Duck (Passenger), Goose (Cargo), Duckling (Economic)

Conceptual Design

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1 Introduction

1.1 Abstract

Many aircraft are designed to be amphibious but struggle to have multiple missions with the same designed fuselage. The Duck, Goose, Duckling is an aircraft designed to have a short landing and takeoff in either land (dirt, grass, metal mat, gravel, asphalt & concrete) or water and be able to do passenger, cargo, and economic missions. When the aircraft is used for passenger missions it is referenced as the Duck, when used for cargo it is called Goose, and Duckling for the economic mission. This design was referenced from many amphibious aircraft that are used today to further understand where the Duck can exceed those past aircraft expectations.

1.2 Mission Requirements

The request for proposal (RFP) is to design an amphibious aircraft capable of multiple mission types. The RFP specifies passenger and cargo missions that utilize the same basic airframe. Although the requirements per mission have different ranges and payload specifications, they are classified as mandatory (M) and tradeable (T) within the mission specifications.

General Mission Requirements include:

- (M)Takeoff and landing (DRY): runway types include anything not a body of water, dirt, grass, etc,
- (M) Takeoff and landing (WET): from fresh and saltwater
- (M) Cruise speed (Vcru): 200 knots or greater
- (M) Capabilities: VFR and IFR flight. Flight in known ice conditions
- (M) Meets FAA 14 CFR Part 23 certification
    - All missions assume reserves and equipment to meet FARs
- (M) Energy cost: at least 20% better than pre-existing aircraft for similar mission length
    - Economic mission range of 150 nmi for single class configuration

- (M) Propulsion system documentation.

Passenger Mission Requirements include:

- (M) Crew: 1 flight crew
- (M) Passengers:
  - Capacity: 19 passengers (seat pitch of 28”)
  - Weight: around 193.6 lb. per passenger
  - Baggage: around 37.4 lb. per passenger, volume of 4 $[ft^3]$
- (M) Range: 250 nmi
  - (M) Takeoff and landing (STOL) runway mission with 19 passengers.
  - Max takeoff and landing field lengths of 1500’ over 50’ obstacle
  - Takeoff, and landing at 50000 ft above mean sea level (MSL) runway landing
  - (M) 250 nmi (STOL) water mission with 19 passengers
  - Max takeoff and landing field lengths of 1900 ft over 50 ft obstacle
  - Takeoff, and landing at 50000 ft above (MSL) runway landing
  - (M) takeoff and land in Seat State 3 conditions.

Cargo Mission Requirements include:

- (M) Payload: 5000 lbs (about 2267.96 kg)
- (M) Range: 200 nmi
  - Takeoff and Landing field length over a 50’ obstacle (Assume MSL ISA +18 F) for both runway and water missions.
- (M) Ability to unload, refuel and load cargo in less than 60 minutes.

Economic Mission Requirements include

- (M) Passengers:
  - Capacity: 19 passengers (seat pitch of 28”)

- Weight: around 193.6 lb. per passenger
- Baggage: around 37.4 lb. per passenger, volume of 4 [ft3]

- (M) Range: 150 Nmi
- (M) Cruise speed: 200 knots or greater

1.3 Representative Amphibian Aircraft Designs

The RFP calls for a single aircraft capable of multiple missions of various types. The RFP also has a Mandatory requirement of a cost per passenger of at least 20% better than existing aircraft. This section outlines a few existing similar aircraft that could be analyzed to meet the 20% or better goal. A summary for each plane and its power plant is provided in table 1. These values were used to calculate an approximate value of the general lift coefficient and drag polar.

The Japanese US-2 is a search and rescue aircraft that has been in service since the early 2000's. It is powered by 4 Rolls Royce AE2100J Turboprop engines each capable of 4590 hp and a total sea level thrust of around 18360 hp. The variable pitch turbo prop propellers give the aircraft excellent takeoff and landing performance.

The Russian Be-200 is also a search and rescue aircraft that has been in service since the mid 1980's. It is powered by 2 Russian D-436TP Turbofan engines each with around 1500 hp and a total seal level thrust of around 30,000 hp. The turbofan engines allow for a greater overall service ceiling and velocities.

The Canadair CL-415 search and rescue has been around since the mid 1990s. The power plant of this beast is 2 Pratt & Whitney R-2800-83 AM engines each with 2100 hp and a combined total power of around 4200 hp. Slightly lower than the Russians and Japanese because this plane was converted from a piston engine to a turboprop.

Table 1. Summary of similar existing aircraft.

Name	**US-2**	**Be-200**	**CL-415**	**Units**
Span	33.15	32.8	28.5	**[m]**

Area	135	117	100	**[m^2]**
Aspect Ratio	8.14	9.19	8.12	
Max TO Weight	47700	41000	20000	**[kg]**
Gross Weight	22070	13400.00	7850	**[kg]**
Empty Weight	25630	27600.00	12150	**[kg]**
Power Plant	**US-2**	**Be-200**	**CL-415**	
Manufacturer	Rolls Royce	Progress	Pratt & Whitney	
Model	AE2100J	D-436TP	R-2800-83AM	
No. of Engines	4	2	2	
Power (Each)	4590	15281	2400	**[hp]**

The gross weight was calculated from the greatest max takeoff weight (DRY or WET). Then the lift coefficient could be calculated and nondimensionalized as

$$L = W = q_\infty S_{ref} C_L \qquad C_L = \frac{W}{q_\infty S_{ref}}$$

The drag was broken down between the zero-lift drag (D_0) and the induced drag (D_i). The zero-lift drag, parasite drag (D0), is expressed in terms of the equivalent skin friction coefficient (C_{fe}) and wetted surface area (S_{wet}) then non-dimensional coefficient form

$$D_0 = q_\infty S_{wet} C_{fe} \qquad C_{D0} = \frac{D_0}{q_\infty S_{ref}}$$

then, Prandtl's lifting line theory defines the induced drag coefficient

$$C_{Di} = \frac{D_i}{q_\infty S_{ref}}$$

is given by

$$C_{Di} = \frac{C_L^2}{\pi e AR}$$

Combining these the general drag coefficient for each plane was determined by

$$C_D = C_{D0} + C_{Di} \qquad C_D = C_{D0} + \frac{C_L^2}{\pi e AR}$$

The values used in the general drag coefficient for each plane can be found in table 2, along with the cruise conditions.

Table 2. Cruse conditions for existing similar aircraft and the values used to determine the drag Coeff.

(Steady Level)	**US-2**	**Be-200**	**CL-415**	**Units**
Cruise ALT	6100	8000	3000	[m]
cruise Air density	0.4671	0.5258	0.9093	$[\frac{kg}{m^3}]$
cruise speed	133	155	92.5	$[\frac{m}{s}]$
Cruse Speed of sound	303	308	328	$[\frac{m}{s}]$
Cruse Velocity	133	150	92.6	$[\frac{m}{s}]$
Cruse Mach No.	0.437	0.503	0.281	
Dynamic Viscosity	1.493	1.527	1.694	$[\frac{N-s}{m^2}] \cdot 10^{-5}$
Reynolds No	1.1	1.8	1.6	$\cdot 10^7$
induced Drag Factor	0.03	0.03	0.03	
spans eff	0.97	0.97	0.97	
Lift Coeff	0.38	0.17	0.19	
Zero lift Drag Coeff	0.025	0.02	0.023	

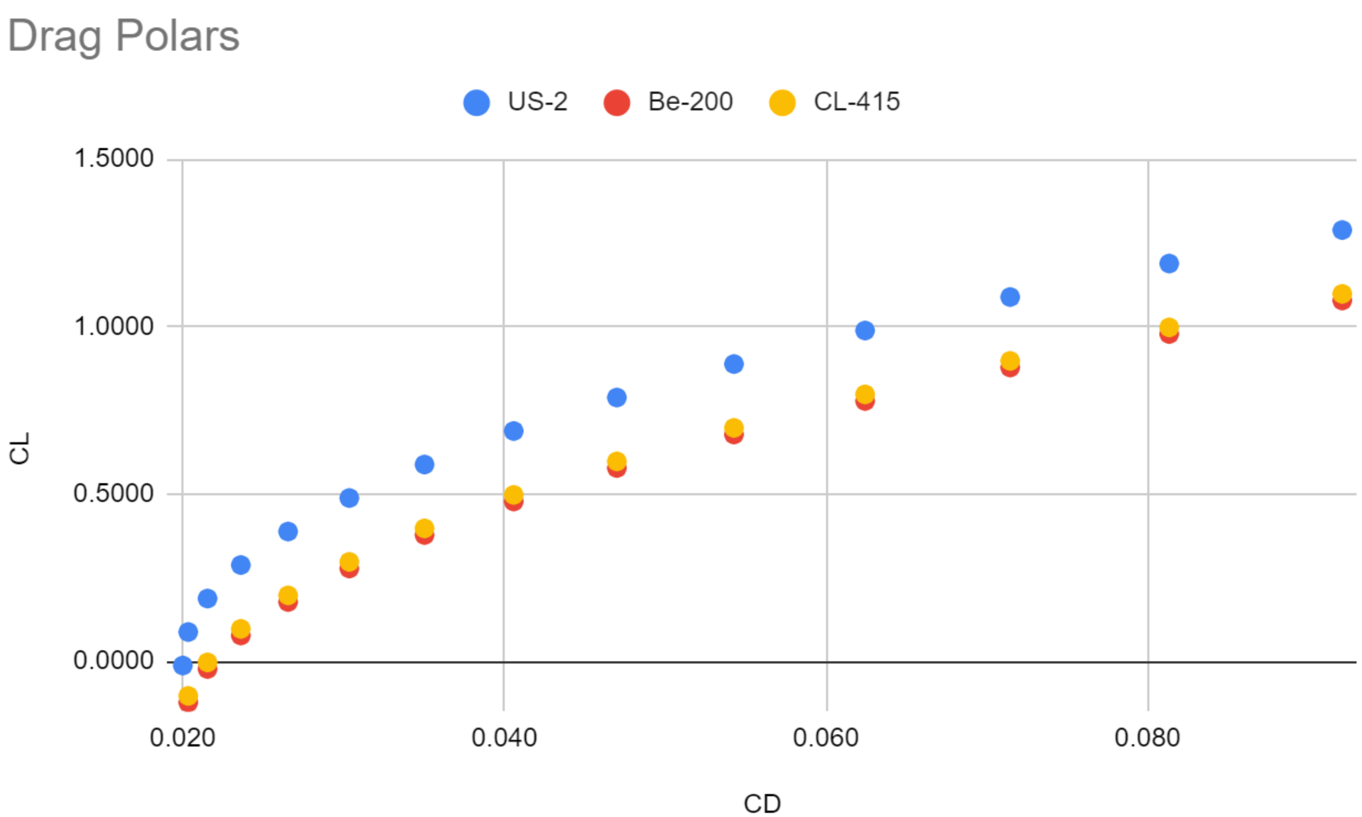
The drag could be found by multiplying the drag coefficient by the dynamic pressure, then lift to drag ratio could be calculated. Note, for approximation purposes, the zero lift drag coeffect is about 0.023 after research on AIAA papers. A summary of performance parameters in steady level flight for each plane is shown in table 3. The general drag polar for each aircraft is shown in figure 1.

Table 3. Performance Parameters calculated for similar existing Aircraft calculated from table 1&2.

(Steady Level)	**US-2**	**Be-200**	**CL-415**
(L/D)	1.27	0.858	0.820
(W/S)	163.5	114.5	78.5
(T/W)	0.785	1.1652	1.2

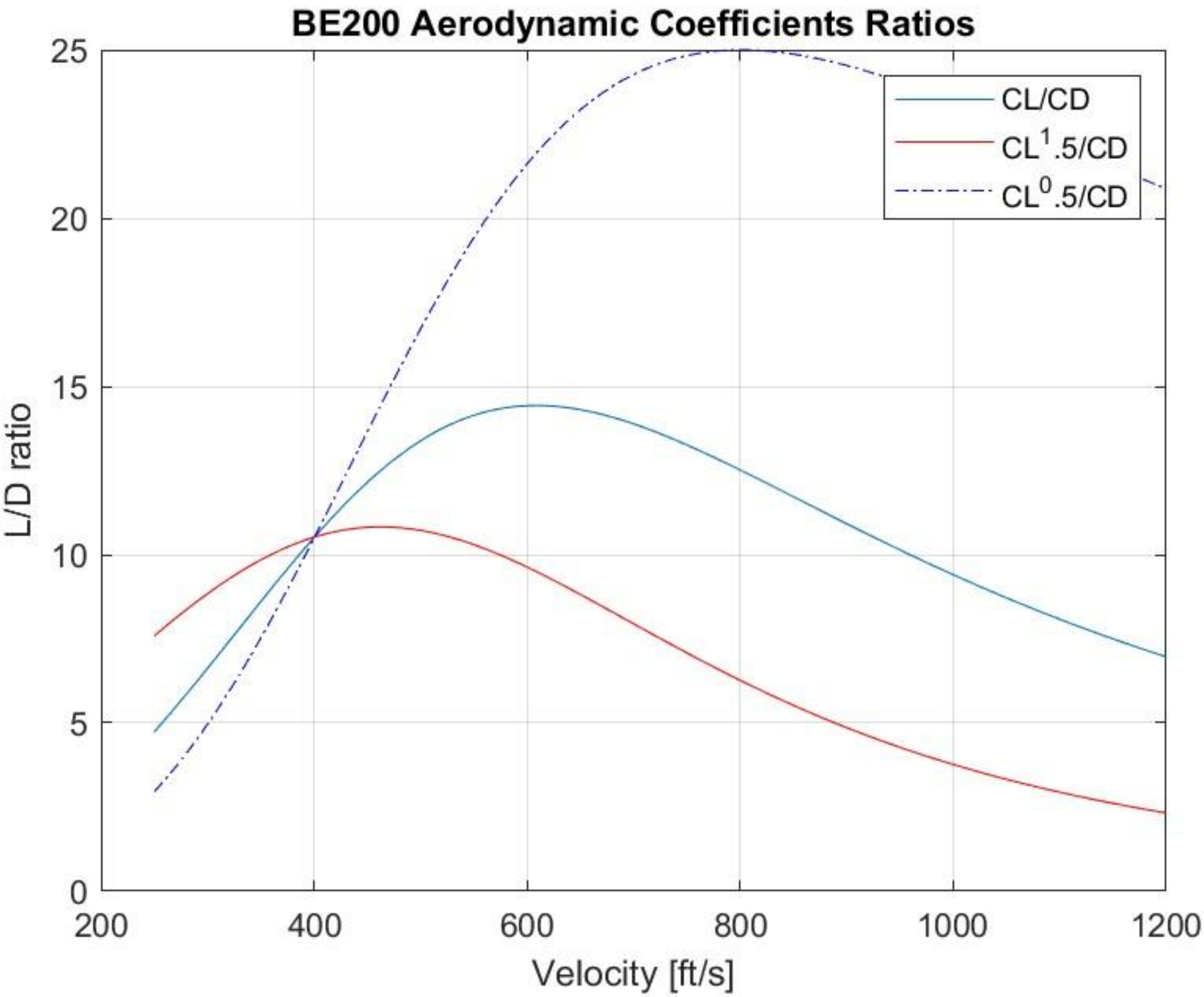
(P/W)	0.620	1.700	0.398

Figure 1. Plot of General Drag Polar curves for similar preexisting aircraft in table 2.



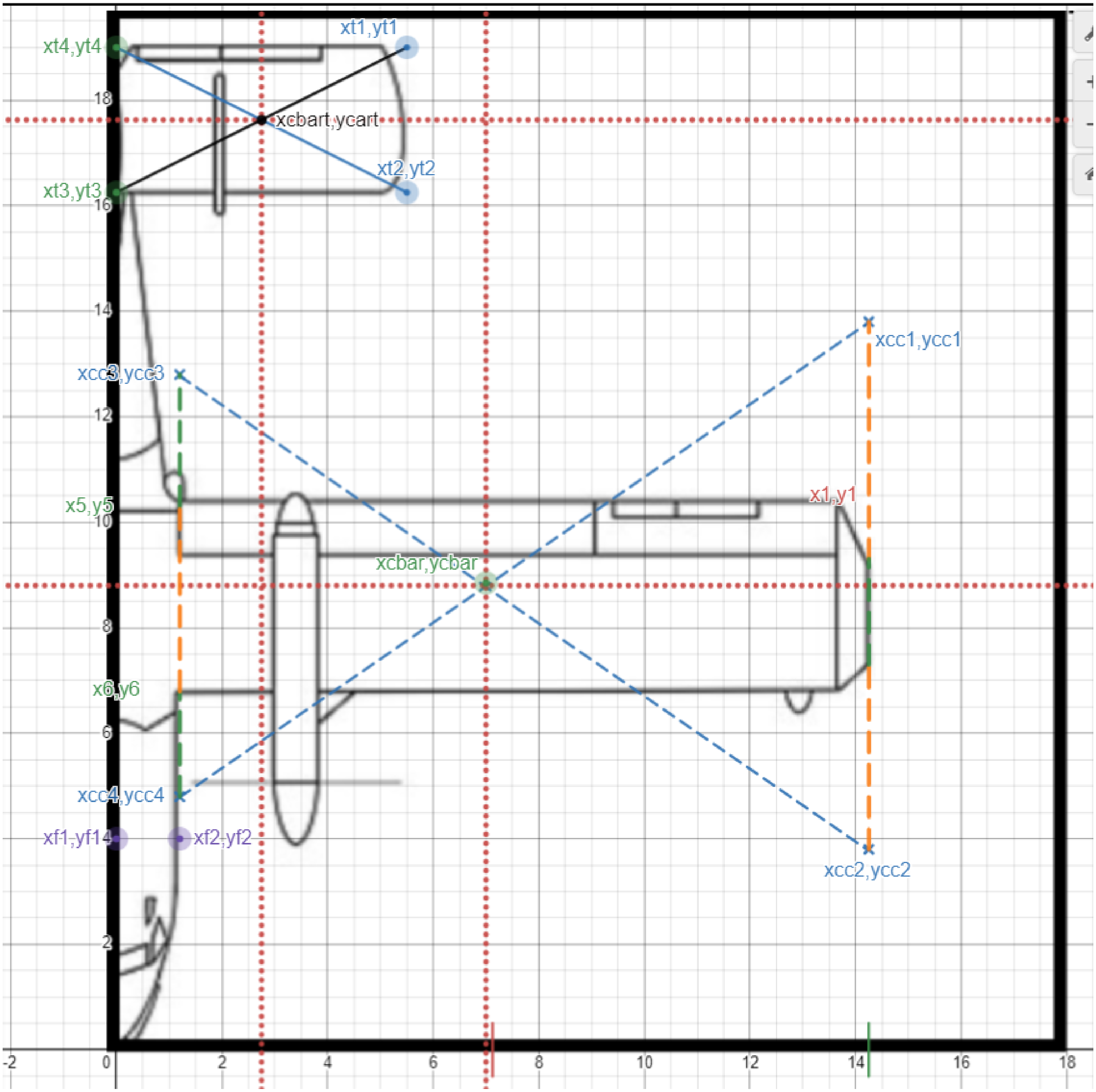
From here, the aerodynamic relation associated with $\left(\frac{C_L}{C_D}\right), \left(\frac{C_L^{\frac{3}{2}}}{C_D}\right), \left(\frac{C_L^{\frac{1}{2}}}{C_D}\right)$ could be approximated for each aircraft then used to bench mark our aircraft design.

Figure 2. Aerodynamic coefficient of the Be-200 generated in MATLAB



After this stability criteria for the wing and the aircraft's tail, like the mean aerodynamic chord and location a few general calculations were done to find some stability criteria for the wing and the tail of the aircrafts, like the mean aerodynamic chord and location, respectively. These were needed to get a feel for how the aircraft would respond to loading configurations. Figure 3 shows the planform view of the CL-415 along with locations of values needed to obtain data from the wing and tail configurations.

Figure 3. Planform view of CL-415 used to calculate mean aerodynamic chord location & length.



1.4 Conclusion

The data collected on preexisting aircraft is needed to evaluate the current market. This also allows us to incorporate the best characteristics of each aircraft into our design. For example, the STOL of the US-2 seaplane is due to its high lift 'plane flap' design and its boundary layer control system. Controlling the boundary layer to increase the laminar flow on the surface of the wing reduces the turbulent flow and the skin friction along with it. Although this data should be used for benchmarking later it gives us a general starting point when considering the design of our aircraft. It will also show how our design compares to the preexisting market.

2 Initial Sizing

2.1. Mission Profile

The Duck, Goose, and Duckling must perform short take-offs/landings while carrying passengers or cargo. Each mission will be described below on how each gross weight is estimated and along with that trade studies done to show how range, payload, passenger amount, and velocity affect the mission.

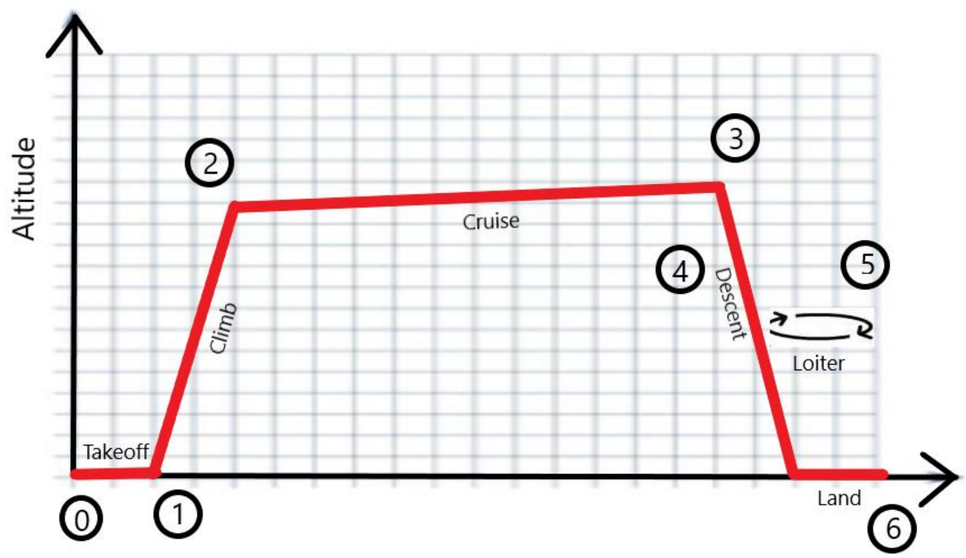


Figure 4. Cruise Mission Profile

2.2. Gross Weight Estimations

Gross weight is the total weight of the aircraft during takeoff, so it is important to note that the gross weight calculated below does not necessarily mean the maximum takeoff weight of the aircraft. The weight calculated below is the design weight that will start the design process of the aircraft. This design weight equation is:

$$W_o = W_{crew} + W_{payload} + W_{fuel} + W_{empty}$$

This is then rearranged to:

$$W_o = \frac{W_{crew} + W_{payload}}{1 - \left(\frac{W_f}{W_o}\right) - \left(\frac{W_e}{W_o}\right)}$$

The next step in the design process is to calculate the empty weight fraction and fuel fraction by examining empirical data from past aircraft to determine the design weight of the aircraft.

2.2.1. Passenger Mission

For the Duck, the gross weight at takeoff is calculated by using Raymer’s textbook equations for initial sizing. This is done by using the empirical data and equations supplied by Raymer. Using Table 3.1 (Empty Weight Fraction vs W_O) in Raymer's textbook the Duck designed choice is using empirical data for a Twin Turboprop aircraft.

$$A = .96$$

$$C = -.05$$

Initial gross weight guess range for a twin turboprop:

$$W_o = 5{,}000 \; to \; 80{,}000 \; lbs$$

Cruise-segment mission weight fractions are found by using the Breguet range equation.

$$R = \frac{V}{C}\frac{L}{D}\ln\frac{W_{i-1}}{W_i} \; or \; \frac{W_i}{W_{i-1}} = \exp\frac{-RC}{V\left(\frac{L}{D}\right)}$$

Loiter weight fractions are found from the endurance equation.

$$E = \frac{\frac{L}{D}}{C}\ln\frac{W_{i-1}}{W_i} \; or \; \frac{W_i}{W_{i-1}} = \exp\frac{-EC}{\frac{L}{D}}$$

Where

$$R = range\;(ft)$$

$$C = specific\;fuel\;consumption$$

$$V = velocity\ \left(\frac{ft}{s}\right)$$

$$\frac{L}{D} = lift\ to\ drag\ ratio$$

$$E\ =\ endurance\ (\mathrm{s})$$

For initial sizing Raymer uses historical data for Mission-Segments Weight fractions for warm-up and takeoff, climb, and landing. These values are found in Table 3.2.

$$Warm\ up\ and\ takeoff\ =\ .970$$

$$C\lim b\ =\ .985$$

$$Landing\ = .995$$

The Duck, Goose, and Duckling initial sizing are propellor-driven aircraft and for that, we use a propeller efficiency of .8. To calculate the specific fuel consumption of a twin turboprop aircraft we use Table 3.4 and find that

$$For\ Cruise\!:\ C_{bhp}\ =\ .5:\ For\ Loiter\!:\ C_{bhp}\ =\ .6\ \frac{lb}{\frac{hr}{bhp}}$$

To convert to C from Cbhp:

$$C = \frac{C_{bhp}V}{550\eta_p}$$

Our L/D estimated value is then found by eqn. 3.12 from Raymer's book. The designed aircraft we are using is going to use the K_{LD} value of 11 (for retractable prop aircraft). Using Fig 3.6 (Wetted area ratios) we see that for our design our $\frac{S_{wet}}{S_{ref}}$is around 6. We chose 6 because we know we will have a larger aircraft than a Beech Duchess (value of 5) but smaller than a Boeing 747 (value of 6.5). That last value chosen for the L/D estimated value is the aspect ratio. This value was chosen as 6.75.

$$\frac{L}{D_{max}} = K_{LD}\ \sqrt{\frac{A}{\frac{S_{wet}}{S_{ref}}}}\ \ (3.12)$$

Now that our L/D is estimated we now use this value for the cruise and loiter mission segments. For the cruise segment for a prop aircraft, we use the L/Dmax, and for the loiter segment we use .866*L/Dmax. To finish the initial sizing portion of our aircraft we use the fuel-fraction estimation equation. This equation is:

$$\frac{W_f}{W_o} = 1.06\left(1 - \frac{W_x}{W_o}\right)$$

The value 1.06 is the 6% allowance for reserved and trapped fuel and the rest of the equation is 100% subtracted by the mission weight fraction. To calculate $\frac{W_x}{W_o}$ you take the summation of the mission segments:

$$\frac{W_x}{W_o} = \frac{W_1}{W_o} \cdot \frac{W_2}{W_1} \cdot \frac{W_3}{W_2} \cdot \frac{W_4}{W_3} \cdot \frac{W_5}{W_4} \cdot \frac{W_6}{W_5}$$

Table 4. Passenger Mission Segments

Mission Segments		
Warm-up Takeoff	$\frac{W_1}{W_o} = .97$	
Climb	$\frac{W_2}{W_1} = .985$	
Cruise	$\frac{W_3}{W_2} = .9595$	Notes: Cruise @ a Range of 250 nm Propeller Efficiency = .8
Loiter	$\frac{W_4}{W_3} = .9365$	Notes: FAA Loiter Time 45 mins Propeller Efficiency = .7
Descent	$\frac{W_5}{W_4} = .995$	
Land	$\frac{W_6}{W_5} = .985$	

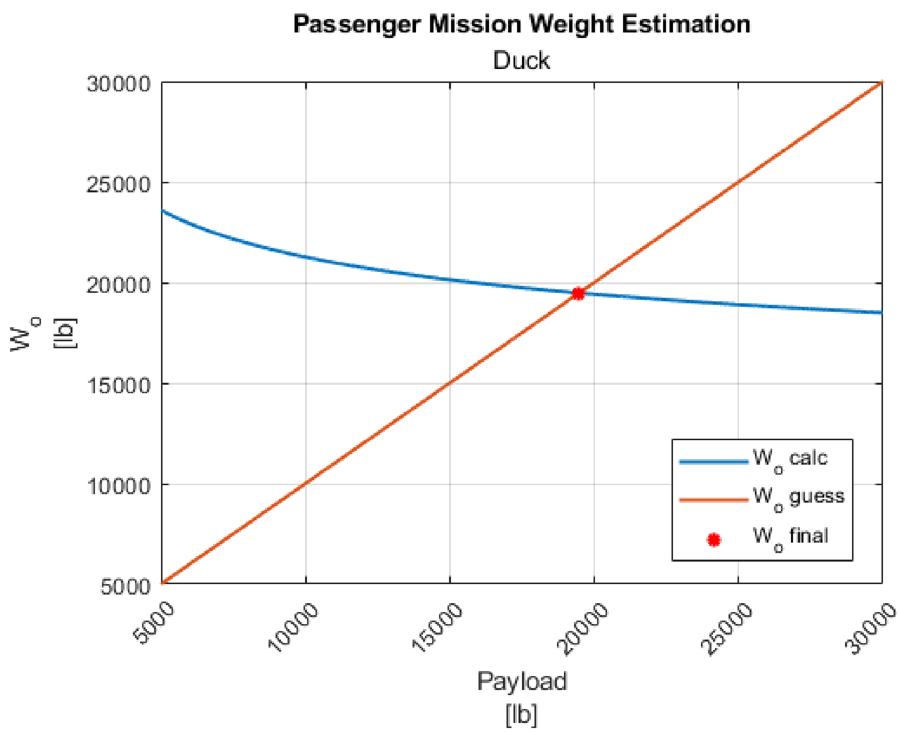
$$\frac{W_x}{W_o} = .8417$$

$$\frac{W_f}{W_o} = .1678$$

By using an iterative process, we can solve for the initial design weight of the Duck

$$Duck\ Intial\ Design\ Weight\ = 19{,}473\ lbs$$

Figure 5.



2.2.2. Cargo Mission

For the next two sections we use the same process but with different Range and Payload variables. These are described in the Tables below.

Table 5. Cargo Mission Segments

Mission Segments

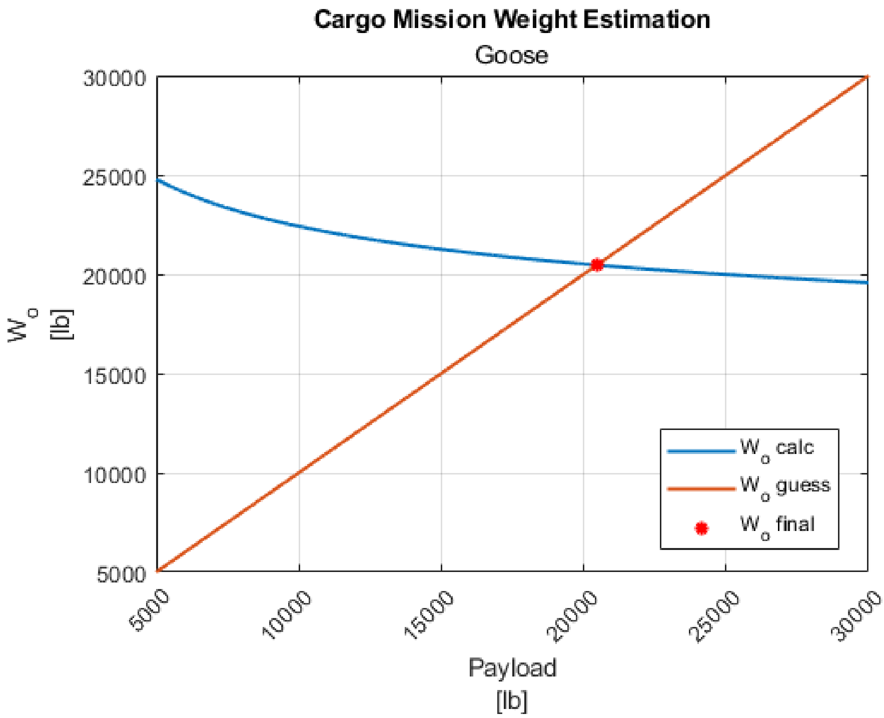
Warm-up Takeoff	$\frac{W_1}{W_o} = .97$	
Climb	$\frac{W_2}{W_1} = .985$	
Cruise	$\frac{W_3}{W_2} = .9677$	Notes: Cruise @ a Range of 200 nm Propeller Efficiency = .8
Loiter	$\frac{W_4}{W_3} = .9365$	Notes: FAA Loiter Time 45 mins Propeller Efficiency = .7
Descent	$\frac{W_5}{W_4} = .995$	
Land	$\frac{W_6}{W_5} = .985$	

$$\frac{W_x}{W_o} = .8486$$

$$\frac{W_f}{W_o} = .1604$$

$$Goose\ Initial\ Design\ Weight\ =\ 20{,}462\ lbs$$

Figure 6.



2.2.3. Economic Mission

Table 6. Economic Mission Segments

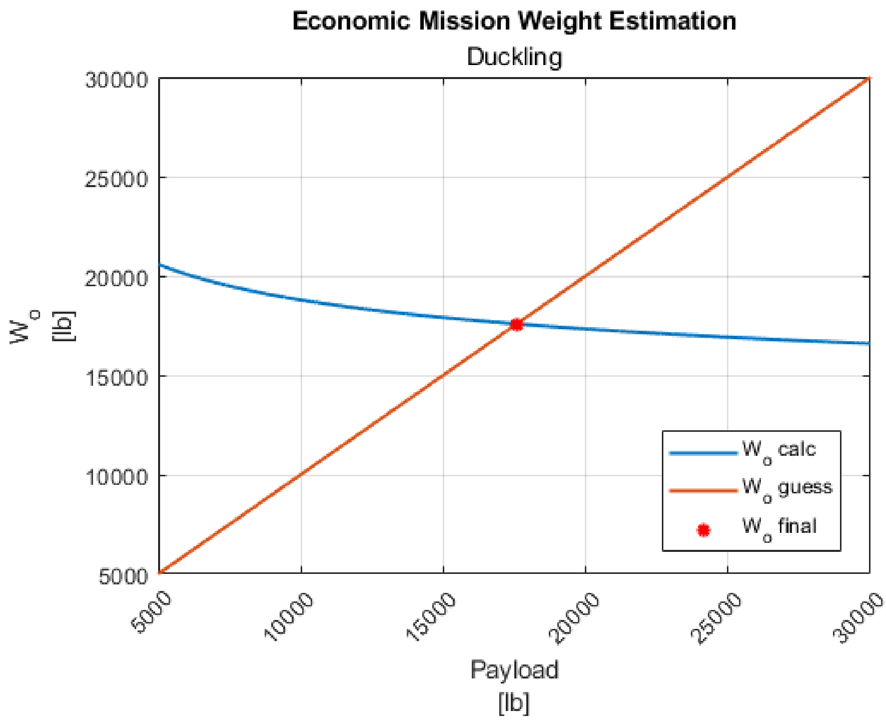
Mission Segments		
Warm-up Takeoff	$\frac{W_1}{W_o} = .97$	
Climb	$\frac{W_2}{W_1} = .985$	
Cruise	$\frac{W_3}{W_2} = .9756$	Notes: Cruise @ a Range of 150 nm Propeller Efficiency = .8
Loiter	$\frac{W_4}{W_3} = .9493$	Notes: FAA Loiter Time 45 mins Propeller Efficiency = .7
Descent	$\frac{W_5}{W_4} = .995$	
Land	$\frac{W_6}{W_5} = .985$	

$$\frac{W_x}{W_o} = .8673$$

$$\frac{W_f}{W_o} = .1406$$

$$Duckling\ Initial\ Design\ Weight\ =\ 17{,}580\ lbs$$

Figure 7.



2.3. Trade Studies

Trade Studies are now done to show how payload, range, and velocity affect the different missions on the aircraft. This is key to understanding what key requirements may be changed to increase the mission's effectiveness. Range and Payload are looked at first as how they change the overall design weight of the aircraft. As we can see that once range and payload is increased you will see an increased value of design weight for the Duck, Goose, and Duckling Mission.

Figure 8.

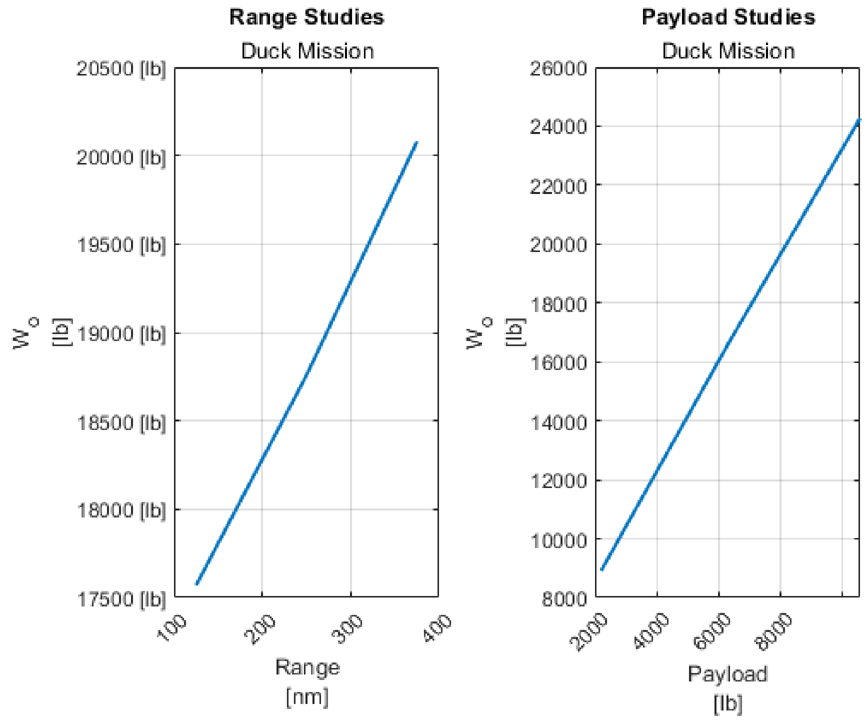


Figure 9.

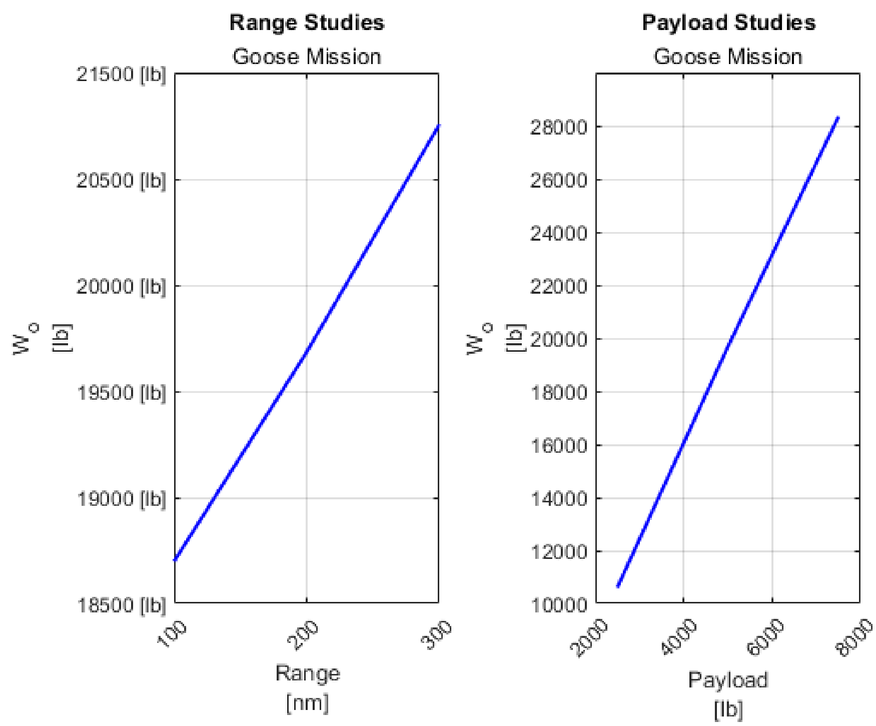
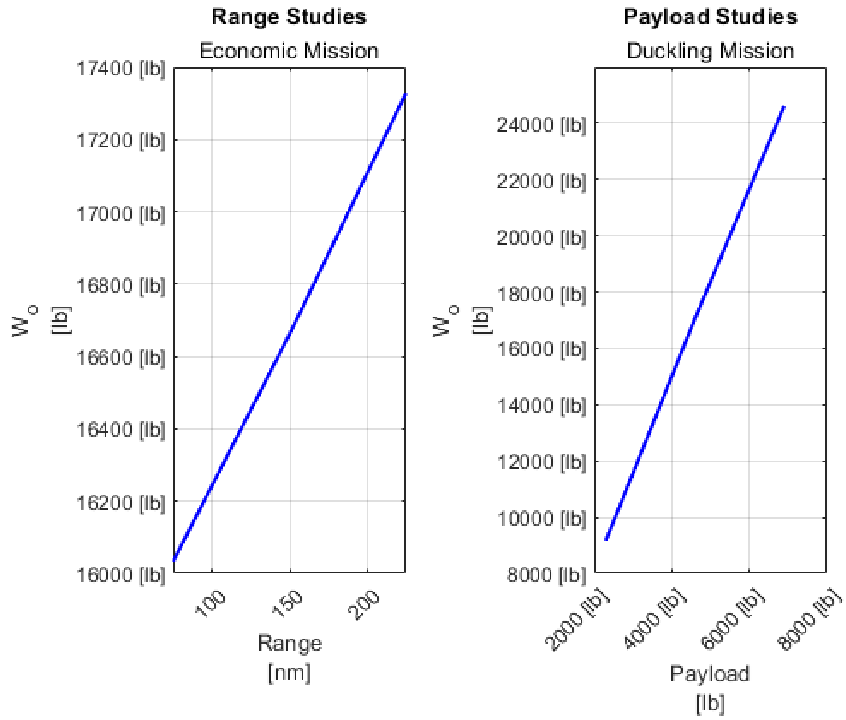


Figure 10.



2.4. Conclusion

For our initial design weight, we chose the weight that is calculated for the Goose Mission due to having to be that weight to complete each of the three missions. This design's weight came to be 19,685 lbs (about 8928.96 kg). The figures below are comparison figures of the missions and as you can see the Goose mission will be the main factor in our design choices for our aerodynamics, performance parameters, and the redefined sizing method used later in the report.

Figure 11.

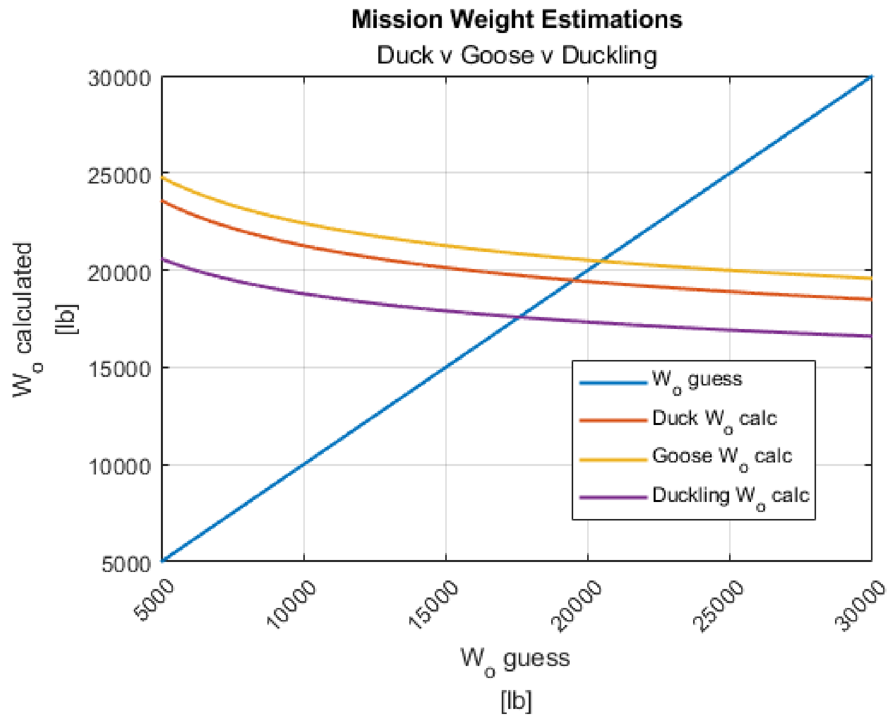


Figure 12.

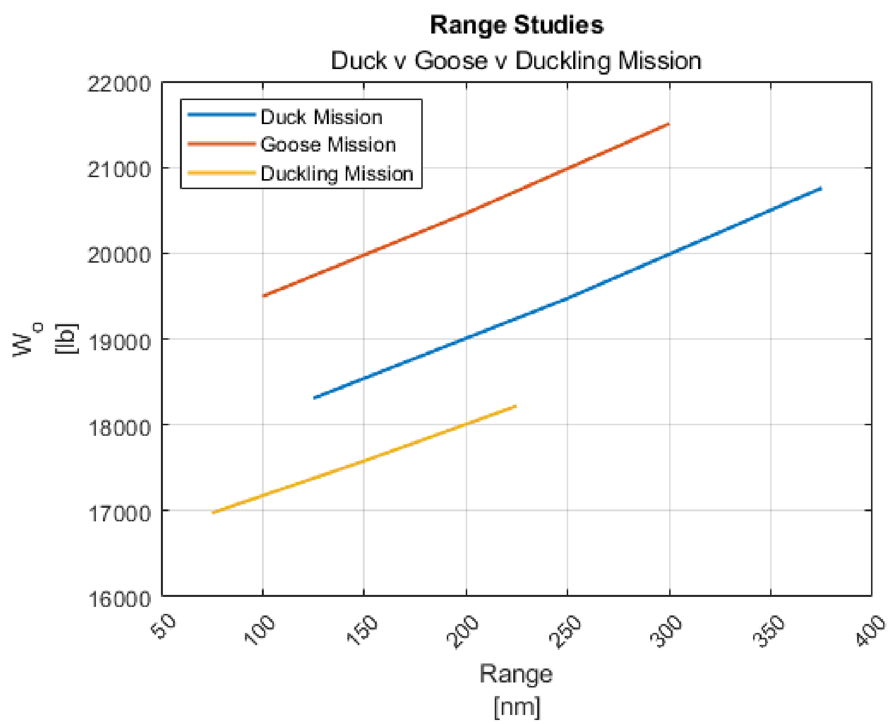
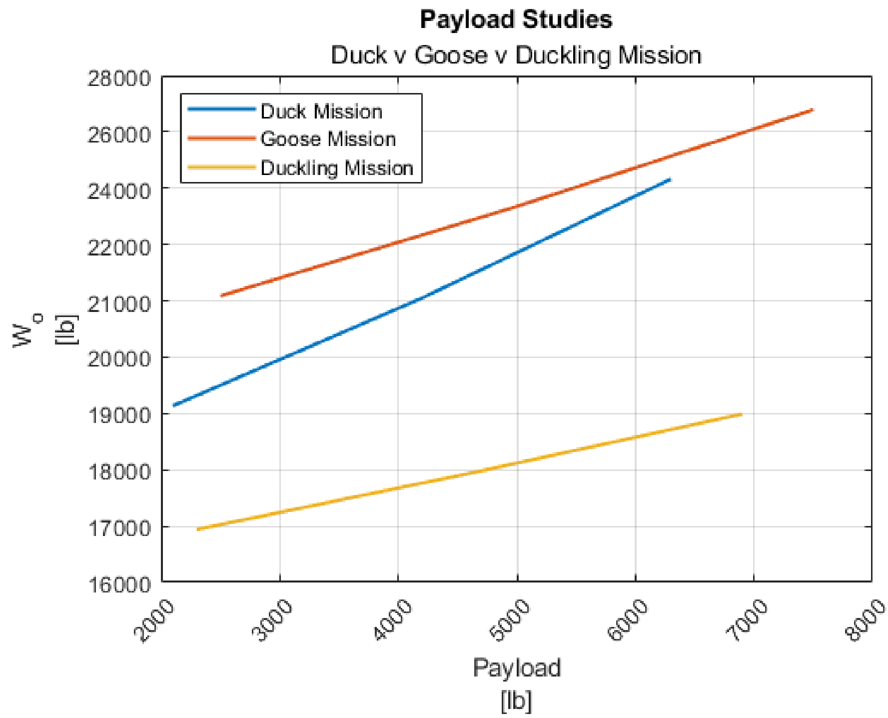
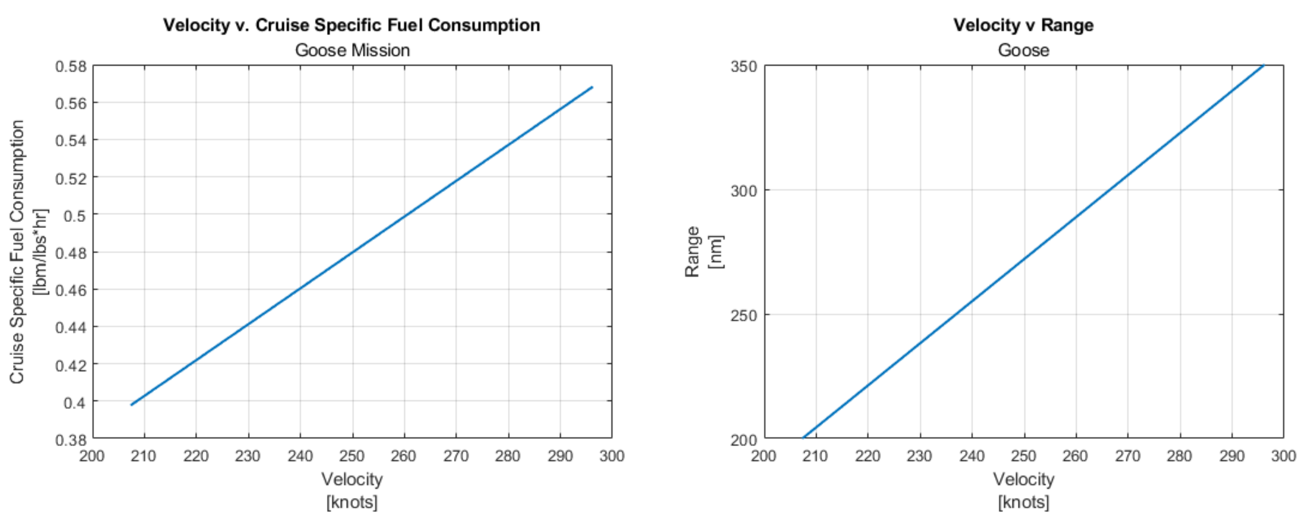


Figure 13.



For the Goose Design weight, there are studies done on how velocity increases affect the specific fuel consumption and how we can find a trade on whether increasing the cruise velocity may be beneficial to certain missions. We look only at the cruise conditions because the aircraft will be flying in the mission's cruise segment for most of its flight. Focusing on this will be the best way to increase the efficiency of the aircraft.

Figure 14. Velocity Studies



As velocity increases, the range and cruise specific fuel consumption also increases. Burning more fuel is not necessarily a good option when discussing a trade-off due to the price of fuel being so high when dealing with aircraft transportation. For range, we see an increase in value which is good, but you must be wary of initialing increasing the velocity to increase the range since fuel consumption is also increasing. By examining the data, the customer may decide on whether to increase or decrease cruise velocity to improve the efficiency of specific missions. For example, when flying the Duckling mission the mandatory required range is only 150 nautical miles so by having a decreased range value we can fly at a faster cruise velocity since payload and range is also decreased for the Duckling mission.

3 Airfoil/Planform Selection

3.1. Design Lift Coefficient

For the design lift coefficient calculation, we assume steady-level flight equations. At a designed cruise altitude of 25,000 ft (about 7.62 km) and using standard atmospheric properties for that specific altitude we can study how ranged cruise velocities affect our choice for design lift coefficient. With a wing reference area of 467.5 ft^2, we get a reasonable value for our wing loading parameter for twin turboprops for takeoff. Though this parameter is discussed later in chapter 4. To calculate the wing loading for the cruise we use

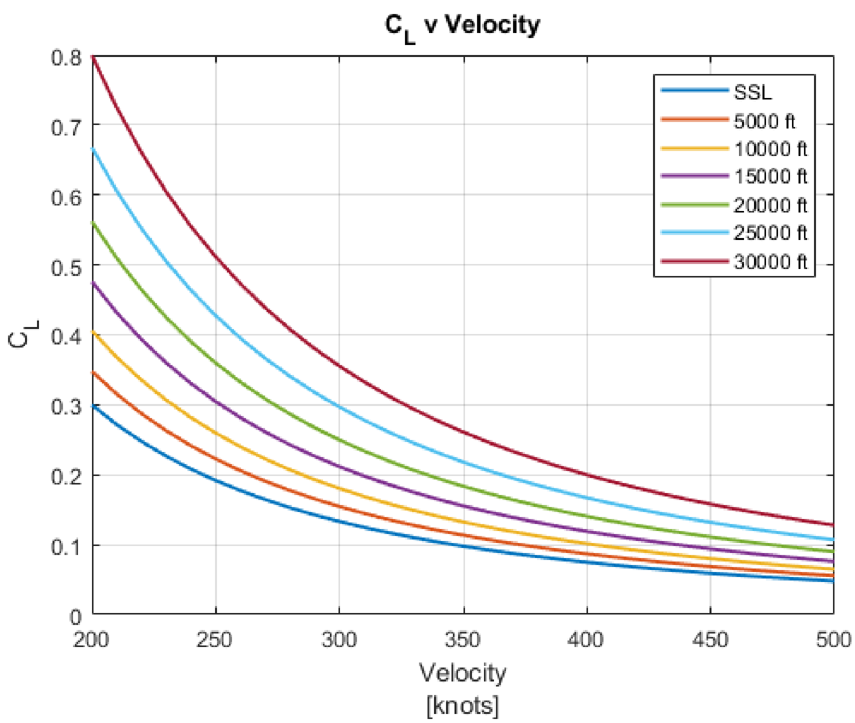
$$W = L = \frac{1}{2}\rho_\infty V_\infty^2 c_L S_{ref}$$

$$\left(\frac{W}{S}\right)_{cruise} = \frac{W_o}{S}\cdot\frac{W_1}{W_o}\cdot\frac{W_2}{W_1} = 40.6\ \frac{lb}{ft^2}$$

$$C_l = \frac{l}{q}\left(\frac{W}{S}\right)_{cruise}\ where\ q\ =\ dynamic\ pressure\ =\ \frac{1}{2}\rho_\infty V_\infty^2$$

For each mission, we have a minimum cruise speed of 200 knots and a target cruise speed of 250 knots. In the following sections we look at how the design lift coefficient changes as cruise velocity and altitude changes. This is a key understanding on how setting a cruise velocity and cruise altitude for different missions can also increase the aircraft's efficiency.

Figure 15. Lift Coefficient vs. Velocity



3.1.1. Passenger Mission

For the Duck mission design lift coefficient, we use a cruise velocity of 250 knots and a cruise altitude of 25,000 ft.

$$C_L = .427$$

3.1.2. Cargo Mission

For the Goose mission design lift coefficient, we use a cruise velocity of 250 knots and a cruise altitude of only 20,000 ft instead of 25,000 ft because the mandatory required range is only 200 nautical miles. Being a shorter distance flight, it will take more fuel to climb to 25,000 ft and causing the aircraft to burn more fuel due to its increased payload. With these parameters the design lift coefficient will be:

$$C_L = .36$$

3.1.3. Economic Mission

For the Duckling mission design lift coefficient, we use a cruise velocity of 300 knots and a cruise altitude of 15,000 ft. This trade choice is due to the shorter mandatory range of 150 nautical miles. As described in the conclusion of chapter 2, it was studied that flying at a faster speed at a shorter range is more efficient than flying at a lower speed. The altitude is quite lower due to its short range. The design lift coefficient is calculated as:

$$C_L = .21$$

3.2. Airfoil Selection

Another important aspect of aircraft design is the airfoil selected for the aircraft. Airfoils are defined by key geometric parameters that define how they perform in different flight conditions. The locations of the leading-edge point and the trailing edge define the overall chord, or length, of the airfoil. Most airfoils designed for lift will also feature a camber, the curving characteristic of the airfoil. The camber produces lift at a zero-degree angle of attack but will also increase the drag and pitching moments of the airfoil. Finally, the airfoil's thickness can also be defined as a point of maximum thickness or a distribution. The thickness of the airfoil affects its profile drag.

Knowing how each of these parameters affect the performance characteristics of the airfoil, it helped to narrow the possible choices for selecting the appropriate airfoil for the design aircraft. Furthermore, doing some research into similar aircraft with STOL capability and seeing what airfoils are used on those also contributed to the possible airfoil candidates for the design aircraft.

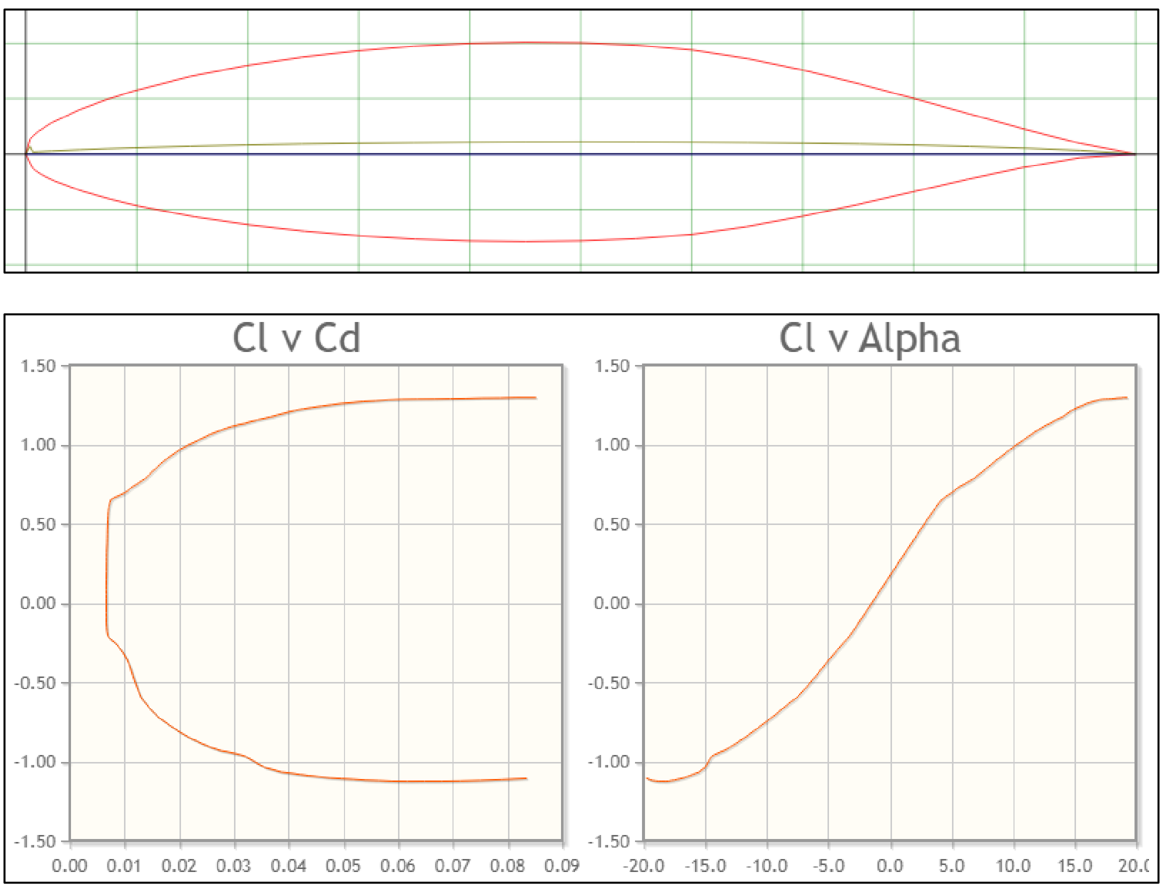
Table 7. Airfoil Studies

Similar Aircraft	Root Airfoil	Tip Airfoil
Beriev Be-200	TsAGI 16%	TsAGI 11.5%
Canadair CL-415	NACA 4417 mod	NACA 4417 mod
CASA 212	NACA 65-218	NACA 65-218
DeHavilland DHC-6	NACA 63A516 mod	NACA 63A516 mod
DeHavilland DHC-7	NACA 63A418 mod	NACA 63A415 mod
ShinMeiwa US-1	NACA 63A221	NACA 63A109

A notable parameter that these airfoils share is their thickness. The root airfoils of these STOL capable aircraft have a thickness ratio of at least 16% to 21%, and the tip airfoils are typically the same or smaller at 11.5% to 18%.

Looking at the general characteristics of the aircraft above, the airfoil from the CASA 212 would be suitable pick for the purpose of the design aircraft, since the performance of the CASA 212 is most similar to what our aircraft is expected to achieve. The NACA 65-218 is an airfoil with a max thickness of 18% and a camber of 1.1% and can produce good lift coefficient values at relatively high angles of attack (1.2 C_l at 12 degrees or 1.25 C_l at 18 degrees depending on Reynolds number).

Figure 16. Airfoil Data



3.3. Aspect Ratio

The aspect ratio is a measure of the span of the wing squared over its reference area, where the reference area is the planform of the wing simplified into a rectangular shape.

$$AR = \frac{b^2}{S_{REF}}$$

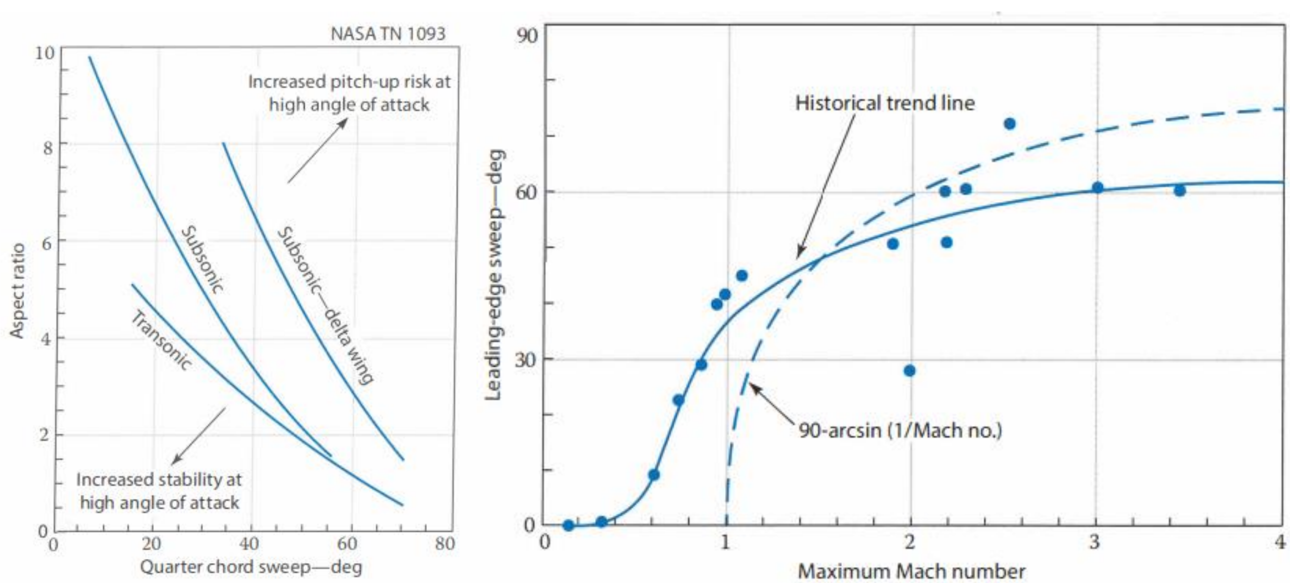
When studying other aircraft with missions like the expected mission of our design, it was found that an aspect ratio of 6 to 9 would be sufficient. For our design, an aspect ratio of 8 was chosen as it was high enough to produce a good lift to drag ratio, but not too high as to impede the maneuverability of the aircraft.

3.4. Wing Sweep, Taper Ratio, Wing Twist, and Other Wing Parameters

When it comes to designing aircraft wings, there are more factors to consider than just the wingspan and area of the wing. Depending on the expected mission of the aircraft, its wing design can vary greatly, so there is no 'one size fits all' wing.

In chapter 4 of the textbook, Raymer provides a figure that displays the relation between aspect ratio and the quarter chord sweep of the wing. In this case the aspect ratio and flight speed of the aircraft should already be known, and using this information the quarter chord sweep angle can be determined. Raymer also provides a figure to show a relation between the aircraft's maximum Mach number and its leading-edge sweep.

Figure 17. Historical Trends for Airfoil Planform



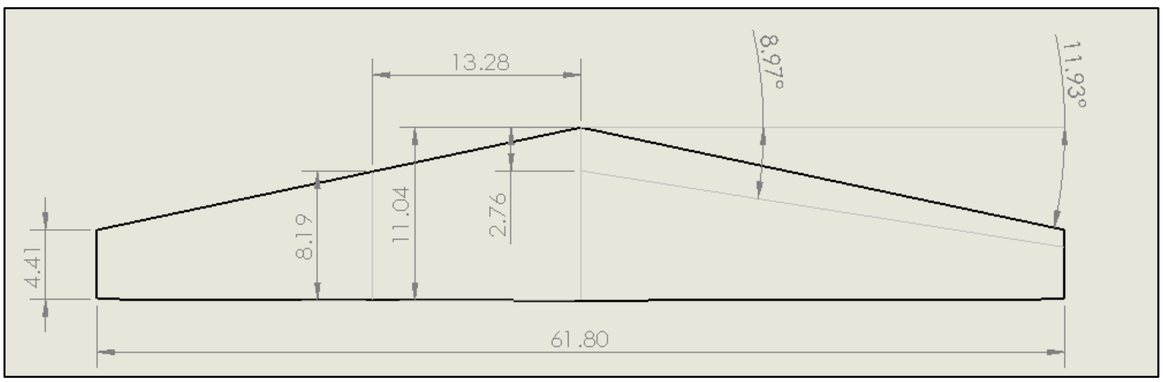
From these figures, it was determined that the wing sweep of the leading edge should be 11 degrees while the quarter chord sweep is 15 degrees. When designing the wing planform, it appears that for a wing with taper ratio 0.4, leading edge sweep of 11 degrees, and quarter chord sweep of 15 degrees is not possible to construct, so only the leading-edge sweep was kept. This results in a quarter chord sweep of roughly 9 degrees.

In the textbook, Raymer also gives suggestions on what the taper ratio should be for a given aircraft. In this case a taper ratio of 0.4 is used. Taper ratio is the ratio of the tip chord of the wing to the root chord.

$$TR = \frac{C_t}{C_r}$$

The taper ratio of an aircraft wing affects the spanwise lift distribution and influences the drag of the wing planform. By knowing the parameters these parameters, we can now visualize the planform.

Figure 18. Hand Sketch Wing Planform



Other wing parameters that cannot be seen from the planform view still influence the flight characteristics of the aircraft. The wing twist angle, or washout, is suggested to be at an angle of -3 degrees by Raymer, while the dihedral angle for a high mounted wing is sufficient at 0 degrees.

4 Important Design Parameters

4.1. Thrust-to-Weight Ratio

The thrust-to-weight ratio is a crucial design parameter for our aircraft as it determines the amount of thrust our engine can generate compared to the weight of the aircraft. It is an important factor in engine selection and determining the maximum takeoff weight of our aircraft. However, since our aircraft is an amphibious aircraft powered by twin-turboprop engines, we will instead be calculating its power-to-weight ratio. To define the engine size of a propellor driven aircraft, the classical term is called the “power loading” ratio as can be seen in chapter 5.2 of the textbook. This ratio is defined as the aircraft weight divided by its power, and it is generally accepted that a power loading ratio can be selected using historical data.

The power-to-weight ratio for cruise is defined using table 5.4 in Raymers text, along with the equation:

$$\frac{P}{W_0} = a \cdot V_{max}^{\{C}$$

The Thrust-to-Weight ratio for cruise is then determined using the following equation:

$$\frac{T}{W} = \left(\frac{\eta_p}{V}\right)\left(\frac{P}{W}\right)$$

We are assuming a 0.8 propellor efficiency based on historical trends. Using these relationships, we are able to calculate the power-to-weight ratio during cruise to be 0.2267 and the thrust-to-weight ratio of 0.1943.

The thrust-to-weight ratio during climb can be defined using equation 5.4 in Raymers textbook:

$$\left(\frac{T}{W}\right)_{takeoff} = \left(\frac{T}{W}\right)_{cruise}\left(\frac{W_{cruise}}{W_{takeoff}}\right)\left(\frac{T_{takeoff}}{T_{Cruise}}\right)$$

Using this relationship, we can calculate the power-to-weight ratio during climb to be 0.3023 and the thrust-to-weight ratio during climb to be 0.2267.

The power required for our aircraft to meet this thrust to weight ratio is 4600kW. We then chose an engine that could meet these requirements for a dual engine aircraft and found that the CT64-820-3 variant of the T64-GE was able to produce 2330kW allowing for a power production of 4660kW for the aircraft.

4.2. Wing Loading

The ratio of an aircraft's weight to the area of its reference wing, known as wing loading W/S, has a significant impact on various aspects of its performance such as stall speed, climb rate, takeoff and landing distances, and turn performance. The design lift coefficient is determined by the wing loading, which in turn affects drag through its impact on the wetted area and wingspan.

We can calculate the maximum wing loading during take-off using equation 5.8:

$$\left(\frac{W}{S}\right)_{takeoff} = (TOP)\sigma C_{L_{TO}}\left(\frac{hp}{W}\right)$$

To do this, the takeoff parameter (TOP) must first be calculated. This is done using figure 5.4 which gave us a result of 140 fps units. The design takeoff lift coefficient must also be calculated using the equation:

$$C_{L_{TO}} = C_{L_{\max flaps}} \cdot 0.8$$

Where $C_{L_{\max flaps}}$ was estimated to be 2.9 from historical values. σ is calculated from the ratio of local density to sea level density. Therefore, the calculated value of the maximum wing loading during takeoff is 77.9688 lb/ft^2. We can calculate the maximum wing loading of the aircraft during cruise by using equation 5.6:

$$\left(\frac{W}{S}\right)_{cruise} = \frac{1}{2}\rho V_{stall}^2 C_{L_{max}}$$

Using this equation, we found the maximum wing loading during cruise to be 42.5186 lb/ft^2. Due to the maximum wing loading being smaller in cruise, this is found to be the driving number for this calculation, and therefore going forward will be used in calculations.

4.3. L/D Ratio

The lift-to-drag ratio determines many different performance parameters of our aircraft including the range, take-off and landing distance, and the thrust-to-weight ratio of our craft. We used equation 3.12 in Raymer's text to define this and we found a value of 11.37 for cruise and 16.0665 for takeoff.

5 Redefined Sizing

5.1 Empty Weight Fraction/Fuel Weight Fraction

Unlike chapter 2 in initial sizing to calculate the empty weight fraction and fuel fraction this improved sizing method uses aspect ratio, thrust to weight ratio (hp to weight ratio), wing loading, maximum speed, and gross weight (design weight from chapter 2)

$$\frac{W_e}{W_o} = a + bW_0^{C1} + A^{C2}\left(\frac{hp}{W_o}\right)^{C3}\left(\frac{W_o}{S}\right)^{C4} V_{max}^{C5}$$

Using Table 6.2 from Raymers Textbook for a twin turboprop aircraft we use the values in the table below

Table 8. Twin Turboprop Empty Weight Fraction

	a	*b*	*C1*	*C2*	*C3*	*C4*	*C5*
Twin Turboprop	.37	.09	-.06	.08	.08	-.05	.30

From here we calculate the fuel weight fraction by taking the mission-segment weight fractions to calculate the fuel burn which is equal to:

$$W_{f_i} = \left(1 - \frac{W_i}{W_{i-1}}\right) W_{i-1}$$

This is then used to calculate the total mission fuel which is:

$$W_{f_m} = \sum_{1}^{x} W_{f_i}$$

We still use the 6% allowance trap fuel which the concludes the fuel weight fraction to be:

$$W_f = 1.06\left(\sum_{1}^{x} W_{f_i}\right)$$

As defined in chapter 2 we are following the same mission profile of warm-up, takeoff, climb, cruise, descent, loiter, and landing. To decide on each mission segments weight fraction, we looked to Raymers textbook for guidance on picking values. For warm-up takeoff Raymer gives a reasonable estimate of .97 to .99. For climb we choose a Mach speed of .39 (about 300 mph) to calculate this mission segments weight fraction using:

$$\frac{W_2}{W_1} = 1.0065 - .0325M$$

The next mission segment is cruise. Deriving Brequet range equation we find that for a propeller driven aircraft cruise mission segment to be:

$$\frac{W_3}{W_2} = \exp\left[\frac{-RC_{power}}{\eta_p\left(\frac{L}{D}\right)}\right] = \exp\left[\frac{-RC_{bhp}}{550\eta_p\left(\frac{L}{D}\right)}\right]$$

During cruise our lift-to-drag ratio will be equal to weight due to the assumption of steady-level flight. This allows us to use:

Duck:

$$\frac{L}{D} = \frac{1}{\frac{qC_{D_o}}{\frac{W}{S}} + \frac{W}{S}\frac{1}{q\pi Ae}} = 11.37$$

L/D is tested at an altitude of 25,000 ft and a cruise velocity of 250 knots

Goose:

$$\frac{L}{D} = \frac{1}{\frac{qC_{D_o}}{\frac{W}{S}} + \frac{W}{S}\frac{1}{q\pi Ae}} = 15.7$$

L/D is tested at an altitude of 20,000 ft and a cruise velocity of 250 knots

Duckling:

$$\frac{L}{D} = \frac{1}{\frac{qC_{D_o}}{\frac{W}{S}} + \frac{W}{S}\frac{1}{q\pi Ae}} = 7.4516$$

L/D is tested at an altitude of 15,000 ft and a cruise velocity of 300 knots

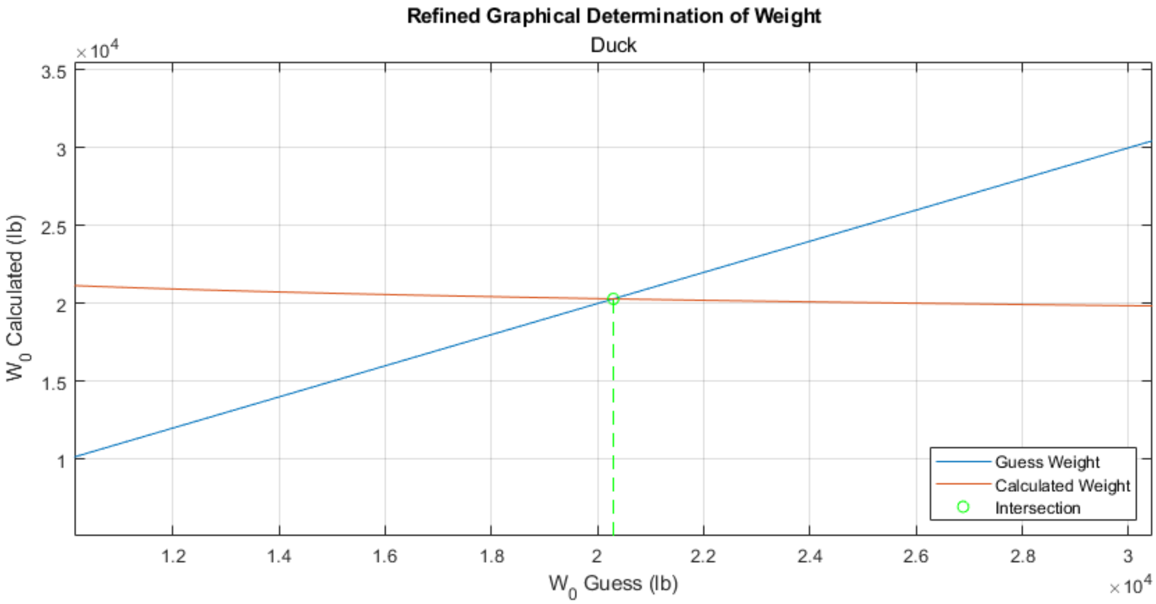
Loiter's mission segment is then calculated by:

$$\frac{W_4}{W_3} = \exp\left[\frac{-EVC_{power}}{\eta_p\left(\frac{L}{D}\right)}\right] = \exp\left[\frac{-EVC_{bhp}}{550\eta_p\left(\frac{L}{D}\right)}\right]$$

For descent and landing Raymer suggests approximating both in a range from .990 to .995 for descent and .992 to .997 for landing.

5.1.2. Duck Mission

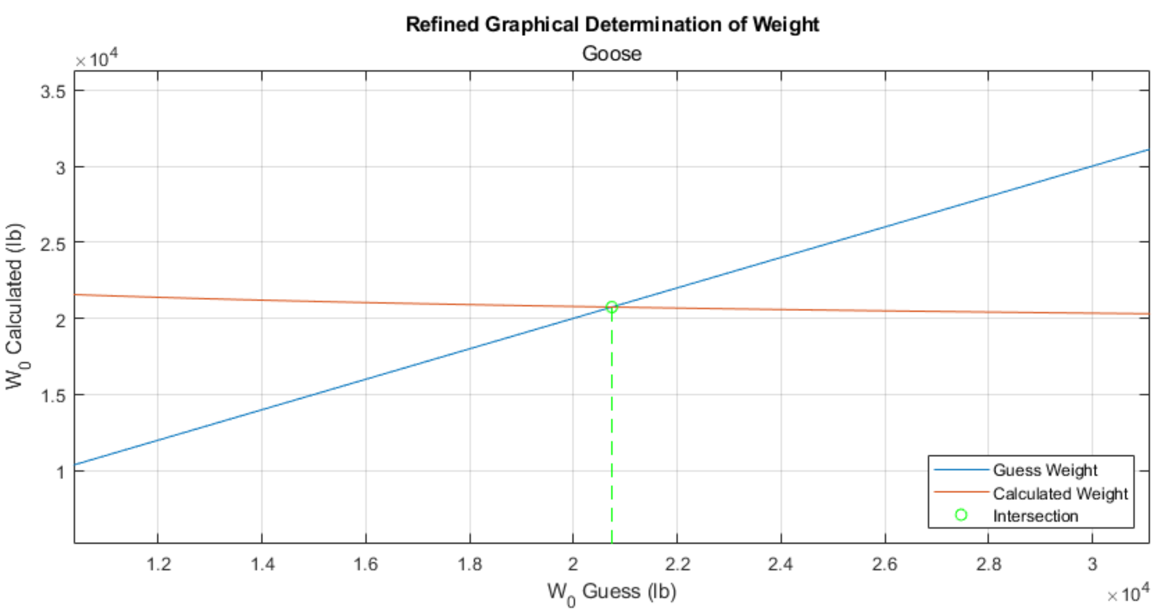
Figure 19.



$$Duck\ Refined\ Design\ Weight = 20{,}298\ lbs.$$

5.1.3. Goose Mission

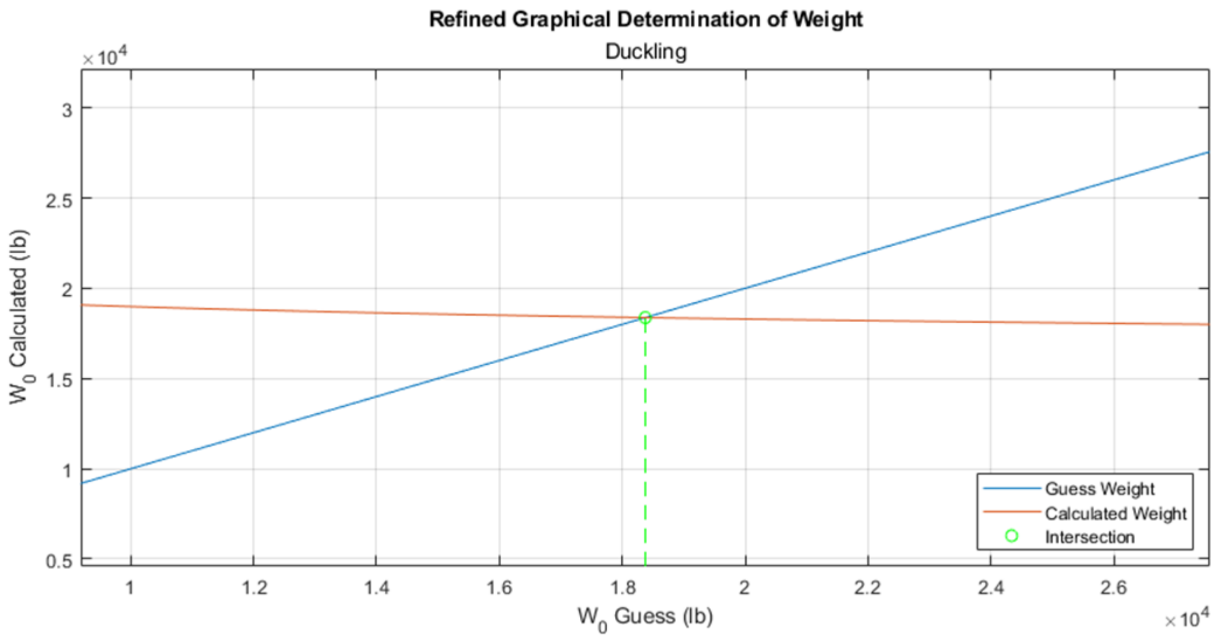
Figure 20.



$$Goose\ Refined\ Design\ Weight = 20{,}740\ lbs.$$

5.1.4. Duckling Mission

Figure 21.



$$Duckling\ Refined\ Design\ Weight = 17{,}580\ lbs.$$

5.1.5 Conclusion

Comparing the refined design weight from the initial design weight we see that they are slightly higher than the initial values, but they are within the 5% error range. This allows us to confirm that our aircraft is within the performance range. This allows us to move on to the next section of design.

5.2 Geometry Sizing

5.2.1 Fuselage Length

According to Raymers textbook we can use Table 6.3 (Fuselage Length vs Wo) to determine the length of the fuselage from our redefined sizing calculations.

$$Length\ =\ aW_o^c$$

The a and c values come from the twin turboprop aircraft and are defined as $a = .37\ and\ C = .51$ which gives a fuselage length of 58 ft. We have chosen to have a fineness ratio of 7 due to our design being a subsonic aircraft.

5.2.2 Tail Volume Coefficient

Using Raymer's textbook again we can find the tail volume coefficient of the horizontal and vertical tail from Table 6.4. Examining twin turboprop aircraft, we see those typical values for the horizontal to be .9 and for the vertical to be .08. The tail arm is going to be about 50%-55% fuselage length due to our design having the engines on the wing.

5.2.3 Control Surface Sizing

Control surfaces are used to adjust the aircraft's attitude. Control surfaces include ailerons, elevators and rudders, these surfaces maintain the yaw, pitch and roll of the aircraft, shown in figure 19. Roll and yaw are coupled derivatives and need to be addressed. At low speeds adverse yaw, the yaw that happens when the airplane rolls, can be addressed with frise or differential ailerons shown in figure 21. At high speeds aerodynamic loads on control surfaces cause aileron reversal, which causes the wing to twist and cause a roll in the opposite direction. Ailerons are typically 50%-90% of the wingspan and 15%-20% of the wing chord. The required aileron area can be approximated using figure 20. Elevator and rudder control surfaces typically extend from the fuselage to tip of the tail span, with a length of 25%-50% of the tail chord. At higher speeds, rudders can extend to 50% of the span to avoid effectiveness issues similar to aileron reversal. Control surfaces need to be balanced to avoid flutter which is caused by oscillations in the control surface due to a mass imbalance. To balance the mass of the control surface, weight is added ahead of the control surface hinge line to offset the weight aft the hinge line. The hinge axis is always less than 20% of the aft the average chord of the control surface.

Figure 22. General aileron determination guidelines.

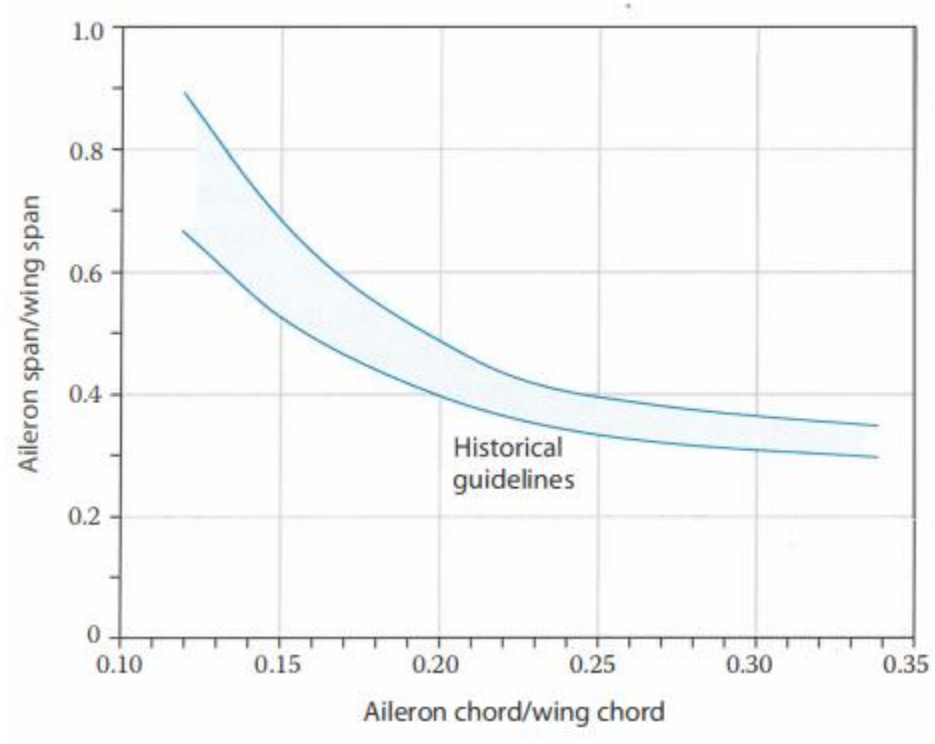


Figure 20. General Control surfaces. a) F-4B phantom. b) T-28B

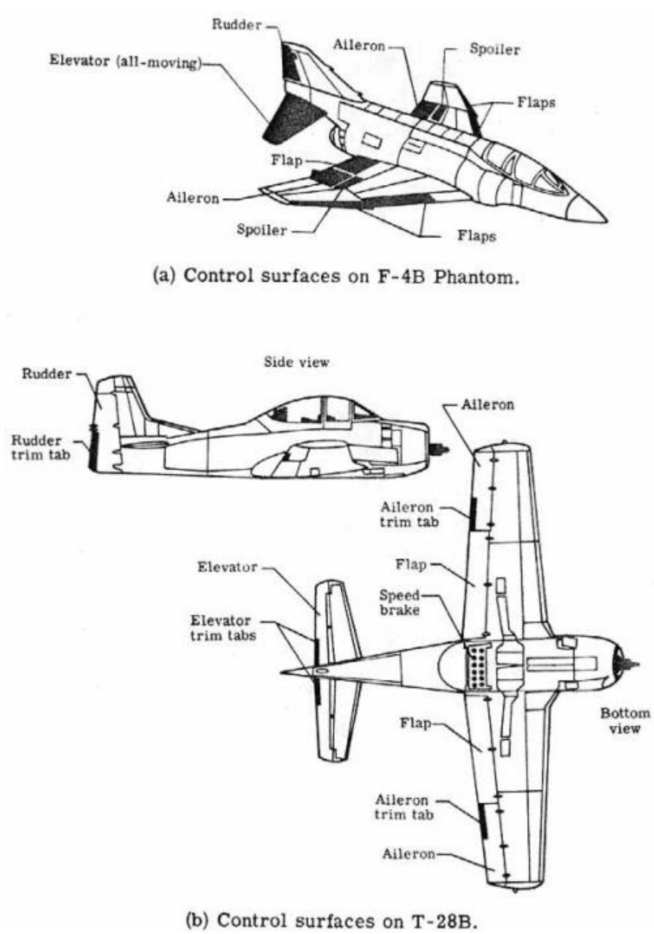
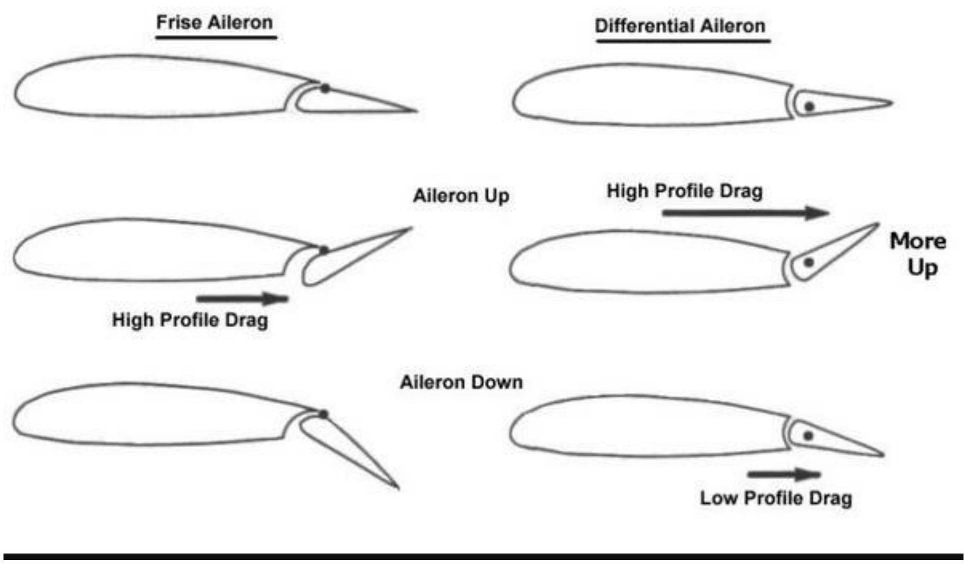


Figure 21. Frise and differential ailerons to correct adverse yaw.



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