Conceptual Aircraft Design Project

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Abstract

In this report, we present the conceptual design of a business transport aircraft. Following design steps from (Raymer, 2012), we were able to successfully meet project mission requirements by creating a 13,730 pounds vehicle, which includes 13 passengers, one pilot and their luggage. Our design layout highly depends on our high lift devices mechanism and its weight impact on performance. Nevertheless, trailing edge devices on modern passenger aircraft have become lighter and more efficient with carbon fiber winding (Technology, 2017), which also increases durability. For future work, with more advanced graduate school knowledge, more details can be approached in structural and aircraft performance aspects to create a preliminary/detailed design layout. This is a process that could take one to two years to be accomplish.

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1. Statement of the problem

1.1. Introduction

"Aircraft conceptual design is a specialist skill and more than just drafting" (Raymer, 2012). When you are creating the geometric description of an aircraft to be built, having a diverse knowledge in aerodynamics, structures, controls, and propulsion, will help assure success. For this assignment, we were prompt to develop a credible aircraft conceptual design from a given set of requirements. As our designing strategy, concepts learned in class and taken from (Raymer, 2012) will be used to create a typical "real" conceptual design of a vehicle.

1.2. Mission Requirements

Our mission requirements, as seen in Table 1, were used to create a propeller business transport aircraft that will carry 13 passengers, one pilot and their luggage in relative comfort, and in a pressurized cabin. These parameters were used to help us draft our initial sketch, and initial sizing calculations for our "Dash One" layout.

Table 1. Design aircraft requirements

Performance requirements		
Cruise Altitud	20000	ft
Maximum Level speed Mid-cruise	250	mph
Range	1200	miles
Service ceiling	25000	ft
ROC at sea Level	1000	fpm
Stall speed	70	mph
Landing distance to clear 50ft	2000	ft
Takeoff distance to clear 50ft	2000	ft
Engine	Rubber	
Passengers	13	#
Pilot	1	#
Luggage	13	#

1.3. Primary Mission: Simple Cruise

Mission profile is a critical factor in gross weight estimates. For our business transport, we selected a Simple Cruise mission, as can be seen on Figure 1. It is the simplest route to meet our design requirements efficiently. As can be observed, the different mission segments are numbered, with zero denoting the start of the mission, these segments will be used later on for our takeoff weight approximation.

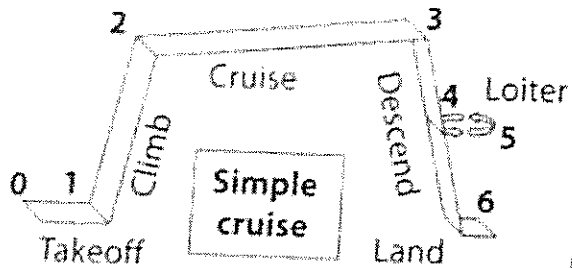


Figure 1. Mission profile.

2. Analysis

2.1. Initial Sketch Layout

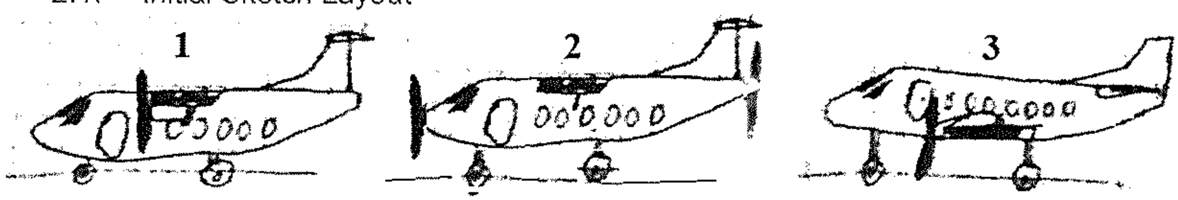


Figure 2. Trade Studies Sketches

As shown in Figure 2, three configurations were considered for the trade studies of our design layout. Each concept was sketched taking into account the practicality in terms of structures, aerodynamics and propulsion requirements. In the first concept, a high wing aircraft with propellers on the wings was proposed. This kind of configuration will result in a highly stable aircraft, but with some structural weight penalties due to landing gear and wing main supporting structures. On the other hand, the required clearance for the aircraft is shorter, which makes the aircraft lighter by means of landing gear length. Due to the location of the propellers, the tail structures will need to be a T-tail, given the constraints created by the propellers wake in stability and control. This also increases the weight of the vertical tail that will support forces from the horizontal tail forces.

This configuration also has structural rods as support in the wings. Some drag could be expected due to the rods, especially with flow separation or increases in boundary conditions. This could be a problem in the fuel consumption of the resulted vehicle at high velocities reducing the range, maximum velocity and overall performance. Continuing with our second concept, we explored the possibility of different locations for the engines where the propellers might operate in a more ideal location for better aerodynamics. Two engines are used for redundancies and to lower the propeller diameter requirement. This way, wing's top area is nearly unpolluted of propellers wake, and suction benefit in pusher propellers can be used to reduce drag.

Although further development of this configuration created much interest for our final decision, some constrains in being a tail heavy aircraft made us draft a simpler third concept. For our last concept we selected a low wing aircraft with some dihedral angle, featuring propellers on the wings and low horizontal tail. This concept lets us create a lighter configuration that can reflect the desired parameters for the mission, although more clearance might be needed for the propellers. Fuel tanks were placed on the wings reducing needed fueling systems for the engines and lowering the risk of fire in the fuselage. Also, much of the in-flight stresses on the wing root are reduced for this configuration by placing engines on the wing.

Since any aircraft design entails a series of tradeoffs, the best possible decision is not always clear. To arrive to an optimal decision several concepts might need to be developed. Or in case of high experienced designers, knowledgeable decisions can reduce cost and design iterations.

2.2. Initial Sizing

Sizing is the most important calculation in aircraft design, other than cost (Raymer, 2012). This is where we determined the size of the aircraft, specifically the weight that the aircraft must be designed so that it can perform the intended mission. The initial sizing analysis was completed for the third previously stated aircraft configuration. Using the initial sizing procedures outlined by Raymer, each individual segment of the mission was evaluated to determine the weight fraction of the mission. The takeoff and landing weight fraction were taken from empirical date, and other values were modified in accordance to requirements. Results can be seen on Table 2.

Table 2. Mission segment results.

#	**Mission segment**	**(Wi/Wi-1)**
1	Warmup and takeoff	0.970
2	Climb	0.985
3	Cruise	0.888
4	Loiter	0.996
5	Landing	0.995

A wing aspect ratio (AR) of 7.3 was selected. Commonly the wing aspect ratio is ultimately determined by a climb requirement. Since this required an initial airplane layout for the analysis, we made a best guess assumption. With the AR we can determine the maximum L/D by historical trends. Using our previously selected configuration layout, we looked at Figure 3 for a twin turbo propeller. Finding a wetted area ratio (S_{wet}/S_{ref}) of approximately 5. This resulted in a wetted aspect ratio of 1.46. And finally as shown in Figure 4, we were able to have our approximation of L/D max of 10.8 to continue with our sizing estimates.

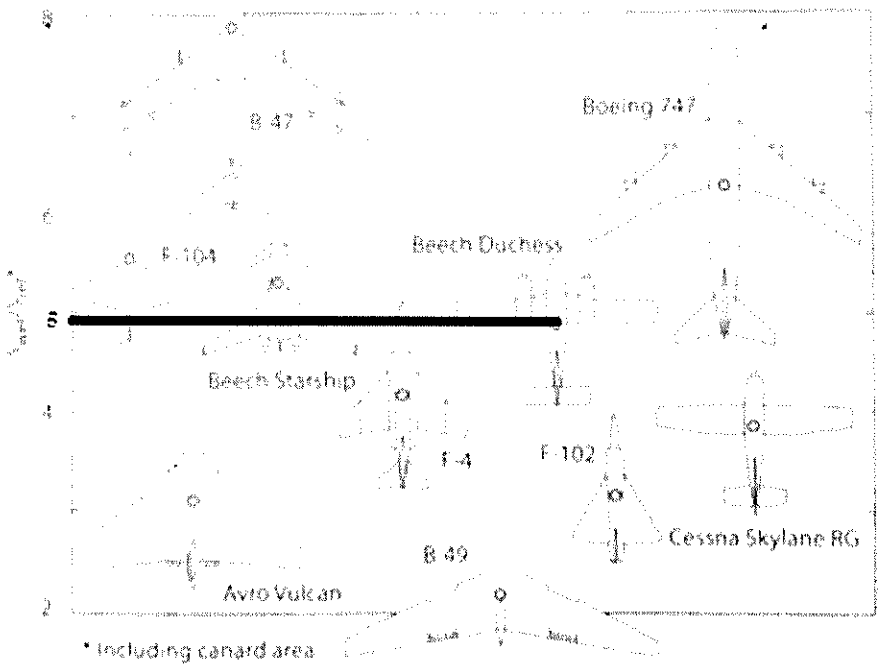


Figure 3. Wetted area ratios from Ref.1.

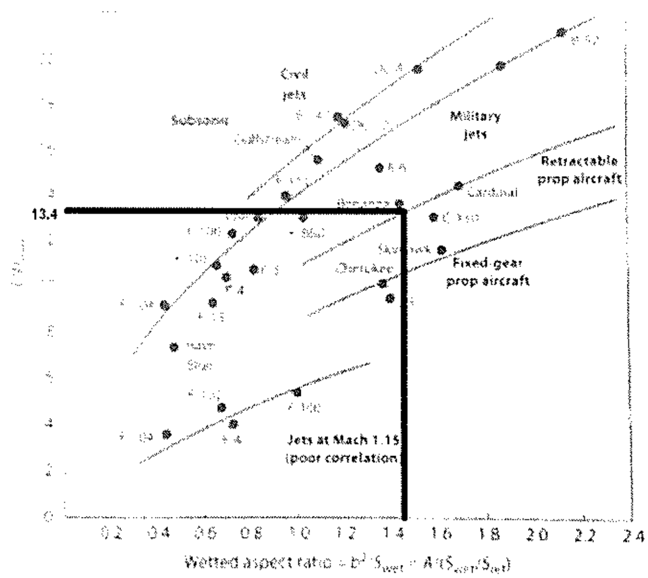


Figure 4. Maximum lift-to-drag ratio trends from Ref.1.

Using the mission requirements and the previously stated selected parameters, we were able to iterate into a takeoff weight of 12,020pounds, as seen in Table 3. We were able to calculate this with the use of Eq. 1-3. Where W_f/W_o is the fuel and gross weight ratio, W_e/W_o the empty and gross weight ratio that are approximated using the mission segment method. Calculations can be seen in Appendix A, with MATLAB and Excel results.

$$\frac{W_f}{W_o} = 1.06\left(1 - \frac{W_6}{W_0}\right) \qquad \textbf{Eq. 1}$$

$$\frac{W_e}{W_o} = 0.916 W_o^{-0.0795} \qquad \textbf{Eq. 2}$$

$$W_o = W_{payload} + W_{equip} + \frac{W_e}{W_o} W_o + \frac{W_f}{W_o} W_o \qquad \textbf{Eq. 3}$$

Table 3. Takeoff- Weight Sizing

W0,guess	We/W0	We	W0,calculated
11990	0.600247	12025.3	11990
12000	0.600222	12023.99	12000
12010	0.600197	12022.68	12010
12020	0.600172	12021.38	12020
12030	0.600147	12020.08	12030

2.2.1. W/S & T/W Estimate

From the airfoil data shown in Figure 5, the maximum lift coefficient of the wing, C_{Lmax} was calculated. High lift devices (HLD) were considered. HLD allowed us to have a larger W/S, while avoiding penalties of a huge wing (wetted area and weight) that would hurt our design in other areas by not meeting a requirement like the max speed. Consequently because of this, Equations 4-5 and Table 4 were used to approximate a C_{Lmax} for our wing and HLD configuration, shown in Figure 6.

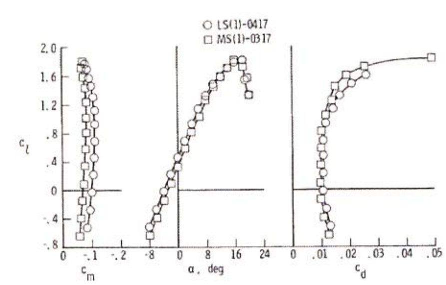


Figure 5. NASA/LANGLEY LS (1)-0417 section data (Robert J. McGhee, 1979).

$$\Delta C_{L_{max}} = 0.9 \Delta C_{l_{max}} \left(\frac{S_{\text{flapped}}}{S_{\text{ref}}} \right) \cos \Lambda_{\text{H.L.}} \qquad \textbf{Eq. 4}$$

$$C_{L_{\max}} = C_{\ell_{\max}} \left(\frac{C_{L_{\max}}}{C_{\ell_{\max}}} \right) + \Delta C_{L_{\max}} \qquad \textbf{Eq. 5}$$

Where C_{lmax} is the maximum lift coefficient of the airfoil, $\Lambda_{0.25c}$ is the sweep angle at the quarter chord location, ΔC_{lmax} representing the lift contributions from the HLD, and ΔC_{Lmax} the change in lift coefficient from lift devices. Which resulted in a C_{Lmax} for our wing with double slotted flaps, and slat of 3.0. Not much focus was given to the airfoil selection. Normally, airfoils are redesign for specific aircraft in a much later step than conceptual design.

Table 4. Approximate lift Contributions of High-Lift Devices.

High-lift device	$\Delta C_{l_{max}}$
Flaps	
Plain and split	0.9
Slotted	1.3
Fowler	1.3 c'/c
Double slotted	1.6 c'/c
Triple slotted	1.9 c'/c
Leading-edge devices	
Fixed slot	0.2
Leading-edge flap	0.3
Kruger flap	0.3
Slat	0.4 c'/c

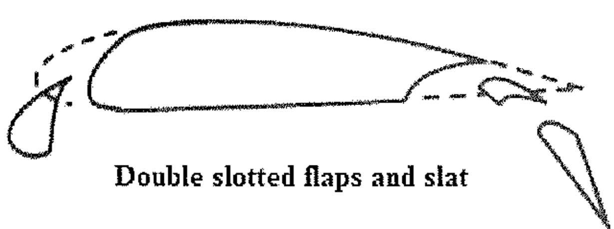


Figure 6. HLD configuration selection.

After knowing our C_{Lmax}, we used the takeoff distance requirements of 2000ft, and over height of 50ft, with Figure 7, and calculated a T/W ratio of 0.0727, which leaded to a 998 BHP requirement. With a known wing loading (W/S) of 37.8741psf, we finalized our main geometrical parameters to draft our Dash One layout.

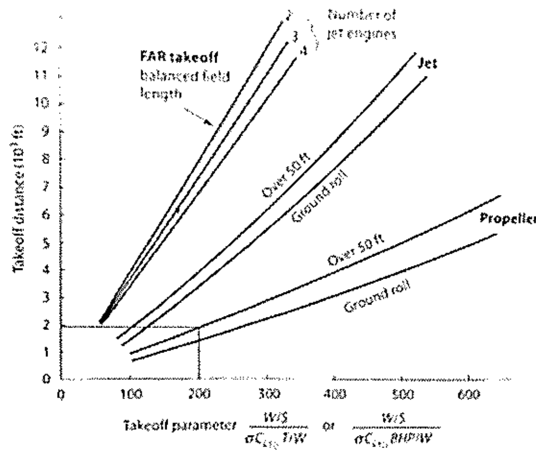
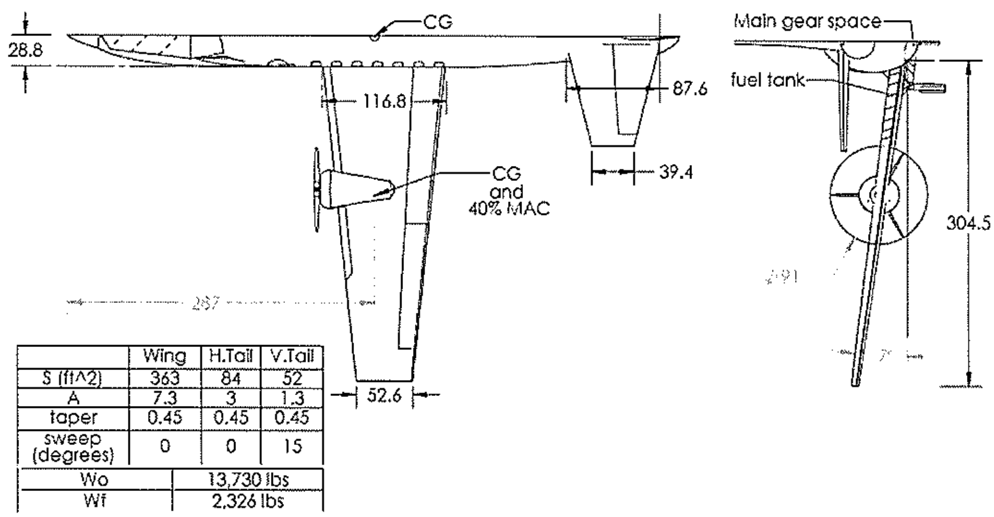


Figure 7. Take-off Parameters.

2.3. Initial Layout

An initial layout was drafted and analyzed with the sizing take-off weight of 12,020lbs. But after using Raymer's Statistical Empty Weight Buildup method a weight of 13730lbs. resulted. Since the difference in weight was of 14.2%/ 1710lbs, a resizing of the aircraft was done using 13730lbs. This gave us dimensions that resulted in a reference wing area and span of 363 ft^2 and 51.4 ft., respectively. A taper ratios of 0.45 was selected to almost completely eliminate those effects for an un-swept wing and produce a lift distribution very close to the elliptical ideal. Concluding in a dash two layout as shown in Fig. 7, that we will analyze further in our following section. From the layout, fuselage length was approximated with statistical equations developed from data provided in (Taylor, 1976). Tail sizing were repeated using tail volume coefficient for twin turboprop. And a 7 degree dihedral angle on the wing was selected from guidelines developed by Raymer for low wing civil aircraft, to help with clearance and stability.



	Wing	H.Tail	V.Tail
S (ft^2)	363	84	52
A	7.3	3	1.3
taper	0.45	0.45	0.45
sweep (degrees)	0	0	15

Wo	13,730 lbs
Wf	2,326 lbs

Figure 8. Dash Two Layout.

2.4. Aerodynamic

The initial sizing was based upon rough estimates of the aircraft's aerodynamics, weights, and propulsion characteristics. Now that we have an initial layout, we can evaluate our vehicle's aerodynamic behavior starting with the lift curve slope, $C_{L\alpha}$, that was found using Equations 6-9.

$$C_{L\alpha} = \frac{2\pi AR}{2 + \sqrt{4 + \frac{AR^2 \beta^2}{\eta^2}\left(1 + \frac{\tan^2 \Lambda_{max\,t}}{\beta^2}\right)}}\left(\frac{S_{ref}}{S_{exp}}\right)(F) \qquad \textbf{Eq. 6}$$

$$\beta = 1 - M^2 \qquad \textbf{Eq. 7}$$

$$\eta = \frac{C_{l\alpha}}{2\pi / \beta} \qquad \textbf{Eq. 8}$$

$$F = 1.07(1 + d/b)^2 \qquad \textbf{Eq. 9}$$

In these equations, AR is the aspect ratio, Λ_{max} t is the sweep at the maximum airfoil thickness, Sref is the wing reference area, S_{exp} is the exposed area of the wing, d is the diameter of the fuselage, and b is the wing span. Giving a 0.083 per degree $C_{L\alpha}$ that resulted in Fig. 9 slope.

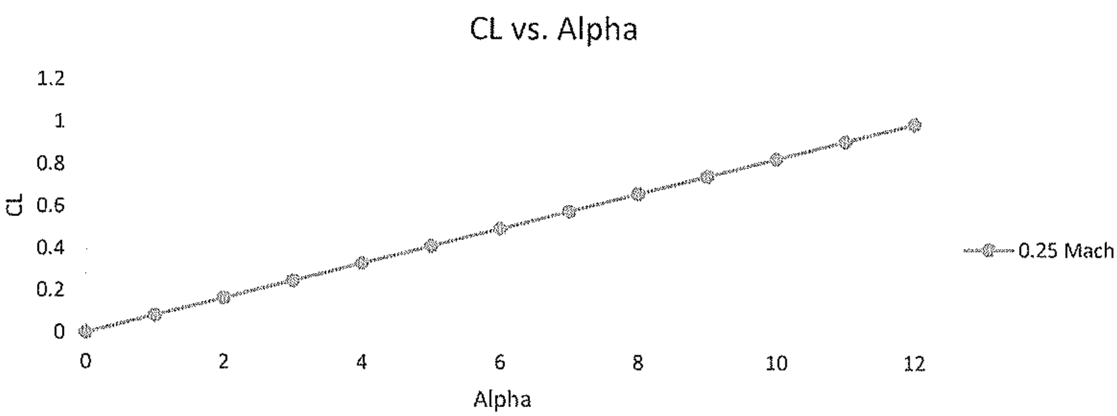


Figure 9. CL alpha slope.

2.4.1. Wetted-Area Determination

The wetted area S_{wet}, which is the total exposed surface area of the aircraft, had to be calculated for each external component. The wetted area was calculated mainly for the drag estimate, as it is a major contributor to friction drag. The exposed component included the wing, tail, and fuselage. The wing and tail wetted areas were approximated by multiplying the true-view exposed planform area $S_{exposed}$, as shown on Figure 9, by a factor based upon the thickness ratio. Given our thickness ration for the wing and tail of 0.08 and 0.04 respectively, Eq. 10 and Eq. 11 were used to make these estimates.

Consequently, the wetter area for the wing, horizontal tail, and vertical tail were found, as shown in Table 5. It was also noted that the true exposed planform area was the projected (top-view) area divided by the cosine of the dihedral angle of 7 degrees for our wing.

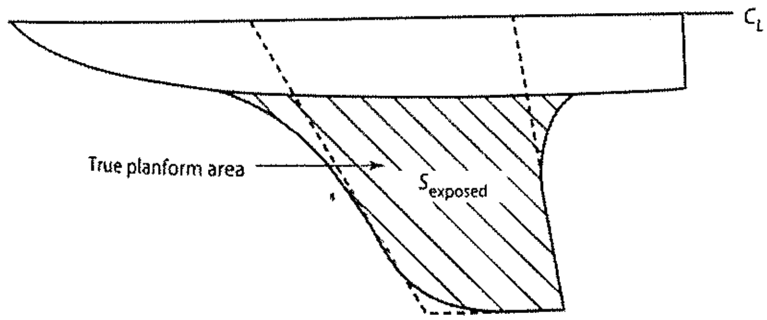


Figure 10. Exposed Area example

If $t/c < 0.05$,

$$S_{wet} = 2.003\, S_{exposed} \quad \textbf{Eq. 10}$$

If $t/c > 0.05$,

$$S_{wet} = S_{exposed}[1.977 + 0.52\,(t/c)] \quad \textbf{Eq. 11}$$

Table 5. Wing and tail wetted area results.

Wetted Area	
Wing	667.0
Horizontal Tail	155.5
Vertical Tail	106.4

In terms of the fuselage, the wetted area was estimated using just the side and top views of the aircraft by the method shown in Figure 10. The side and top-view projected areas of the fuselage were measured from the drawing and the values were averaged. Eq. 12 was used given that our cross-section represented a typical aircraft with a mixture of a circular and rectangular shapes, which yielded a resulting 752.5 ft^2.

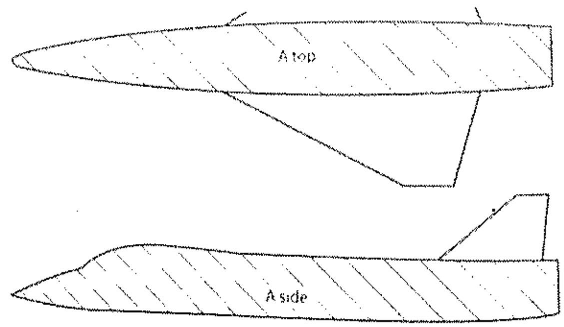


Figure 11. Fuselage top and side areas for wetted area calculations.

$$S_{wet} \cong 3.4\left(\frac{A_{top} + A_{side}}{2}\right) \qquad \textbf{Eq. 12}$$

2.4.2. Friction Drag Determination

After each exposed component wetted area was acquired, parasite drag calculations were done using Raymer component buildup method. As shown in Table 6, calculated parasite drag resulted in a 0.032 taking into account the wing, cockpit, fuselage, and aircraft tail components. Oswald span efficiency factor resulted in 0.83.

Table 6. Parasite drag estimates.

	Reynold	Cf	FF	S-wet (ft^2)	Cdo
Fuselage	4.69E+07	0.0023	1.8745	753	0.009
Wing	7.27E+06	0.0032	1.4272	655	0.008
Vertical Tail	6.53E+06	0.0032	1.2576	93	0.001
Horizontal Tail	5.46E+06	0.0033	1.2576	150	0.002
Cockpit	-	-	-	-	0.003
Misc..Engine	-	-	-	-	0.002
3xCooling	-	-	-	-	0.004
				Total + 10%=	0.032

2.5. Propulsion

Our propulsion system design started with our engine selection shown in Table 7. Where the takeoff parameter helped us determine our power requirement of 1000HP, with the use of two PT6A-6, illustrated in Figure 11, of 500bHP with CbHP of 0.4(0.5) #/NHPxHR in cruise (loiter), and 2700RPM.

Table 7. Selected rubber engine.

Engine	bHP	Weight (lbs.)	Length (in.)	Width (in.)	Height (in.)	Cost
G	500	700	50	38	23	110,000

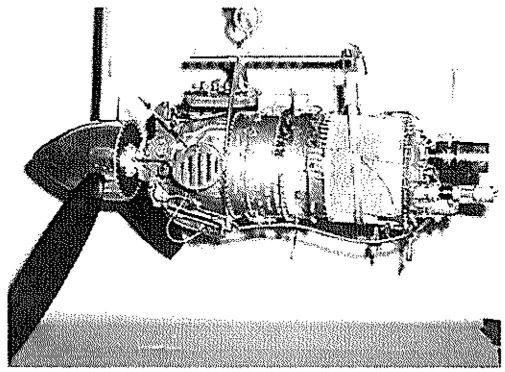


Figure 12. PT6A-6 engine.

The diameter for the propeller was calculated for a 3 blade using Equation 13, where Kp is a number of blades dependent constant (kp= 1.6 for No.3). This resulted in a 7.6 ft propeller diameter with our 500HP engines. As a mesurement of efficiency, the advance ratio was calculated resulting in 0.7544, where Figure 12-13 were used to find specific efficiency for our propeller with changes in speed.

$$D = K_p \sqrt[4]{(hp)} \qquad \textbf{Eq. 13}$$

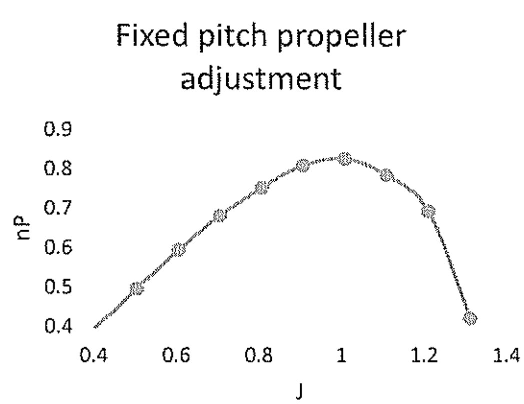


Figure 13. Fixed-pitch propeller adjustment.

Finally as seen in Fig. 14, with the use of Eq. 14-15 we were able to find our Thrust-Velocity curve. This with an effective propeller efficiency of 0.984, helped us find our thrust estimates for cruise, stall, and static as shown in Table 8. Further details of the calculations can be seen in Appendix A.

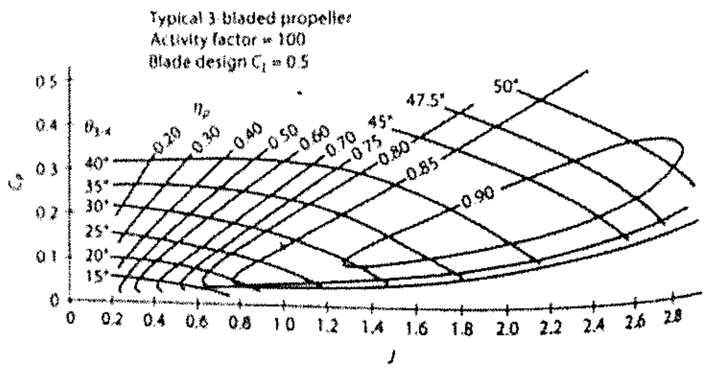


Figure 14. Forward-flight thrust and efficiency.

$$T_1 = \frac{550(hp)\eta_p}{V} \qquad \textbf{Forward Flight} \qquad \textbf{Eq. 14}$$

$$T_2 = \frac{c_T}{c_p}\frac{550(hp)}{nD} \qquad \textbf{Static} \qquad \textbf{Eq. 15}$$

Table 8. Thrust results.

Thrust cruise	Thrust stall	Thrust static
1688 lbs	2022 lbs	2744 lbs

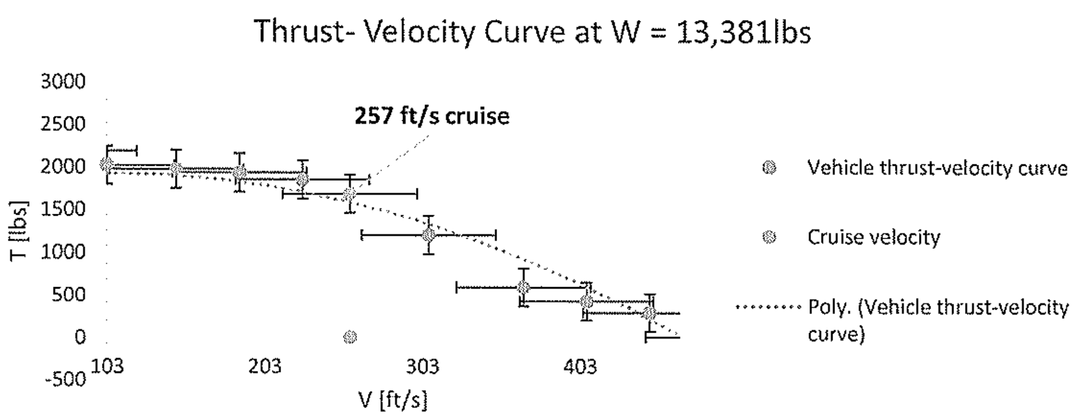


Figure 15. Thrust-velocity curve for the candidate motor at 20,000ft.

2.6. Weights

Weights analysis was done using Raymer's Statistical Empty Weight Buildup Method (Raymer, 2012) for a general aviation aircraft, as seen in Table 9. The takeoff weight of 13,730 lbs. was used as the structural design weight, which resulted in a 1710lbs of difference from our initial sizing estimates. For this reason, all geometrical parameters were changed using an initial 13,730 take-off weight for our final layout analysis. In the center of gravity analysis, all components were placed in their respective locations, as seen in Figure 15. The wing was placed so the CG would end up in our 40%MAC longitudinal location. Finally, all used inputs can be seen in Appendix A.

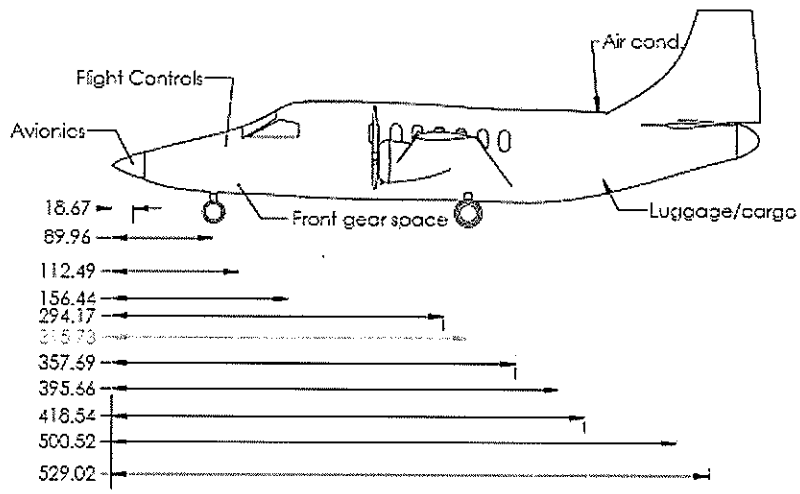


Figure 16. Center of gravity moment arms.

Table 9. Weight estimates

General Aviation Weights Statement			
Components	**Weight lbs**	**Loc ft**	**Moment ft-lbs**
STRUCTURES			
Wing+Fuel+System	1252	294	1873556
Horizontal tail	141	501	70778
Vertical tail	83	529	43796
Fuselage	1073	361	387372
Main Landing gear	69	316	21810
Nose Landing gear	24	112	2693
PROPULSION			
Engine-installed	2163	n/a	Included
Fuel system/tanks	183	n/a	In
			Wing Moment

EQUIPMENT			
Flight controls	556	90	50049
Hydraulics	114	290	33060
Electrical	480	500	240117
Avionics	1082	19	20205
Furnishings	280	296	82880
Air conditioning	619	397	245822
USEFUL LOAD			
Crew	170	156	26595
Fuel-usable	2775	n/a	0
Passengers	2210	296	654160
Cargo/payload	455	419	190436
Empty	8120	Total.M	3943328
TAKEOFF GROSS	13730	CG Loc.	287

2.7. Performance

All performance items were calculated and compared on Table 10. Our resulted performance met the mission requirements with acceptable differences.

Table 10. Performance comparison.

Performance items				% Diff
A	Maximum speed	250 mph	262 mph	4.80%
B	Range	1200 miles	1303 miles	8.58%
C	Service ceiling	25000 ft	26,000 ft	4.00%
D	ROC at sea level	1000 fpm	1,044 fpm	4.40%
E	Stall speed	70 mph	70 mph	0.00%
F	Landing distance to clear 50ft	2000 ft	1729 ft	15.55%
G	Takeoff distance to clear 50ft	2000 ft	1817 ft	9.15%

Figure 17. Vehicle turn rates at 20,000ft.

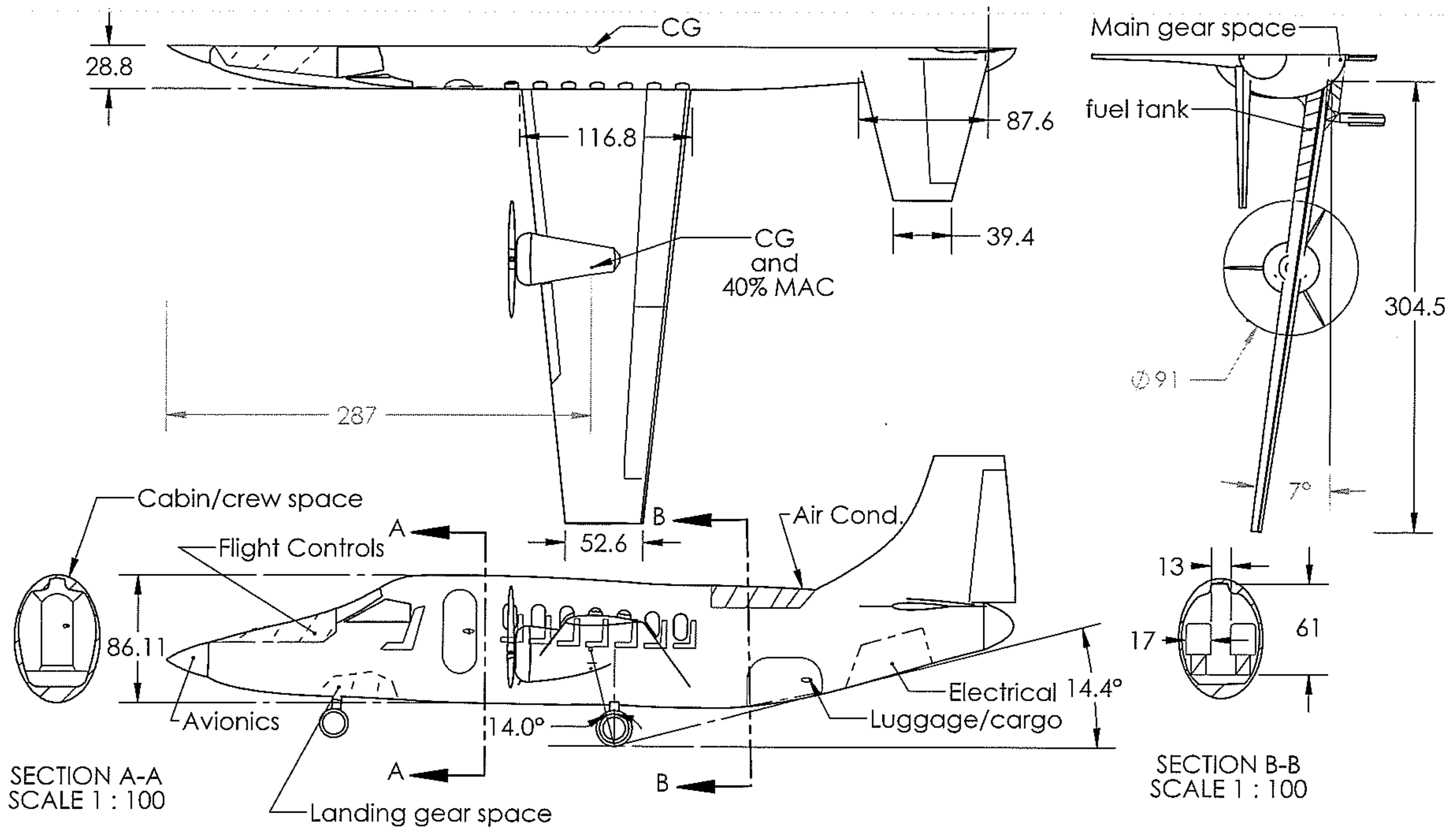
As illustrated in Figure 16, our aircraft is almost in its roof with little maneuverability at a 20,000ft altitude.

3. Final Configuration

For our final configuration, the aircraft internal volume was used as a measurement of the reasonableness of the design. Although a conceptual design layout cannot show all of the internal components that will be packed inside, main component volumes were accounted for in our layout. A statistical approach was used to determine if there was enough room in the design after the design layout was completed. As done with the wetted-area on the fuselage, the internal volume was quickly estimated with Eq.4. During this analysis, our main interest was that our layout could meet fuel tank, cockpit, passengers, landing gear, wing beam, structural supports and luggage volume requirements.

$$\mathrm{Vol} \cong 3.4 \frac{(A_{\mathrm{top}})(A_{\mathrm{side}})}{4L} \qquad \textbf{Eq. 16}$$

A full size drawing of the aircraft is shown on the next page, and Figure 10 shows the aircraft as a render image.



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S (ft^2)	363	84	52
A	7.3	3	1.3
taper	0.45	0.45	0.45
sweep (degrees)	0	0	15

Wo	13,730 lbs
Wf	2,326 lbs

NAME
DRAWN Nelson Ojeda
COMMENTS:
Span = 617 in
Fusegalege L = 572in

Polytechnic University of Puerto Rico

TITLE:
Conceptual Design
13 Passengers Aircraft

SIZE A DWG. NO. Performance_FP REV 2

SCALE: 1:100 SHEET 1 OF 1

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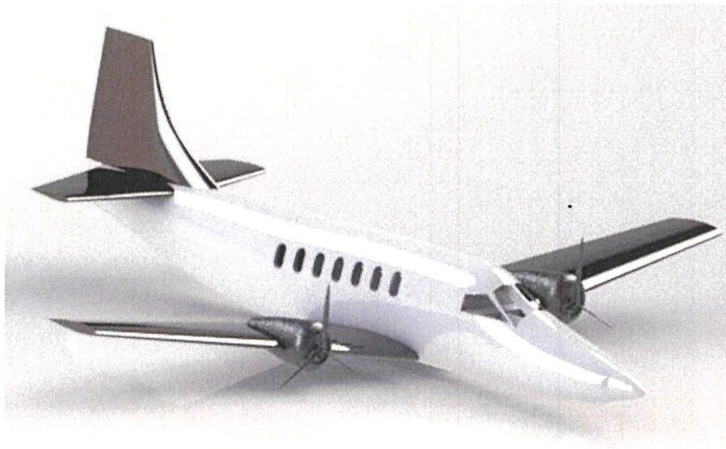

Figure 18. Rendered view

4. Recommendations

During our conceptual design, the requirements for good stability, control, and handling qualities were addressed through the use of tail volume coefficients and the proper location of the wing with respect to the aircraft center of gravity. Although this rule of thumb methods resulted in a design that will probably be as stable as desired and controllable as required, it is critical that we confirm with analysis. Because of this, we recommend that a full flight dynamics analysis of the final layout is done. A stable and controllable aircraft is of importance for a successful flight certifications of the vehicle. In large aircraft companies, controls experts analyze the aircraft by using a six-degree-of-freedom (DOF) aircraft dynamics computer program with inputs from computational fluid dynamics and wind tunnel test. After learning how our design might behave in flight, we can continue with a more detailed design, to be able to do a more accurate stress analysis.

5. References

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