) LALPHA
WRITE(*,*) 'Enter the Highest Angle of Attack to Compute: '  
READ(*,*) HALPHA
ALPHA=LALPHA-WALPHA-1.0d0
WRITE(*,*) 'Enter All Angles of Attack Relative to the Wing.'
ALPHA=ALPHA+1.0d0
BALPHA=ALPHA*PI/180.0d0
910 FORMAT(' Enter the Coefficient of Lift at ',F5.0)
WRITE(*,910) ALPHA+WALPHA
READ(*,*) CLWING
CDLW=CLWING**2/(PI*AR*E)+CLWING*THETA*2.0D0*pi*N+(THETA*2.0D0*pi)**2*OMEGA
cdalphab=2.0d0*balpha**2*Sh/Sbref+nu*cdc*Sp/Sbref*balpha**3
Cdiwb=Cdlw+cdalphab*Sbref/SREF
Cd=Cdowb+Cdov+Cdoh+Cdiwb+Cdmisc
WRITE(*,*) 'Drag Angle of Attack'
WRITE(*,*) Cd, ALPHA
WRITE(1,*) ALPHA, CD, CLWING
IF(ALPHA.LE.HALPHA-WALPHA) THEN