MMAE 414: Aircraft Flight Design

High Capacity Short Range Transport Aircraft Design Proposal

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Contents

Nomenclature

C_l lift coefficient of 2-D airfoil

C_L lift coefficient of 3-D wing

C_l roll moment coefficient

C_m pitch moment coefficient

C_n yaw moment coefficient

$C_{M\alpha}$ pitch moment-alpha curve slope

$C_{L\alpha}$ lift-alpha curve slope

SM Static Margin

X_{NP} neutral point

X_{CG} Center of Gravity

W/S Wing loading

T/W thrust to weight ratio

TOP Takeoff parameter

V_{mp} velocity minimum power

1 Executive Summary

This report contains the planning, organization, design process, analysis and final preliminary design of a high capacity short range transport aircraft. Our objective was to address the growing popularity in commercial air travel and allow an option for those wishing to travel into smaller, less congested airports. The plan would be to have this aircraft enter service in 2029 with a passenger capacity of 400 and a range of 3,500 nautical miles. With the growing popularity in air travel we believe that a high capacity short range aircraft will have an unmet growth due to the lack of historical trends combining large seat count and short range.

1.1 Key mission requirements

Along with the general design goal of combining short range and large capacity, other design parameters were also considered. Particularly our mandatory requirements for this aircraft are:

- Capable of taking off and landing on both asphalt or concrete runways
- Capable of VFR and IFR flight with autopilot
- Capable of flight in known icing conditions
- Meets applicable certification rules in FAA CFR Part 25
- 400 passengers in a dual class configuration
    - 50 passengers in Business class with 36” pitch, 21” width
    - 350 passengers in Economy class with 32” pitch, 18” width
    - 5 cubic feet per passenger for baggage
    - Galleys, Lavatories, and Exits to meet 14 CFR Part 25
    - Number of aisles appropriate to the passenger layout
    - Average 230lbs per person, crew and passengers, including luggage
- 3,500 nmi range with proper energy reserve requirements via 14 CFR Part 25
- Maximum takeoff length of 9,000 ft over a 35ft obstacle with dry pavement, sea-level ISA, 15 degrees C at maximum takeoff weight
- Price of Jet-A fuel is \$3/gallon plus \$3/gallon carbon tax

Along with these requirements the team’s goal is to minimize the operating cost and production cost in order to make the aircraft both marketable and profitable. As well as making the aircraft reliable and cut down on maintenance time.

2 Management Summary

Our team consists of five members with varying experience in research and industrial work all with a solid background education. The variety of experiences lends itself to continuous improvement in knowledge and ensures the team is constantly in development. Each member has selected their own expertise based on prior experience and interest as well as their ability to work closely with team leaders from dependent teams. This team met in person weekly for at least an hour to ensure all progress was steady as well as to share our findings to standardize procedures across the board. All members were active and reliable throughout the project.

2.1 Team organization

Component	Owner	Back-up Owner
Aerodynamics, Performance, Stability, and Control	*Titilayo Fasoro*	*Benjamin Thrun*
Configuration, CAD, and Dimensions	*Brett Ott*	*Zachary Whritenour*
Structures and Loads	*Zachary Whritenour*	*Brett Ott*
Propulsion	*Benjamin Thrun*	*Titilayo Fasoro*
Costs	*Savannah Kelley*	*None*
Report Finalization	*Savannah Kelley*	*All*

Table 2.1: Team Responsibilities

3 Conceptual Design Process

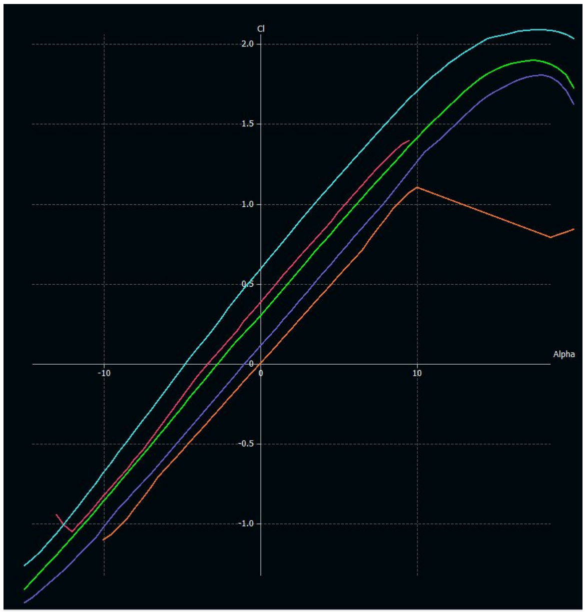
With the RFP in mind, as well as keeping in compliance with 14 CFR Part 25. The team approached decision making by incorporating many trade studies to have a wider appreciation for the options at hand. Initially a trade study was performed to choose an airfoil which would best suit the mission requirements. Upon selecting the NASA SC(2)-0714 airfoil, the team performed another analysis in order to find the best engine for the aircraft. With a fixed engine and initial sizing performed, the team performed another trade study in order to find an optimal thrust-to-weight and wing loading parameters.

3.1 Aerodynamics

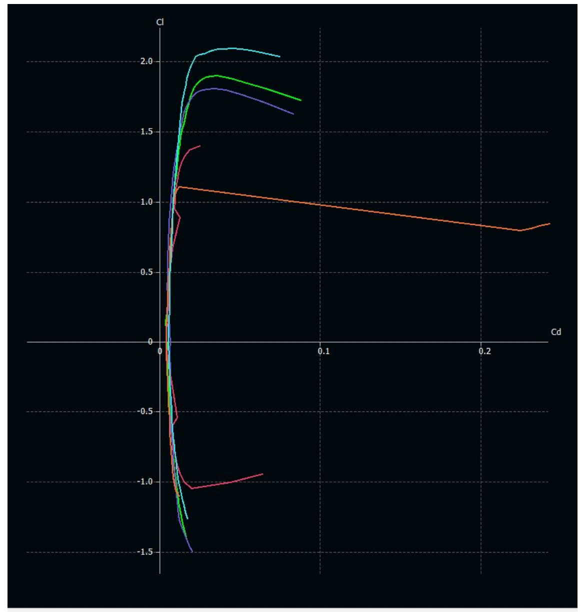
3.1.1 Aerofoil selection

3.1.1.1 Wing

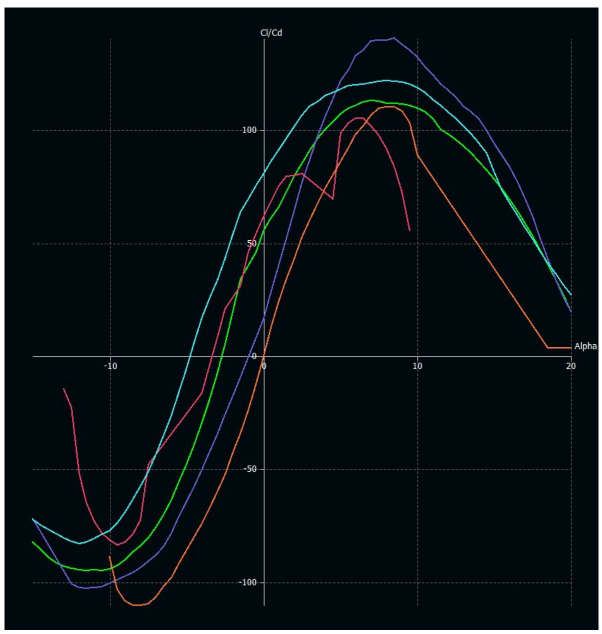
The choice of airfoil was a key determinant in the aircraft design process that affected the cruise and stall speeds, take-off and landing distances and performance of the airplane. At the beginning of our airfoil selection process for the wing, airfoils similar to representative designs listed in the proposal were compared using computational solutions of an analytic tool, xflr5. The maximum lift coefficient, maximum lift to drag ratio and stall angle were considered in our analysis. Due to limitations of the tool, only subsonic speeds less than 0.2 Mach were used in the analysis. The figures below show the variation of parameters for each airfoil.



(a) Cl versus alpha curve



(b) Cd versus Cl curve



(c) Cl/Cd versus alpha curve

Figure 3.1: Aerodynamic characteristics for each airfoil from xflr5
NASA SC(2)-0714(cyan); NACA 23112(violet); NASA SC(2)-0606(red);
BOEING AIRFOIL J (green); Ryan BQM-34(orange)

After much examination, the supercritical airfoil SC(2) 0714 was selected. It showed the highest lift coefficient with an acceptable lift to drag ratio. Being a supercritical airfoil, one of the remarkable characteristics it displayed was its ability to minimize the effects of transonic speed by increasing the critical Mach number. The image below shows the airfoil cross-section.

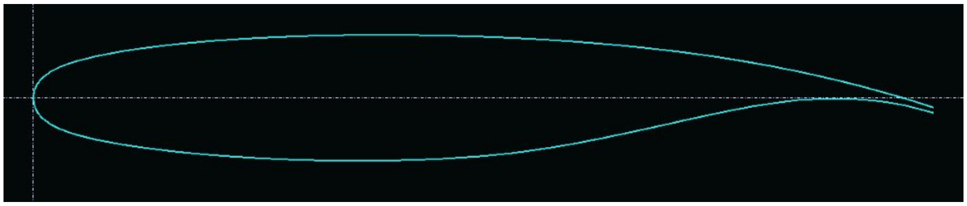


Figure 3.2: NASA SC(2)-0714 airfoil

3.1.1.2 Tail

Empennage airfoils are usually symmetric with 0009 and 0012 being the most common. The NACA 0010 airfoil was selected for both the horizontal and vertical stabilizers respectively. Since the horizontal tail was required not to stall, its stall properties were not required.

3.1.2 Wing Area and Geometry

The area of the wing was determined from the wing loading and takeoff weight calculated during the initial design process as described in the performance section. This was determined to be 4471 sq. ft, a value similar to many conventional commercial aircrafts.

The shape of the reference wing was determined from its aspect ratio, taper ratio, and sweep. The initial selection of these parameters was based on historical trends of existing aircrafts and statistical values provided by Raymer[4].The wing geometry parameters selected for this design are listed under the configuration section.

Our wing also features a low wing design which is most common to commercial aircrafts. Additionally, winglets are added to reduce drag and increase the lift to drag ratio.

3.2 Propulsion

3.2.1 Selection

When considering the propulsion system that should be used for this mission. There were seven engines which were compared, these engines were inspired from aircraft with similar missions. These engines are as follows:

- Rolls Royce Trent 700
- Rolls Royce Trent 800
- Rolls Royce Trent 900
- General Electric CF6-80E1
- General Electric GE90 -76B
- General Electric GE90 -110B1
- Pratt & Whitey PW4000

After performing a trade studies shown in section 3.2.4, the Rolls Royce Trent 800 was chosen for the mission.

3.2.2 Sizing

The dimensions for the Trent 800 are as follows;

Length: 4.37 m (172 in)
Diameter: 2.79 m (110 in)
Dry weight: 7,484 kg (16,500 pounds)

3.2.3 Airframe Integration

The placement of engines along the wing is critical to performance. For this project the decision was made to choose the engines to be below the wing at the respective location with the drawing. The engines were chosen to be forward and beneath the wing. The reason for the engines to be ahead of the wing is to reduce flutter which is an oscillation caused by the elastic and aerodynamic forces. Another reason for the engine to be ahead is so that the flow going into the engine is less likely to be disturbed by the airplane then if it was set back on the wing. The reason for the engine to be below the wing and instead of on top comes from maintenance and accessibility to work on or repair the engines for the ground crew. Stability is also another concern and having the engines on top of the wing would be less stable than having them below the wing. If the engine was on top of the wing it would want to tip over.

3.2.4 Trade Studies

The data gathered on the engines listed in section 3.2.1 is as follows:

Engine Name	**Rolls-Royce Trent 900**	**Rolls-Royce Trent 700**	**General Electric CF6-80E1**	**Pratt & Whitney PW4000**	**Rolls-Royce Trent 800**	**General Electric GE90 -76B**	**General Electric GE90 -110B1**
Cost (US$)	25,000,000	23,250,000	12,200,000	16,000,000	22,000,000	27,500,000	27,500,000
Weight (lb)	13,770	13,580	11,225	9,420	16,500	17,400	19,316
Thrust to Weight	5.5	5.105	6.04	5.935	5.6	5.59	5.98
Takeoff Thrust (lbf)	75,000	69,300	67,800	56,000	84,200	89,185	113,150
Cruise TSFC (lb/lbf/h)	0.522	0.562	0.385	0.353	0.56	0.545	0.545
Engines Needed	4	2	4	4	2	2	2
Year Used	2004	1990	1994	1986	1996	1993	1993

Table 3.1: Values found for the respective engines.

	Desired Specifications	"Weight" for Conditions
Cost	21,000,000	0.35
Weight	14,000	0.15
Thrust to Weight	6	0.2
Takeoff Thrust	79,200	0.2
Cruise TSFC	0.446	0.1

Table 3.2: Desired Specifications and weighted values used for the trade study

Engine Name	Rolls-Royce Trent 900	Rolls-Royce Trent 700	General Electric CF6-80E1	Pratt & Whitney PW4000	Rolls-Royce Trent 800	General Electric GE90 -76B	General Electric GE90 -110B1
Cost (US$)	0.147	0.316	0.301	0.229	0.334	0.267	0.267
Weight (lb)	0.076	0.154	0.093	0.111	0.127	0.121	0.109
Thrust to Weight	0.194	0.222	0.187	0.191	0.202	0.203	0.189
Takeoff Thrust (lbf)	0.189	0.175	0.171	0.141	0.212	0.225	0.285
Cruise TSFC (lb/lbf/h)	0.081	0.079	0.116	0.126	0.079	0.082	0.082
Engines Needed	0.687	0.947	0.869	0.799	0.956	0.898	0.933

Table 3.3: Non-Dimensional weighted values for the engines based on the trade study conditions

The decision for the weighted conditions was one rooted in a primary concern for cost. Since these engines have similar values for TSFC, this was the lowest priority in terms of “weighted” condition. Weight, Thrust to Weight, and Takeoff Thrust all have some values and distinction which were determined appropriately. When the Trade study was completed, it was shown that the Rolls Royce Trent 800 would be the best propulsion system going forward.

The Rolls Royce Trent 800 also offers a solution to the requirement of the engine being in service until 2029. The Trent 800 has been around since 1996, it is safe to assume the Trent 800 will still be used by 2029 due to its popularity and usage on the Boeing 777.

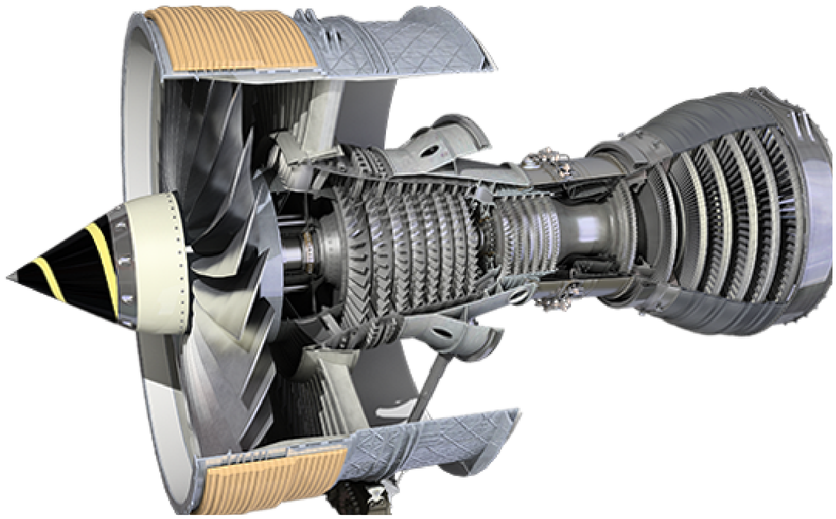


Figure 3.1: Depiction of Rolls Royce Trent 800 adapted from reference [5]

3.3 Stability and Control

Stability analysis of this aircraft involves its angular derivatives. The angle of attack has very little effect on the roll and yaw moments, hence the longitudinal(pitch) and lateral (roll and yaw) stability analyses.

3.3.1 Static Margin

The static margin is the distance in percent MAC between the aircraft's neutral point and centre of gravity which is expressed in the equation below.

$$SM = \frac{(X_{NP} - X_{CG})}{C_{mac}} * 100\% = -\frac{C_{M\alpha}}{C_{L\alpha}}$$

At the most aft c.g position for this aircraft, the static margin was given to be about 5% which falls in the range for transport jets given by Raymer[4]

3.3.2 Longitudinal Stability

To attain static pitch stability, the pitch moment derivative must be negative, i.e there must be moments that oppose a given change in angle of attack. To obtain the pitch derivative from xflr5, the aircraft wing and tails were put together with their airfoil cross-sections, weights and sizes. After completing analyses for the 2-D airfoil cross sections at reynold's number ranging from 100,000 to 9,000,000, a type 1 analysis at zero sideslip and a velocity of 50mph was performed for the 3-D aircraft. The pitch derivative for this aircraft was shown to be -0.037 and the moment at zero angle of attack was 0.14.

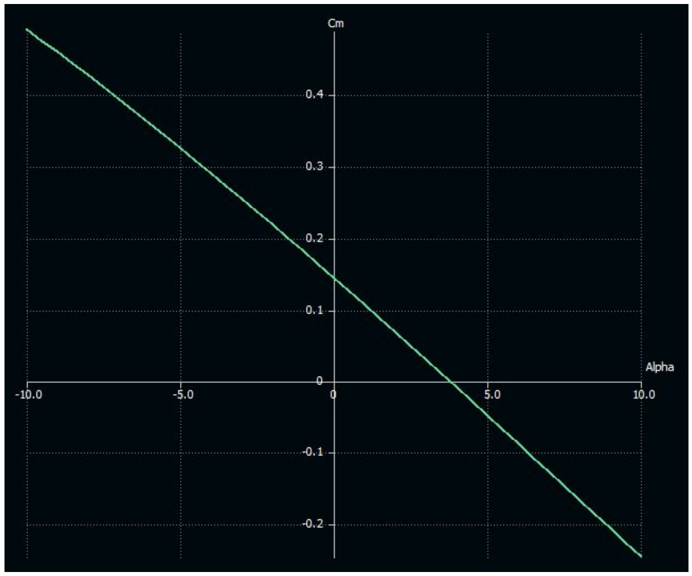


Figure 3.1: Cm versus alpha curve at zero sideslip from xflr5

3.3.3 Lateral - Directional Stability

The lateral-directional analysis involves the yaw and roll moments. Both moment terms are impacted by the sideslip angle beta and are not directly affected by the pitch or roll angles. The yaw and rolling moment derivatives with respect to sideslip as examined in xflr5 are presented below:

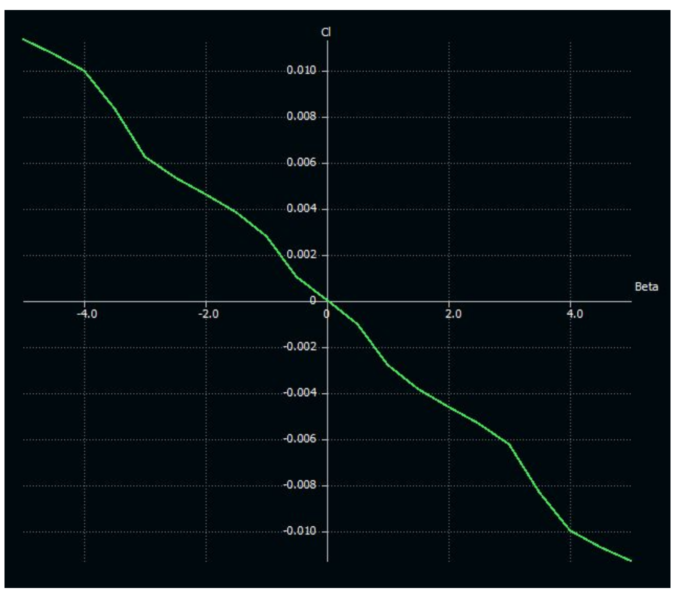


Figure 3.2: Cl versus beta from xflr5

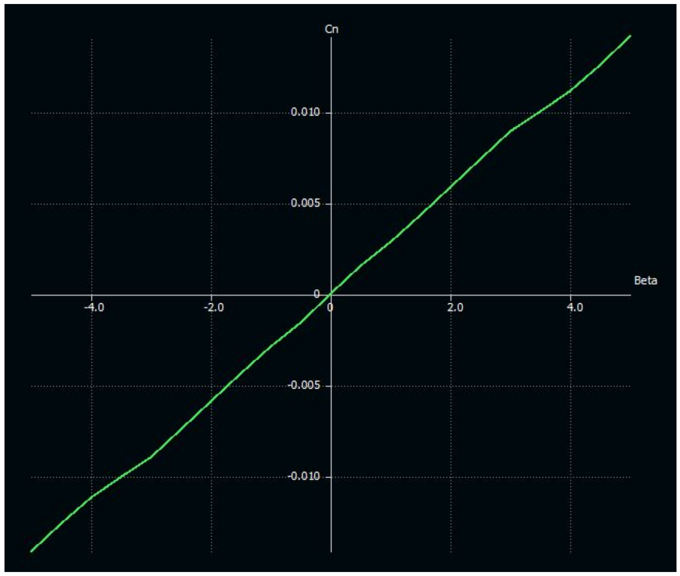
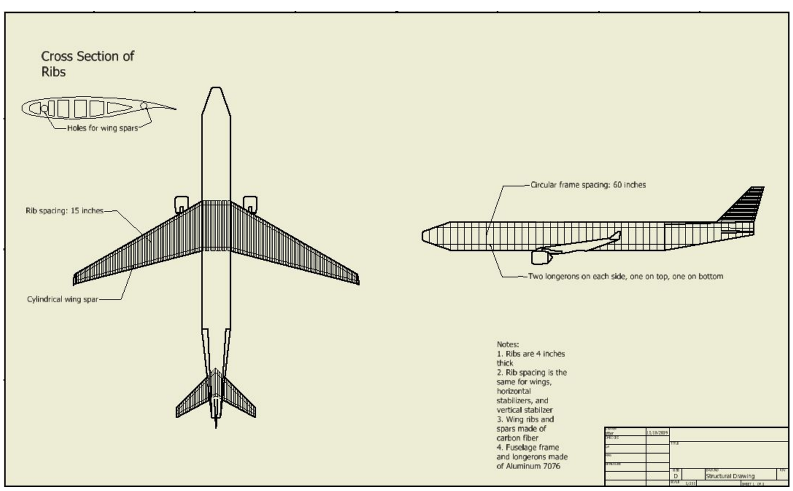


Figure 3.3: Cn versus beta from xflr5

According to Raymer [4], the roll derivative should be of negative sign with a magnitude about half that of the yaw derivative value at subsonic speeds, and about equal to it at transonic speeds. Additionally, he presents typical yaw-moment derivative values. The yaw and roll moment derivatives were found to be about –0.13 and 0.16. These values were well in range with the typical derivatives for such aircraft. However, it should be noted that these values were calculated at low speeds due to the limitations of the computational tool. Further detailed and precise analysis will require more sophisticated tools.

3.4 Structure



3.4.1 Weight Statement

In this section of the report the individual components that make up the structural mass of the aircraft will be discussed. The reason for each decision will also be validated.

3.4.1.1 Propulsion

The engine we decided to go with was the Rolls-Royce Trent 800. The weight of this engine is 16,500 lbs. Our aircraft requires 2 of these engines for required performance. Refer to section 3.2.1 for validation

3.4.1.2 Landing Gear

This aircraft will use the same landing gear as the Boeing 777. It will have two rear bogies with 6 weels per bogie. It will have one forward strut with 2 wheels. This will provide ample force and braking power for the weight and speed of our aircraft.

3.4.1.3 Wing

The weight of the wing was calculated from the size of the wing, using equations from Raymer's book(4).

3.4.1.4 Empennage

The weight of the Empennage was calculated from the size of the wing, using equations from Raymer's book(4).

3.4.1.5 Fuselage

Weight for individual parts of the aircraft such as Fuselage, Empennage, Wing, Vertical/Horizontal Stabilizers, Seats, Controls, ect. Were found from equations from reference(4) as well as historical data from past aircraft.

Structure	Weights (lbs)
Engines	33000
Wing	30171
Fuselage	87036
Vertical Stabilizer	3747
Horizontal Stabilizer	11609
Landing Gear	17919
Other (Seats, Controls, etc)	55295
Empty Weight	238777
Fuel	165162
Operating Weight	403939
Payload	105800
Take Off Weight	509739

Table 3.4.1: Breakdown of weights by structure

3.4.2 Payload Range

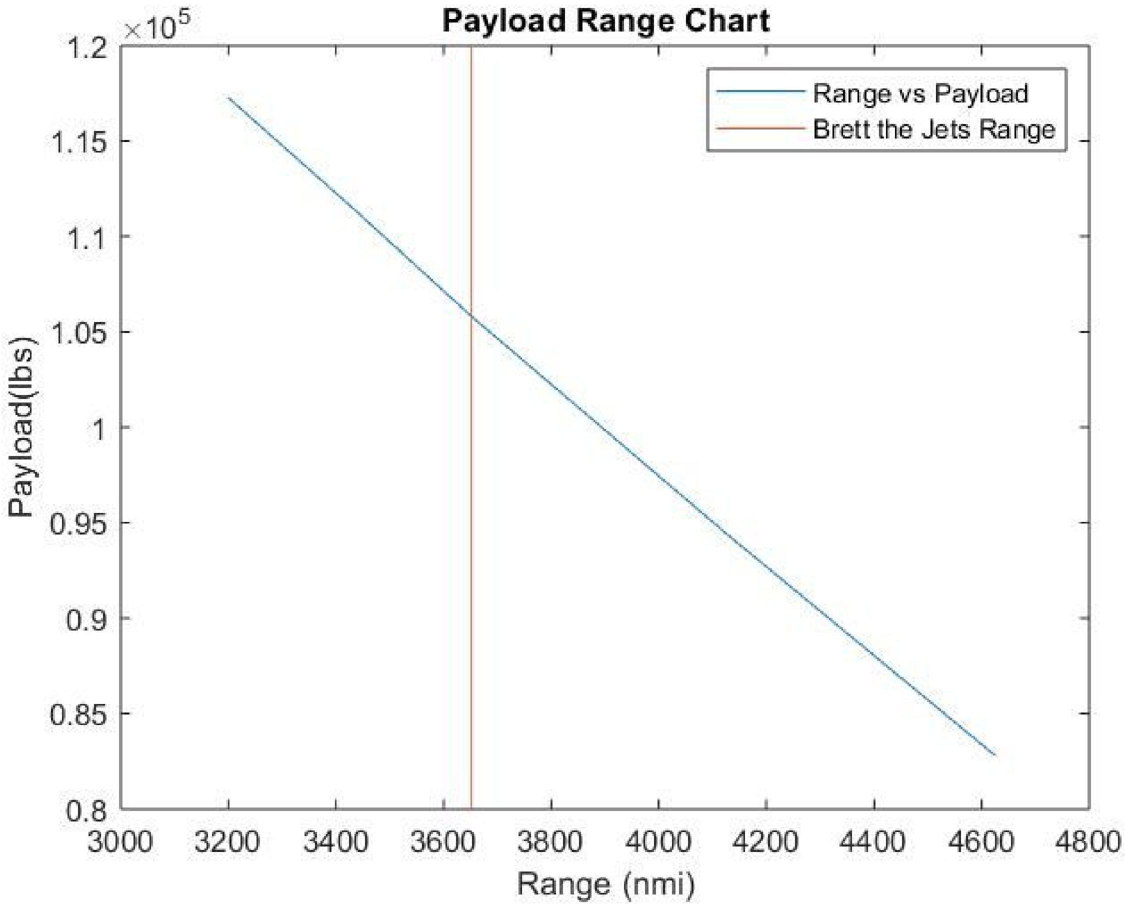


Figure 3.4.1: Graph of the Range vs Payloads with Brett the Jets Range shown.

3.4.3 Materials

Aluminium makes up a large portion of the aircraft's weight. The skin of the aircraft, as well as the structure of the fuselage, vertical and horizontal stabilizers are all made of aluminum. The type of aluminum used is alloy 7075. This particular alloy is high-strength and has zinc and copper added to it to increase ultimate strength.The copper makes it easy to weld. All together this alloy is easiest to work with and has the best strength for flight applications. Carbon fiber is used for the ribs and spars in the wings for added strength as well as weight benefits.

3.4.4 Cabin Pressurization

The cabin will be pressurized to roughly 11-12 psi. This is the pressure seen at around 8000 ft. The reason for this is to try and minimize the difference between the internal and external pressure of the cabin. Air will be constantly supplied to the cabin through the compressor and two output valves at the rear of the aircraft will regulate the pressure. This pressure also allows the structure of the aircraft to comfortable handle the associated loads.

3.5 Configuration

The dimensions and configuration of the aircraft are of the utmost importance to the overall design. This will affect everything from weight to aircraft performance to seating layout. We began the design with the assumption that the aircraft would have a wingspan of 196.86 ft (60 m) to ensure that it could operate at most major airports in the world. The rest of the dimensions were chosen based around that assumption. Some parameters were decided based on historical data, but shown below are some of the equations used to calculate various dimensions of the aircraft:

$$D_{fuselage} = \frac{L_{fuselage}}{fineness\ ratio}$$

$$c_t = \lambda c_r$$

$$c_{avg} = \frac{c_t + c_r}{2}$$

$$S = bc_{avg}$$

$$AR = \frac{b^2}{S}$$

$$MAC = \frac{1+\lambda+\lambda^2}{1+\lambda}\frac{2}{3}c_r$$

$$\overline{Y} = \frac{b}{6}\frac{1+2\lambda}{1+\lambda}$$

$$c_{VT} = \frac{L_{VT}S_{VT}}{b_W S_W}$$

$$c_{HT} = \frac{L_{HT}S_{HT}}{C_W S_W}$$

Shown below are two isometric three-dimensional views of the aircraft. These are shown without windows, as their purpose is solely to give a proportionally-sized physical representation of the aircraft.

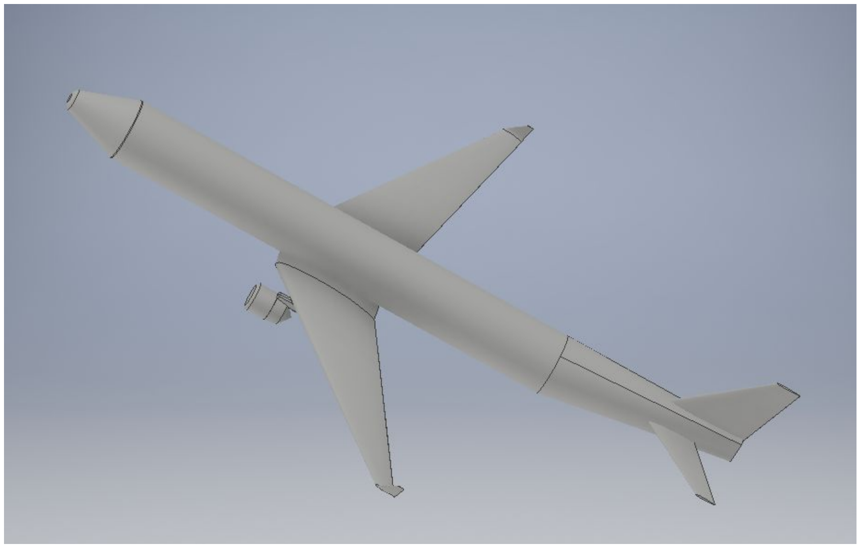


Figure 3.5.1: Isometric 3D view of the aircraft

Figure 3.5.2: Isometric 3D view of the aircraft

The table below is a summary of the geometry of the aircraft's lifting and stabilizing surfaces. It is useful to have the geometry specifications of the main wing, horizontal stabilizer, and vertical stabilizer all in one place.

	Main Wing	**Horizontal Stabilizer**	**Vertical Stabilizer**
Area (ft^2)	4471.39	884.532	472.456
Wingspan (ft)	196.86	66.161	26.621
Aspect Ratio	8.67	4.95	1.50
Taper Ratio	0.2	0.3	0.3
Root Chord (ft)	35	22.619	27.304
Tip Chord (ft)	7	6.786	8.191
Mean Aerodynamic Chord (ft)	25.838	16.123	19.463
Quarter-Chord Sweep (degrees)	30	40	45
Dihedral (degrees)	5	0	0
Twist (degrees)	5	0	0

Table 3.5.1: Dimensional data of the main wing and stabilizers

3.5.1-3.5.3 Front, Left, and Top View

While three-dimensional models give a nice visual representation of the aircraft, two-dimensional drawings are much more important. They are annotated to show the dimensions of the aircraft, which gives significantly more information than 3D models.

Shown below is a detailed dimensional drawing of the top, front, and left views of the aircraft:

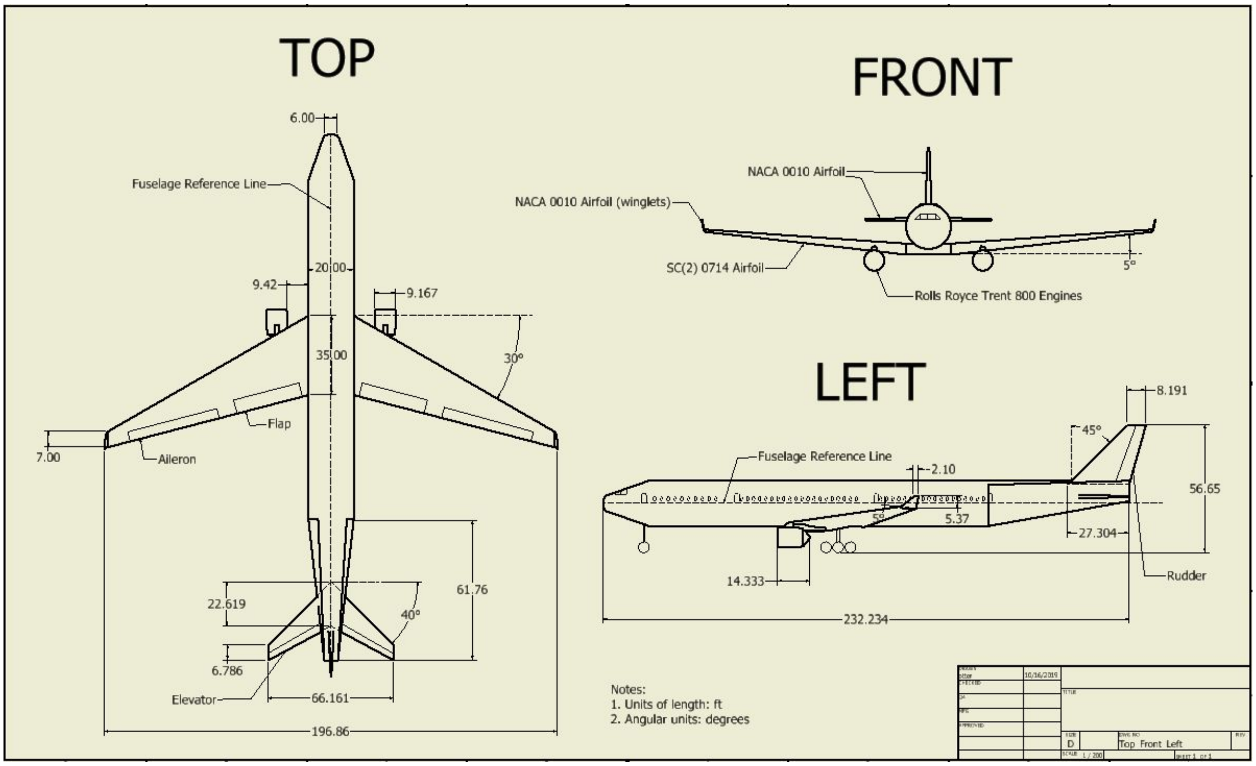


Figure 3.5.3: Dimensional drawing of the top, front, and left views of the aircraft

3.5.4-3.5.6 Aerodynamic Center, Center of Gravity, and Tail Moment Arms

The tail moment arms were calculated from the equations referenced earlier in this section. It was assumed, based on normal trends, that the quarter-chord of the main wing is the aerodynamic center and that the center of gravity is located just aft of the x-location of 30 percent of the mean aerodynamic chord. The location of these parameters are shown in the figure below:

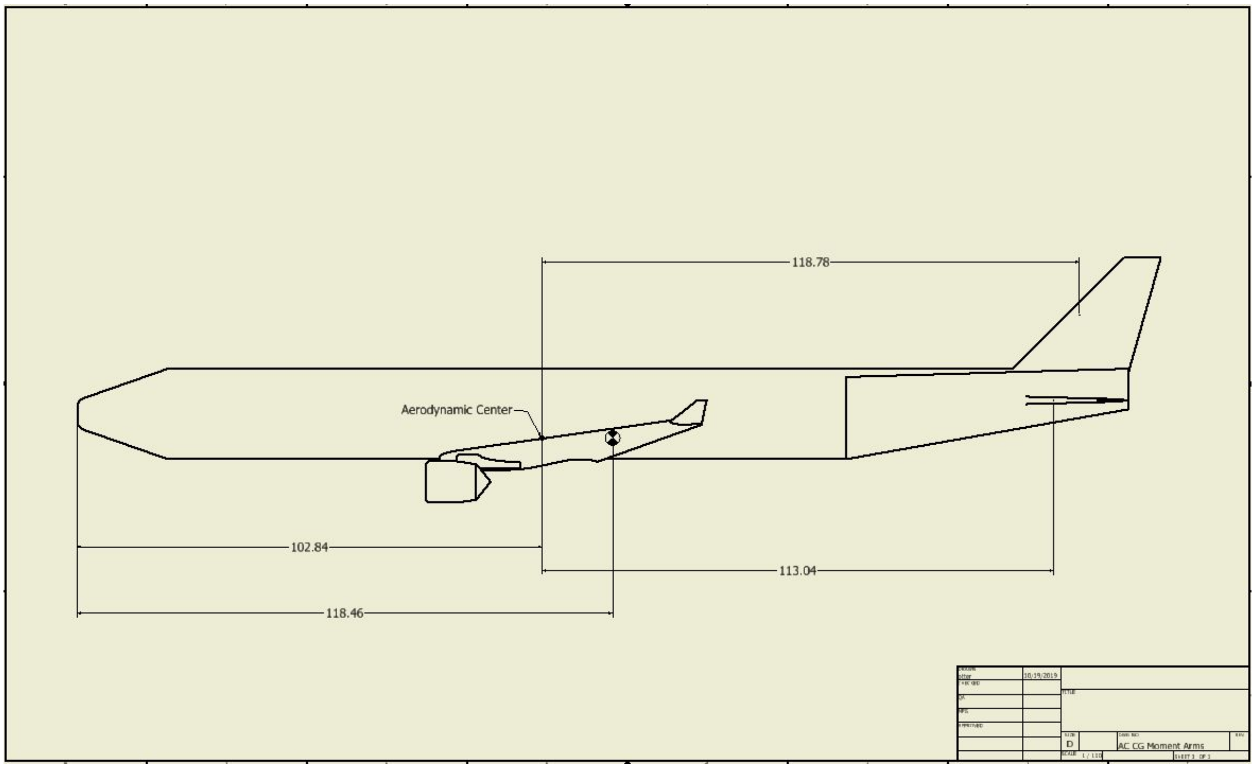


Figure 3.5.4: Dimensional drawing of the locations of the aerodynamic center, center of gravity, horizontal tail moment arm, and vertical tail moment arm

3.5.7-3.5.8 Cross Section/Cabin Layout/Cargo Layout

This aircraft has a maximum passenger capacity of 450, with 70 in business class and 380 in economy class. Business class seats have a pitch of 36 inches and a width of 21 inches. Economy class seats have a pitch of 32 inches and a width of 18 inches. The seating chart is shown in Figure 3.5.5 below, including seats, emergency exits, galleys, and lavatories per FAA regulations. Figure 3.5.6 shows a cross-sectional view of the seats and cargo areas for both business and economy classes.

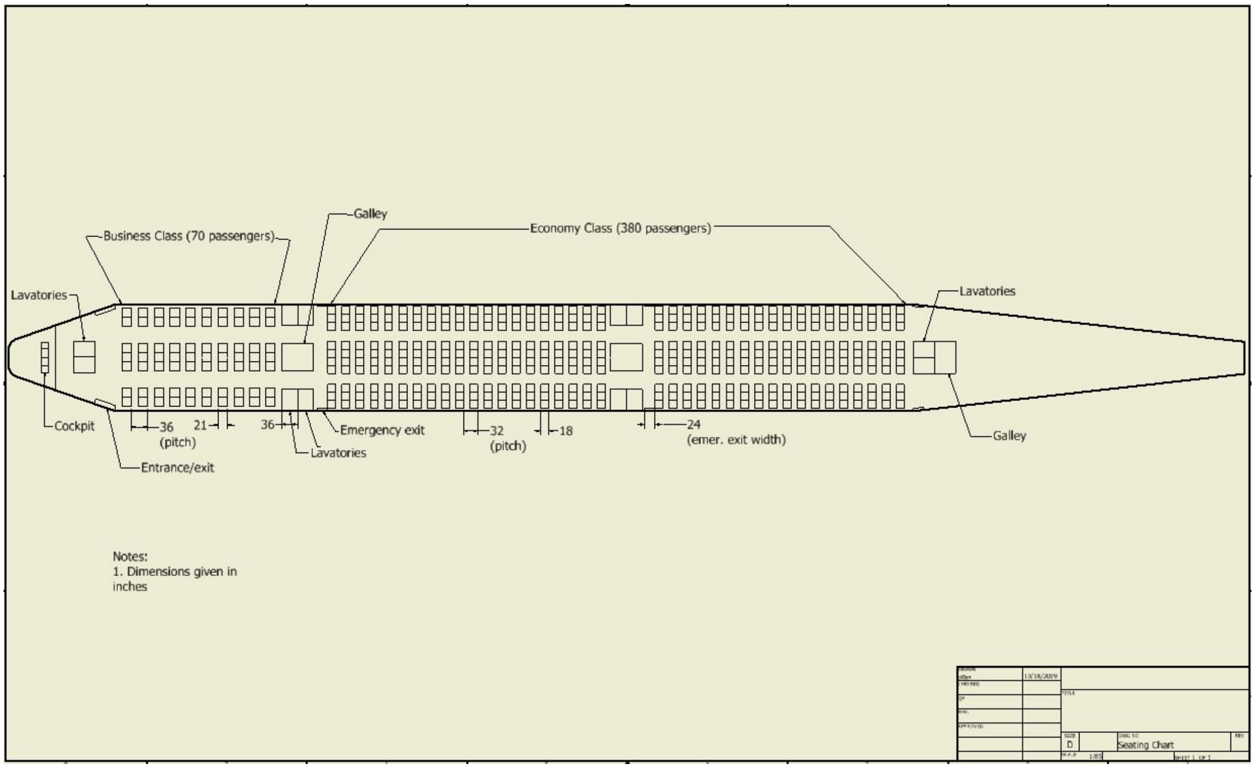


Figure 3.5.5: Aircraft seating chart

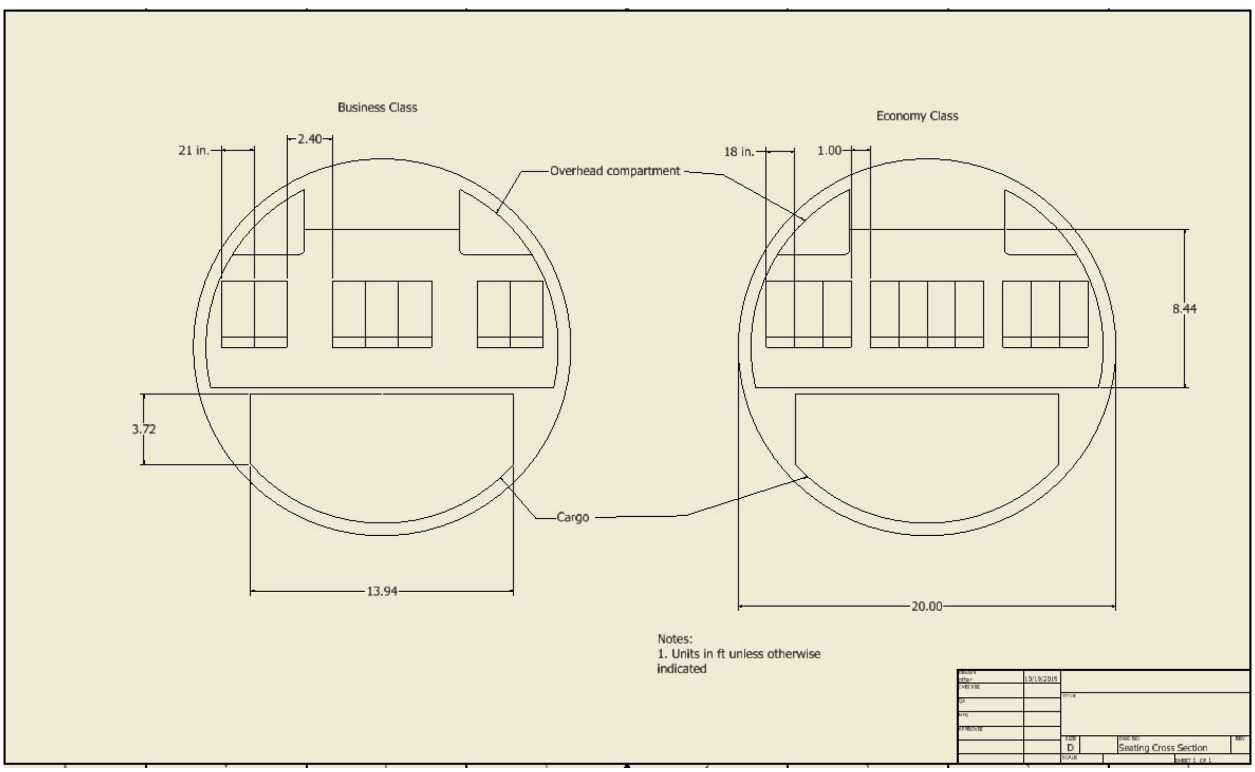


Figure 3.5.6: Layout of seats and cargo for business and economy classes

3.5.9 Layout of Cockpit

Shown below is a simple diagram of the cockpit layout. This includes a key identifying the instruments, controls, and displays available to the pilots during flight.

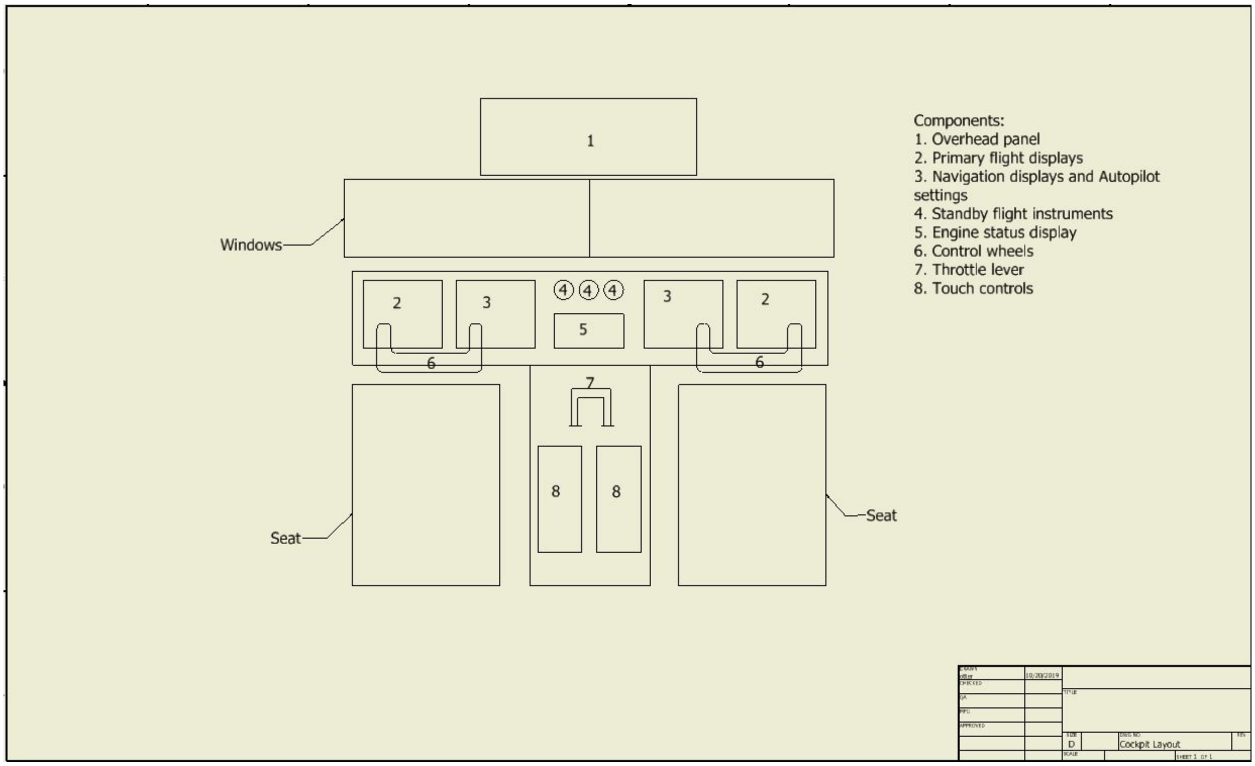


Figure 3.5.7: Cockpit layout

4 Performance Capabilities

4.1 Aerodynamic Performance

4.1.1 Mission Profile and Sizing

The mission profile of the designed aircraft is defined below:

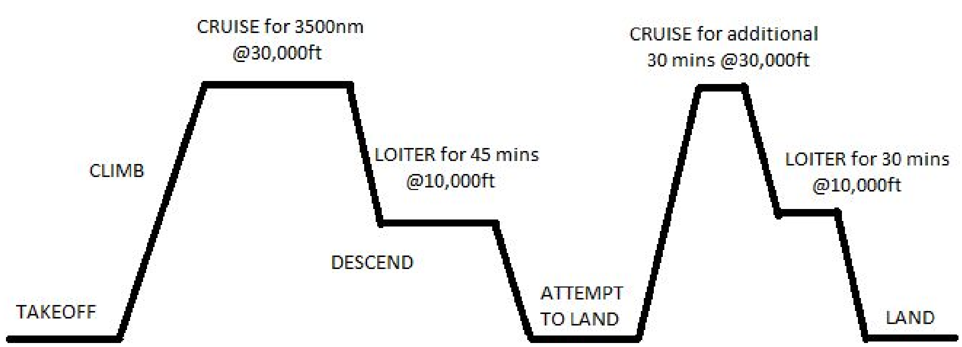


Figure 4.1: Mission Profile of the Aircraft

The initial sizing estimate of the aircraft began by picking a thrust to weight ratio of 0.3, a historical value. The maximum takeoff wing loading, 101 lb/sq.ft was determined from the smallest of the wing loadings at different flight conditions using equations (1)-(5).

FLIGHT CONDITIONS	Scaled takeoff Wing loading (lb/sq ft)
Stall	162
Takeoff	101
Cruise (ratioed to takeoff)	101
Landing	163

Table 4.1: Takeoff wing loading at different flight conditions

$$\frac{W}{S}_{stall} = \tfrac{1}{2}\rho V_{stall}{}^{2} C_{L_{max}}$$

$$C_{L_{max}} = 0.9 C_{l_{max}} cos(\Lambda_{0.25c})$$

$$\frac{W}{S}_{takeoff} = (TOP)\rho C_{L_{TO}} T/W$$

$$\frac{W}{S}_{landing} = \tfrac{1}{80}\left(S_{landing} - S_a\right)\rho C_{L_{max}}$$

$$\frac{W}{S}_{cruise} = q\sqrt{\pi A e C_{D_0}/3}$$

The cruise wing loading is ratioed up to takeoff using:

$$\frac{W}{S}_{takeoff} = \frac{W}{S}_{cruise}\frac{W_0}{W_{cruise}}$$

Given a wing aspect ratio of 9 and using weight fraction estimation methods provided by Raymer[4], our initial estimate was 524,000lbs. With a fixed engine given, the weight estimate had to be increased to a weight of 560000lbs which slightly reduced our flight performance. As a result, a thrust to weight and wing loading optimization technique was employed using this estimate as the baseline.

4.1.2 Thrust-to-weight and Wing Loading Optimization

The thrust to weight optimization technique was performed by forming a sizing matrix for wing loading values 70, 100 and 130 lb/sq.ft and thrust to weight ratios 0.2,0.3 and 0.4. The same procedure in computing the baseline values was also employed in determining the takeoff weight and performance parameters (listed under the Take-off, Cruise and Landing Parameters' section). Consequently, a sizing matrix table was achieved.

T/W	0.2	0.3	0.4
W/S			
70	W0 = 841068.1059lb Range = 4326.7386nm TOP = 176.4583 Landing distance = 3345.2315ft	W0 = 561832.4692lb Range = 3073.6524nm TOP= 117.6389 Landing distance = 3345.2315ft	W0 = 421375.3573lb Range = 1984.5416nm TOP = 88.2292 Landing distance = 3345.2315ft
100	W0 = 841067.6218lb Range = 5459.2401nm TOP = 252.0833 Landing distance = 4552.3045ft	**BASELINE** W0 = 560712.1173lb Range = 3884.5291nm TOP = 168.0556 Landing distance = 4552.3045ft	W0 = 421375.1183lb Range = 2532.1374nm TOP = 126.0417 Landing distance = 4552.3045ft
130	W0 = 841068.6357lb Range = 6013.3748nm TOP = 327.7083 Landing distance = 5713.322ft	W0 = 560712.6502lb Range = 4262.7656nm TOP = 218.4722 Landing distance = 5713.322ft	W0 = 421375.4345lb Range = 2759.3089nm TOP = 163.8542 Landing distance = 5713.322ft

**TOP = Take off parameter*

Table 4.1: Sizing matrix table

Upon interpolation and matching parameters to W/S and T/W locations, the graph below was obtained. It is worth noting that the lines for each takeoff weight are vertical because the thrust is based upon a fixed pre-selected value.

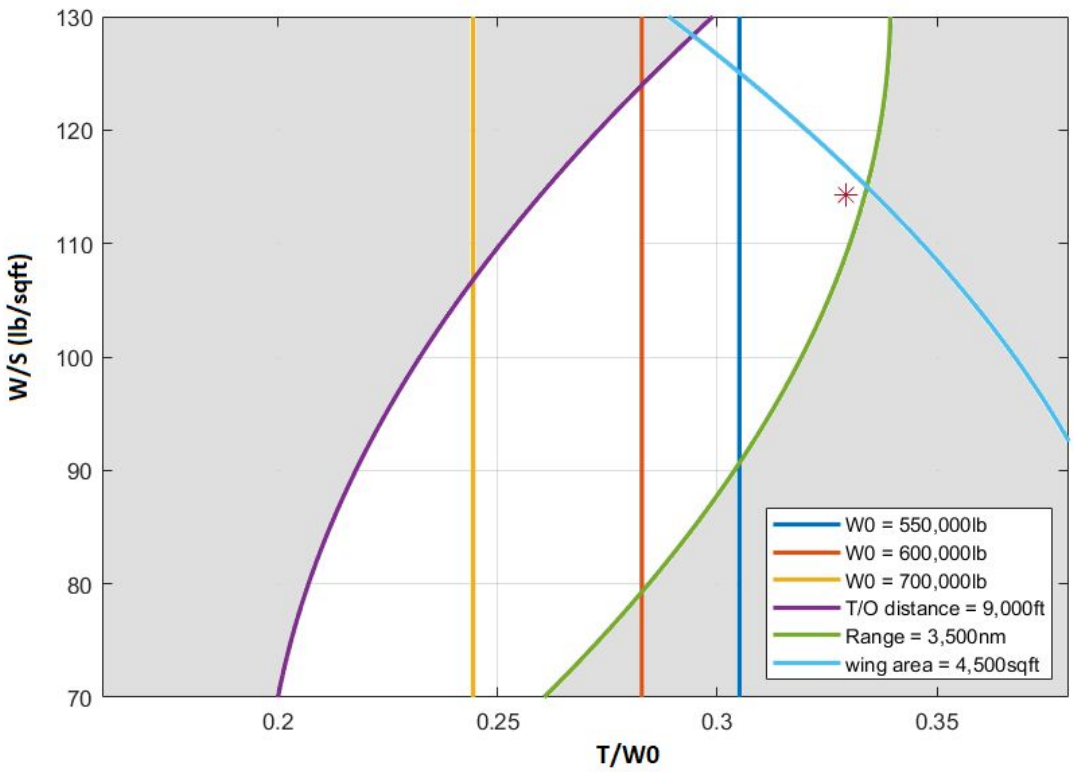


Figure 4.2: W/S-T/W optimization plot
(*red asterisk represents optimized choice and grey areas do not meet requirements*)

Apart from the required performance plot boundaries and weights, a line for a desired wing area (4500sq.ft) was added to help aid the selection process. This assisted in choosing an optimization point that also gave a reasonable wing area. Ultimately, the most optimal values that met the performance requirements with a low weight gave a wing loading of 114lb/sq.ft and thrust to weight ratio of 0.33 which brought our optimal weight to 509,000lb. These not only met the requirements, but were well within range of values for existing commercial aircrafts.

4.1.3 Take-off, Cruise and Landing Parameters

Given the optimal thrust-to-weight and wing loading above, the performance parameters of the aircraft are presented in the table below:

Range *(excluding extra range for missed landing)*	3652nmi
T/O distance *(from TOP)*	~6000ft
Landing distance	5080ft
Cruise velocity *(TAS)*	844ft/s
Cruise Mach number	0.849
Cruise altitude	30,000ft
Wing Area	4471sq.ft

Table 4.3: Performance Parameters

The takeoff distance is determined from the takeoff parameter introduced by Raymer[4].

Takeoff parameter:

$$TOP = \frac{W/S}{\sigma C_{L_{TO}} T/W}$$

The landing distance is given by:

$$S_{landing} = 80\left(\frac{W}{S}\right)\left(\frac{1}{\sigma C_{Lmax}}\right) + S_a$$

Where Sa=1000ft

The cruise velocity and Mach number are:

$$V_{cruise} = \frac{2W/S}{\rho\sqrt{\pi AeC_{D_0}/3}}$$

$$M_{cruise} = \frac{V_{cruise}}{a}$$

Finally, the wing area is given by:

$$S = \frac{W_0}{W_0/S}$$

4.2 Propulsion Performance

The table below is a summary of the propulsion performance metrics of the aircraft. It is useful to understand the velocity at different conditions due to their being constraints on the max approach speed given in the RFP. Approach speed was found to be 110 KCAS which is an ideal condition considering that it may be dangerous for an aircraft to have an excessive approach speed. Thrust and absolute ceiling of the aircraft were also calculated to have a deeper understanding of Brett the Jett's limitations propulsively.

Propulsion Performance	Velocity
V_{stall} (ft/s) TAS	231
$V_{takeoff}$ (ft/s) TAS	253
$V_{approach}$ (ft/s) TAS	300
$V_{approach}$ KCAS	110
V_{cruise} (ft/s) TAS	844
$T_{takeoff}$ (lbf)	168400
T_{cruise} (lbf)	33680
Absolute Ceiling (ft)	37,857

Table 4.2: Calculations of Velocity, Thrust, and Absolute Ceiling

Equations and theory behind the calculations can be attributed to Raymer and Dr. Smith reference material. Below are some of the equations used to calculate table 4.2:

$$V_{takeoff} = 1.1V_{stall}$$

$$V_{approach} = 1.3V_{stall}$$

$$V_{approach} = 1.3V_{stall}$$

$$T_{cruise} = 0.2T_{takeoff}$$

$$V_{mp} = \left(\frac{B_e}{3A_e}\right)^{1/4}$$

$$B_e = \frac{KW^2}{0.5\rho_{SL} S}$$

$$A_e = 0.5C_{D0}\,\rho_{SL}\,S$$

$$_{mp} = A_e V_{mp}^{\ 3} + B_e V_{mp}^{\ -1}$$

$$_{ceiling} = \left(\frac{_{mp}}{P_{SL}}\right)^{2/3}$$

$$\rho_{ceiling} = (_{ceiling})(\rho_{SL})$$

4.3 Flight Envelope

Design load factor max and min were taken from Raymer's book, the load factor maximum was n=3.5, and the load factor minimum was n=-1.5. Gust was also taken into consideration with speeds of ±25 (ft/s) and ±50 (ft/s) these speeds were taken from 14 Code of Federal Regulations (CFR) Part 25.

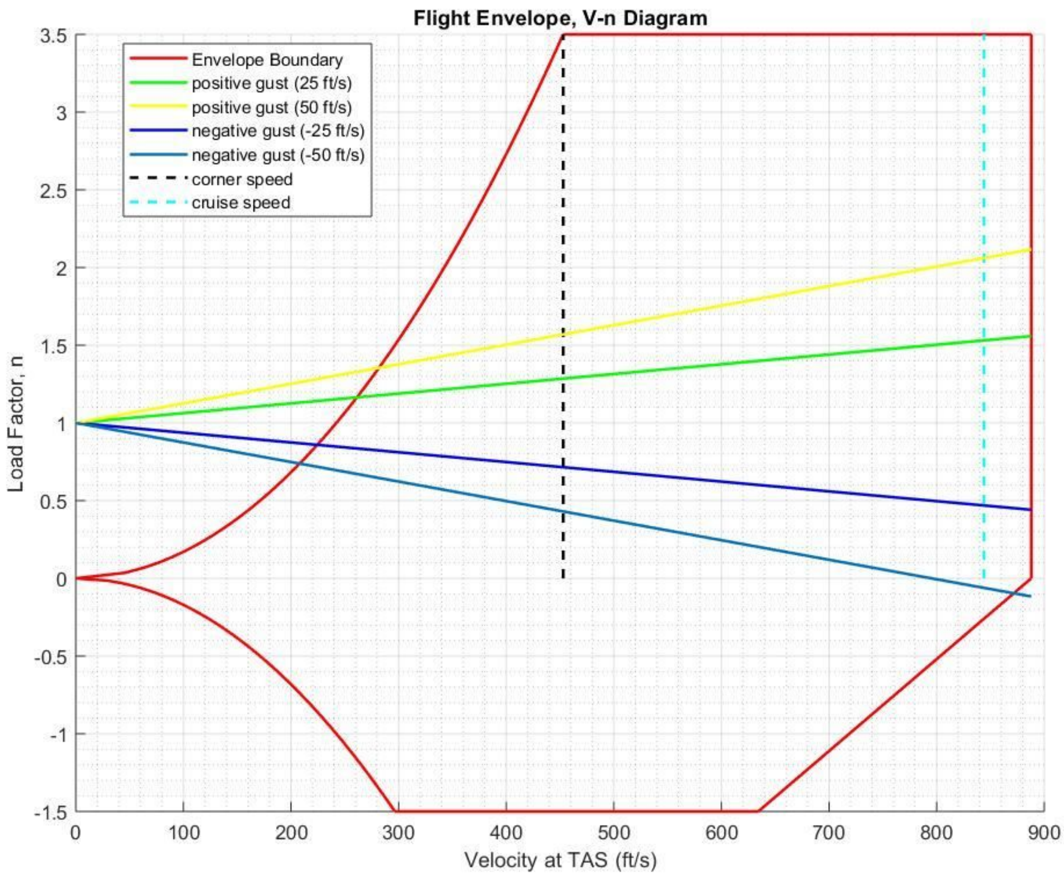


Figure 4.3: V-n Diagram which incorporates gust, corner speed, cruise speed, envelope boundary

5 Cost

5.1 Business Case Summary

With the rising popularity of commercial air travel major airports are becoming very congested. In order to encourage this growth in the industry we must turn to less congested satellite airports and increase the amount of short-range high capacity aircraft. This is a market that doesn't currently have many options available. Short range aircraft such as the A220, ATR 42/72, or Bombardier CRJ700/900/1000, all have passenger capacity under 160. High capacity aircraft such as the Boeing 777/747, A340/380 have ranges up to 8,208 nautical miles. There are little options between these two groups. This aircraft will supply an option to allow for high capacity short range travel.

5.2 Cost Groups

5.2.1 Assumptions

During the production planning it was assumed that this aircraft will take the format of OEM supplied final assembly line based in suburban United States. This means engineering and labor would only be applied to the final product assembly and quality. Being based in the United States also dictates average salaries, facility costs, and tariffs.

5.2.2 Design Choices

After initial preliminary design was finished, there seemed to be quite a lot of similarities between optimal short range high capacity design and the Boeing 777. For this reason, a decision was made to make the Boeing 777 our base design and apply for modifications to that design instead of starting a fresh application. This saved time and money not only in the developmental stages but also within production and

5.3 Cost Estimation

5.3.1 Development Costs

Developmental costs consist of one-time non-recurring fees at both the design and productions levels. Design costs include engineering labor spent on the design as well as the FAA/EASA design certification of airworthiness, this consists of testing and analysis as well as material and facilities needed to perform such tests in addition to the fees associated with the certification. Production costs encompasses fees per airplane produced such as production tooling, facilities and labor. According to the modified DAPCA IV Cost model developmental support, and flight test costs can be estimated using the following equations resulting in 62M in developmental support and 5M in flight testing with a total developmental cost of 67M.

$$C_D = 91.3 W_E^{0.630} V^{1.3} \{fps\} = 67.4 W_E^{0.630} V^{1.3} \{mks\}$$

$$C_F = 2498 W_E^{0.325} V^{0.822}\ FTA^{1.21}\ \{fps\} = 1947 W_E^{0.325} V^{0.822}\ FTA^{1.21}\ \{mks\}$$

where

W_E = empty weight in lb or kg
V = maximum velocity in kt or km/h
FTA = number of flight test aircraft

5.3.2 Engineering

Another large factor in developmental costs is the engineering labor costs associated with the design as well as quality control and manufacturing support. This can be estimated using the following equations and results in $350k in engineering costs, $110k in quality costs, and 815k in manufacturing costs. For a total of $1.3M in engineering associated costs.

$$H_E = 4.86 W_E^{0.777} V^{0.894} Q^{0.163}\ \{fps\} = 5.18 W_E^{0.777} V^{0.894} Q^{0.163}\ \{mks\}$$

$$H_Q = 0.133 H_M$$

$$H_M = 7.37 W_E^{0.82} V^{0.484} Q^{0.641}\ \{fps\} = 10.5 W_E^{0.82} V^{0.484} Q^{0.641}\ \{mks\}$$

where
Q = number of aircraft to be produced in 5 years

5.3.3 Certification

One area we for see major cost savings is in certification. Basing our design certification off of the Boeing 777 will not only greatly reduce the cost from a new design application estimated to cost $25-50M to a modification application estimated to cost $5-20M. This results in an averaged $47.5M savings. Not only will this result in monetary savings but this will also save time in the certification process. On average a new design application could take up to 12 years, however a modification application could be as short as 3 years. Savings of 9 years will also result in large savings. In addition each aircraft will have to be licensed and registered at an average cost of two to eight grand, however this is paid by the customer, paid regardless of which plane they choose, and frankly small enough to neglect. Therefore certification has been assumed to cost roughly 15M to apply for airworthiness of our design modifications.

5.3.4 Production Tooling

Another great cost savings opportunity that arises from the base design being a Boeing 777 is that production tooling is also similar. This means that the design and search of production tooling is null. Resulting in the only price accumulation is the purchasing of these already existing tools or if produced by Boeing the use of already owned tooling. With the elimination of custom tooling design comes the elimination of the majority of production tooling budgets. Due to the fact that buying existing tooling eliminates hiring designers and producers of the tooling and working with them through production iterations. This can eliminate up to 70% of the cost. Therefore production tooling can be estimated by the following equation resulting in a very modest 65k in production tooling costs.

$$(10/3)*H_T = 5.99W_E^{0.777}V^{0.696}Q^{0.263} \ \ \{fps\} \ = 7.22W_E^{0.777}V^{0.696}Q^{0.263}$$

5.3.5 Facilities and Labor

Facilities and labor are averaged on a 6,000 sqft rented facility at an average of $20/sqft/month. Tended to by a 3 shifts of 50 person labor force working 6 days a week at an average hourly wage of $39/hour. Producing 20 airplanes per month or 240 planes per year.

Cost group	Price (USD)
Facilities per year	1.4M
Labor per year	13.5M
Total per year	14.9M
Per Plane	62K

Table 5.3.1:Facilities and labor cost

5.4 Fly away cost

Fly away cost consists of all production costs including labor and materials that go into the final assembly of the product, including the airframe, engines, and avionics. The decision to make this plane in an OEM final assembly line would move the production costs for everything besides the final product out of house. While this may cause an increased cost for various products due to the profit mark ups and tariffs in most cases these profits would be neutralized against what would need to be done to produce these products in house. For example, when buying an engine you can either pay the profit mark up or you can pay for the engineering labor to design the engine, additional facilities to produce and test the engine, additional labor to make the engine, having to absorb the scrap costs of defective parts, etc. An OEM supplied final assembly line allows for a steady price on major components resulting in a flat flyaway cost per aircraft. Our airframe price is based on the cost of the Boeing 777, engine cost was from listed price, and avionics estimated by 25% total fly away cost.

Product	**Cost (USD)**
Airframe	38 M
Engine	44 M
Avionics	28 M
Facilities and Labor	62 K
Materials	46 M
Total Fly Away Cost	**156.1M**

Table:5.4.1:Fly away cost

5.5 Operating Cost

5.5.1 Consumables and Interior

Our consumable costs include mainly fuel but also interior design choices to keep our cabin modern and pleasant. As this aircraft is designed for short range flight, a 700nmi or roughly 2 hour flight will be the standard flight. Meaning this aircraft could feasibly perform up to 8 flights per day assuming a very narrow margin for airport traversing due to the design allowability to avoid congested airports. This means on average this plane will have approximately 5,800FH/years. With a TSFC of 0.56 at cruise for our engine and $6/gallon jet fuel including carbon tax, Brett the jet will incur roughy $90M in fuel costs per year.

Our interior aesthetic maintenance can be estimated per passenger. Specifically for the initial design this could include up to $4,200/passenger for a more modern design. In this case resulting in a $1.9M additional cost.

5.5.2 Operations and Maintenance

O&M costs groups are often considered in percentages of overall O&M cost. For commercial airfare generally acceptable percentages tend to be fuel 38%, crew 24%, maintenance 25%, depreciation 12%, insurance 1%, landing fees 2%.

Cost Group	Price per Year (USD)
Fuel	90M
Crew	56M
Maintenance	59M
Depreciation	28M
Insurance	2.3M
Landing fees	4.7M
Total O&M cost	**240M**

Table 5.5.2: O&M cost

5.6 Price

Taking all of these cost groups into consideration and selling our aircraft at a 15 % profit margin Brett the Jet would sell for 276.5M.

Cost group	Price (USD)
One time developmental costs	83.5M
One time developmental cost per plane with 1 year production payback	350K
Production cost per plane	240M
Total cost of plane	240.4M
Selling price on 15% profit	276.5M

Table:5.6.1:Selling Price Calculations

5.7 Profitability

From the airlines perspective, this is an incredibly profitable plane.This means that at an average first class ticket cost of $1,000 and economy ticket price of $400 per 2 hours of flight time and 16 hours of flight time per day you could make $650M/year. Bringing your capital expenditure pay back time to just 5 months and an average profit of $650M/year following, or a $16.3B profit over Brett the Jets 25 year lifetime. This plane offers a wide enough profit margin to offer incredibly competitive ticket prices and airline growth in the rising popularity of air travel.

6 Design Summary

The team's high capacity short range transport aircraft design proposal, "Brett the Jet" exceeded the design conditions outlined in the RFP. The specifications are in compliance with 14 CFR Part 25, listed below in Table 7. With this proposal, the team was able to offer a solution to not only the RFP, but to the aircraft market as a whole. "Brett the Jet" offers a competitive price point to high capacity short range aircraft missions.

Requirement	Request for Proposal	Brett the Jet
Passengers in Business	50	70
Passengers in Economy	350	380
Range (nmi)	3,500	3652
Takeoff Length (ft)	9,000	6000
Landing Distance (ft)	9,000	5080
Approach Speed (KCAS)	145	110

Table 6: RFP requirements in contrast to Brett the Jet

7 References

[1] Airworthiness Standards: Transport Category Airplanes, 14 Code of Federal Regulations (CFR) Part 25 (2019)

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[4] Raymer, Daniel. (2018). Aircraft Design: A Conceptual Approach Sixth Edition.Playa Del Rey, California: AIAA Education Series

[5] Rolls Royce Tent Series. (n.d.) Retreived from https://www.rolls-royce.com/products-and-services/civil-aerospace/airlines/power-of-trent.aspx#/

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