USA 5000 Design Competition

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A course at Purdue University, within the school of Aeronautical and Astronautical Engineering, called Intro to Aerospace Design or AAE251, creates a design project for students to complete throughout each semester. Our team was tasked with one of the many design projects they have assigned. The problem presented to our team was to design an aircraft for a new air race organization and race called the USA 5000. Similar to the Red Bull Air Race Series this competition will display many impressive aircraft designed by our peers within the course. Each aircraft is limited to all assumptions and constraints presented by the organization. This report is a guide to how our team approached each part of the design process. We started by introducing the scope of the problem, the stakeholders, what their needs and requirements were, and possible risks involved with not only our design but also the race. From there our team provides a thorough description of the design process starting from researching reference aircraft to calculating expected weights, takeoff and landing distances, wing and tail parameters, and total drag. Next, our team reviews the concept selection process. This includes our initial concepts for the layout of the aircraft we chose based on certain performance calculations and research. Finally, a conclusion is given with what we learned throughout the process and how we expect future work on the project to continue. Overall we found that our design choices led to a sufficient aircraft with very similar aerodynamic characteristics to reference aircraft.

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1 INTRODUCTION

A new air race organization created a competition called the USA 5000 and like the Red Bull Air Race Series, each pilot must fly against the clock to complete the track. Not only will the pilots have to maintain a strict pace but also follow all constraints set by the new organization. Within this race, each contender must fly a single-pilot airplane and complete 25 laps consisting of two parallel straight lines of 100 nautical miles each. Tight turns will have to be completed after each straight away. There are also two designated waypoints represented with air gates along the track that all aircraft and their pilots must pass through. These waypoints are placed at different altitudes and distances from each other. Waypoint One is placed at an altitude of 50 meters above sea level and Waypoint Two is at an altitude of 790 meters. The distance between waypoint one and waypoint two is one nautical mile which will cause the pilot to make a quick climb and descent between these points. Overall the race is meant to test each team that enters the competition and their ability to design an aircraft to maximize aerodynamic efficiencies while also being limited by certain constraints. Our team has been tasked with entering the race, designing an aircraft, and competing against our fellow peers to reduce race time.

To understand the scope of the problem and the governing assumptions/constraints, as a team we looked into the Red Bull Air Race as mentioned before, and the guidelines posted by the FAI or the Fédération Aéronautique Internationale. From this research we found that each team who enters such a race must comply with the FAI's regulations and must nominate their pilots and officials according to the specified limits found in section 1 of the "General Rules for International Aerobatic Events" (FAI Sporting Code Section 6 Regulations for the Conduct of International Aerobatic Events Part 1 Powered Aircraft Fédération Aéronautique Internationale, 2021). Safety regulations and other constraints for the problem/task at hand can also be found within this section. A full list of the most important assumptions for this project and problem can be found below:

Assumptions/Constraints:

- The competition will take place within ideal weather conditions. Ideal weather conditions are determined by the FAI Sporting Code of Conduct section 3.6. In this section, it dictates both the weather conditions, time, and wind speeds that the aircraft can compete in.
- The aircraft will pull over 10 Gs of force (The pilot must have adequate training)
- There will be well-trained ground crews to service the plane in case of emergencies and mid-race refueling.
- Aircraft will be assumed to be flying in the standard atmosphere to help calculate different aerodynamic efficiencies.
- The sum of linear dimensions (wingspan + fuselage length + airplane's height with extended landing gear) < 20 m
- Fuel can only be stored in a 34 US gal tank in the fuselage.
- The refueling turnaround time, in minutes, is obtained with the following model: [10+ 5 ($V/34$)] where V is the fuel volume added during the refueling. The turnaround time
- encompasses the times of descent, landing, refueling/pilot change, takeoff, and climb back to cruising altitude.
- maximum power plant output is 350 hp.
- The maximum allowed Mach number in the race is 0.4.
- The "pit boxes" are located in a paved 1500 m long airstrip between waypoint #1 and the left corner (where the flight altitude is lower).
- Refueling can only happen when the aircraft is flying from waypoint #1 to the left corner, and not the other way around.
- The team does not need to fully refuel the airplane. Fuel amount can be strategically decided.
- Making the landing gear retractable adds 100 kg to the airplane.

Overall these assumptions and constraints will be the determining factors when designing our aircraft and making sure we meet all regulations set by the FAI and the new organization putting together the USA 5000.

Team Viper Mission Statement:

Our goal is to develop a high-performance, reliable single-pilot aircraft capable of agile flight at varying altitudes and covering long distances without mechanical issues. With a fuel tank capacity of 34 gallons and a maximum horsepower of 350, our aircraft is designed to excel in the USA 5000 air race, demonstrating exceptional speed, endurance, and maneuverability.

2 NEEDS, REQUIREMENTS, AND RISK ANALYSIS

The following section will describe the basic aspects of our project, detailing how we have identified stakeholders and their needs, established requirements based on these needs, and conducted a preliminary risk analysis.

2.1 STAKEHOLDERS AND THEIR NEEDS

As a team, we referred to the scope of the problem and project objectives as a way to build a sufficient foundation for identifying everyone who would potentially be involved. Also, to make sure we did not miss an important factor, we explored all of the entities that could be impacted by our design process and participation within the USA 5000. As a result, a list of potential stakeholders was developed and can be seen below:

Pilots: The pilots will be the people flying the aircraft; they are stakeholders because they are the ones in the aircraft operating it during the race

Race Organization: Organizes the race and gets money from people watching and each racing team.

Public / Spectators: Pays to watch the race, and participate as a form of entertainment

Design Team: Creates the aircraft, and loses money if they create a poor design.

Funding Organization/Sponsors: Funds the racing team through advertisements that help their business.

Race Crew/Race Team: Gets more money if they win the race and help the airplane and pilot operate.

Part Suppliers: Supplies parts for the aircraft/engineers, teams pay these companies to get parts for their airplanes.

Environmental Organizations: Their values might not align with the racing teams and could create barriers to the performance of the airplanes.

2.2 REQUIREMENTS

The list of stakeholders above incorporates all entities involved in our project and each has an important set of needs. To identify these needs our team asked what each stakeholder must have in order to perform or participate in this race. From this process, we developed a list of the most crucial needs and design requirements to fulfill the needs of these stakeholders. This can be seen below, where stakeholders are listed, followed by a bulleted list describing their needs, followed further by the sub-bulleted list describing the design requirements associated with each need.

Stakeholder 1: Pilot

- The aircraft is able to support a pilot of a certain height and weight
    - The pilot must not exceed 180 lbs and must be below 6ft tall. The pilot should be light and not too tall so the plane can operate as fast as possible. A heavier pilot would cause more inertial forces than all the performance of the plane. The pilot should not be too tall because the cockpit would affect the drag on the aircraft. The weight of the pilot is verifiable through a scale system and the height is verifiable through a height measurement. Weight is generally dependent on genetics and lifestyle and height is only dependent on genetics.

Stakeholder 2: Race Organization

- The speed of the aircraft is within the limits specified in the constraints
    - Aircraft are required to remain below the Mach 0.4 limitation created as a maximum speed constraint for the aircraft. This is a requirement made by the organization and will be enforced at the USA 5000. The speed is verifiable through measurements made by the pilot during test runs. The speed of the

aircraft is dependent on the pilot's ability to maintain a constant speed and not go over the limit set by the race organization.

Stakeholder 3: Spectators

- The race is fun and exciting to watch
  - Each plane is capable of flying consistent lap times within 5-6 seconds of each other. Close racing will keep fans entertained as there is intrigue and suspense on who will win as multiple competitors will have the opportunity to win. Lap times will be measured with stopwatches and timers and will be measured in the format minutes: seconds.
- To lower safety risks to spectators, planes are required to land if they exhibit technical issues in the air
  - If sensor readings from any other part of the engine as specified by the manufacturer (Lycoming, n.d.) are outside the safe operational limits of the aircraft. Will be measured through various engine sensors and the ECU and relayed to ground crews. This metric is verifiable as there are a series of numeric values provided by the ECU that are defined by the operations manual. These measurements are dependent on the performance of the engine and the inputs by the pilot.
- The spectators are safe from the planes and the stands are structurally sound
  - Pilots cannot exceed a maximum G force of 12 Gs similar to the rules of the Red Bull Air Race (*Red Bull Air Race*, 2017). This is verifiable through acceleration sensors that measure changes in the speed/direction of the aircraft. This metric is dependent on both the health limits of the pilot and the stresses that the aircraft is designed to undergo

Stakeholder 4: Design/Engineering Team

- The plane is affordable to race teams relative to competing aircraft
  - The cost of purchasing the airplane is below $400,000. Similar existing aircraft, such as the Zivko Edge 540, sell for an average of $340,000 before any further modification (Aero Corner). If this is a viable aircraft design to compete with our aircraft, we should aim to be competitively priced against it so that we don't price out teams who could potentially want to purchase our

aircraft. This is dependent on our choices of components, especially our choice of engine, and the amount of testing that takes place in the development process.

- The plane is able to achieve near-maximum allowable race speeds
    - The plane should be able to fly within Mach 0.05 of the maximum speed set by the competition limitation of a Mach number of 0.4. This can be verified using calculations once the aircraft design has been set and all necessary parameters have been determined. Maximum speed is dependent on many factors of the aircraft design, such as power, drag coefficient, and surface areas of different aircraft components.

Stakeholder 5:Race Team

- Aircraft can be easily refueled in the race so the team can compete for success
    - Fuel tanks can hold up to the maximum fuel capacity allowed of 34 gallons of fuel. This is verifiable by ensuring there is enough volume in certain areas of the aircraft to hold this amount of fuel, focusing on the areas ahead or behind the cockpit. This is dependent on the design characteristics of the aircraft to ensure this is possible.
- Aircraft calibrations and systems are up to date
    - Aircraft must receive maintenance before the race to check the altitude calibrator to see if the accuracy is within +/- 50 ft. The altitude at which our aircraft will fly is vital in ISA or standard atmosphere calculations for the race team. This is verifiable through a systems check by the race team and is dependent on the pilot's ability to read the gauge correctly.

Stakeholder 6: Part Suppliers

- Parts should be easily manufacturable, potentially on a short timetable if needed
    - Parts must be delivered by using AIRSPACE services. The mission statement for this company is "When an aircraft is in need of a critical part, we provide real-time tracking and the fastest delivery times to help you get moving again" according to (BV, n.d.). This will improve the time to get mission-critical parts as they are needed throughout the design process. This

is verifiable through order confirmations and receipts and is dependent on what parts are mission-critical based on the design team's needs.

- Parts should be consistent and not change often for ease of manufacturing
  - Design decisions will be finalized two and a half weeks before the end of the semester, which allows for final calculations to be completed to determine aircraft performance. Once design changes are frozen, manufacturing could theoretically be prepared for since no further changes would take place.

Stakeholder 7: Environmental Organizations

- The aircraft should be non-harmful to the environment in emissions released(12)
  - Reduce impact on the environment by producing an average of 90 grams of CO_2 per passenger per kilometer (Graver, Rutherford and Zheng, 2013). Reducing the amount of greenhouse gas emissions within our aircraft design will have a positive impact on global warming and climate change. This is verifiable by a measurement of the engine production rate of CO_2 and dependent flight time in the air and how many passengers are within the aircraft.

2.3 PRELIMINARY RISK ANALYSIS

Within the world of aviation, there are many risks that not only the pilot has to face but also everyone on the ground who is potentially involved. Within this subsection, our team has developed a set of potential risks and possible mitigation strategies for each one. The risks in order from highest consequences to lowest consequences are as follows. The risk matrix for these risks can be found in figures 2.1 and 2.2.

Pilot Health

Pilot health problems such as heart attack, stroke, seizure, nausea, passing out, etc. could cause the pilot to lose control of the plane, causing it to crash into the ground, an obstacle, or another pilot, meaning the consequences of this happening is very high. The pilot is most likely to experience this risk when performing high-stress maneuvers or flying at high speeds/accelerations. The probability of a pilot health issue occurring is somewhat low, but not zero as there are no records online of this happening to racing pilots. Mitigation can occur by giving a thorough mental and physical health examination before each race, which will lower the likelihood of this happening but will not lower the consequences of a health crisis occurring.

Engine Power Loss

Engine power loss occurs when the engine overheats or if a component such as an engine rod breaks or a head gasket blows. Power loss could be as little as a few horsepower and could be as severe as total engine failure. In the most severe case, engine power loss could cause the plane to make a crash landing, making it on the higher side in terms of consequences as pilots could die as an effect of this happening. This is most likely to happen at cruise speed in the race as the pilot would be flying as fast as possible, pushing the engine to its limit, which could cause failure. The probabilities are typically low because plane engine manufacturers typically have very good quality control. Engine power loss could be mitigated by putting more strict quality control and quality engineering practices in place for the engineers and manufacturers. Mitigation would decrease likelihood but if engine failure occurred the consequences would be similar.

Aircraft Collision

Aircraft collision is very dangerous and typically results in pilot death. It occurs when two planes collide in mid-air. This could occur at all mission phases but is more likely to happen at takeoff/early stages of the mission as the planes are closer together when they all first take off. The consequence of this happening is almost always death between pilots. This doesn't happen very often in air racing so the likelihood is not super high (Though it did happen in 2023 (NBC news)). Mitigation can be done with pilot training for aircraft collision avoidance. It could also be mitigated by implementing motion sensors that control ailerons to avoid planes nearby. Mitigation would cause the likelihood to decrease.

Aileron/Rudder Control Failure

The plane system could fail in the aileron/rudder control, causing the pilot to lose control of the direction they want to plan to fly, ultimately causing it to crash. This could occur in a mission phase of heavy maneuvering as the plane ailerons/rudders would be under a lot of stress at that moment. The consequences are very high. If the plane lost control the pilot would likely die if they couldn't eject their seats. In terms of likelihood, it is not very likely, considering these systems are well-engineered. To mitigate this problem, more stressful quality control and reliable design would have to take place, which could lower the likelihood, but not the consequences were it to happen.

Ground Impact

Ground impact is defined as any case where the plane touches the ground. This can be either in landing or if a crash were to occur. In the worst case, ground impact would be a crash that kills the pilot, so the consequences are high. This risk typically happens in the landing stage for refueling of the plane in our mission. The extreme consequences are unlikely, as pilots have landed their planes many times. To mitigate this problem, the pilot could use an altimeter to see how fast they are descending and know what altitude they need to pull up at to land safely.

Spectator Safety

The spectators could be in danger of a plane crash, and if they are not careful and break rules such as running on the track or touching the planes, they could get hurt. Environment or mission phases that this could occur are before the race starts, takeoff, landing, and after the race ends, as well as in any maneuvers where the plane could lose control. The consequence of spectators being out of line could be very high, as people could die as a product of their/the pilot's actions. The likelihood of this happening is low assuming there are safety measures in place for the crowd. In terms of mitigation, keeping the crowd away from the planes will decrease risk, as well as having some kind of device that notifies the pilot if they are too close to the spectators. Having ample security to keep spectators off the track can also help. Mitigation would decrease the likelihood of the risk.

Bird Strikes

Though somewhat unlikely, bird strikes can occur in air racing. A bird strike is when a plane collides with a bird and can cause some serious damage. Depending on the speed of the plane, the bird could break the window of the plane and hit the pilot or it could damage the engine or wing and keep the plane from flying properly. Bird strikes can happen anywhere in the race, as birds can fly up to the maximum height the planes will be flying at. Bird strikes are pretty unlikely as there is a lot of space for the planes to fly and pilots can usually see the birds coming, but there is still a possibility. To mitigate bird strikes, they could have the race in an area where there aren't many birds, such as a desert or the ocean. They could also have a sensor that detects small objects in the air so pilots know to keep an eye out. Mitigation would decrease the likelihood but not change the consequences.

Human Error

Human error is when the pilot makes the wrong decision while racing which leads to something not ideal for the pilot. This could be making a bad turn, accidentally deploying landing gear, or using too much throttle among other things. This is a risk at all mission phases of the race, as human error can happen at any time. The consequences can be low such as losing some time in the race or high if crashing. The likelihood of this happening is somewhat low assuming the pilots are properly trained. To mitigate this happening, pilots can go through specific training to keep them safe if they mess up and make an error. After mitigation, the likelihood and consequences decrease since the pilot is more prepared for errors and has more practice flying.

Fuel Issues

The plane could have fuel issues such as a leak in one of the fuel lines causing the fuel to drain out/not get to the engine, or the fuel could not be filled up all the way. Fuel issues typically happen from faulty maintenance on the plane or wear/old age on the components. The consequences of this happening can be pretty high because the pilot could run out of fuel while in the air or the fuel could leak out and combust and damage the plane. Fuel issues are unlikely depending on your manufacturer/engineering team. To mitigate this from happening, a good maintenance crew on your team and good quality/reliability engineering from the manufacturer is necessary as well. Mitigation will lower the likelihood of fuel issues happening.

Weather Conditions

Weather conditions such as rain/storming, heavy wind, dense fog, and snow could affect the pilot's ability to fly safely and effectively, as they could impair the pilot’s vision or make them lose control. Weather conditions happen when they happen, but there isn’t a typical stage of the race that they occur in. Weather conditions can cause substantial damage, but are pretty avoidable so their likelihood is in the middle. Mitigation of weather risks involves looking at the weather forecast ahead of the race and planning a race on a day with clear skies and little wind. Mitigation will decrease likelihood and consequences as it is less likely to happen, but if it happened it would probably be a light occurrence.

Regulatory Compliance

Regulatory compliance involves the plane not being cleared to fly because it doesn’t meet the standards set but the FAA or the competition hosts. Regulatory compliance issues can include not having maintenanced the plane in a while or using out-of-date components in the plane. The consequences of this happening are low as they would make you not be able to fly in the race, so no one’s lives are at risk. The likelihood of this happening is in the middle because there are a lot of compliance standards set for the planes and teams are pretty good at maintaining their planes. To mitigate regulatory compliance issues, the plane should be maintenanced as needed and the team should be up to date on all the compliance rules. After mitigation, the likelihood will decrease.

Air Turbulence

Air turbulence occurs when the plane encounters air in the atmosphere that is irregularly flowing or in a vortex, from friction between the earth’s surface and the atmosphere. (weather.gov). This will happen in the mission when there are irregularities on the ground under which the plane is flying. The consequences are that the pilot loses a bit of control, but most pilots are used to turbulence. In the worst case, the pilot could fully lose control and crash but that is highly unlikely. Air turbulence is common for pilots to experience. To mitigate air turbulence, the race should be held over a flat area, which will minimize the likelihood.

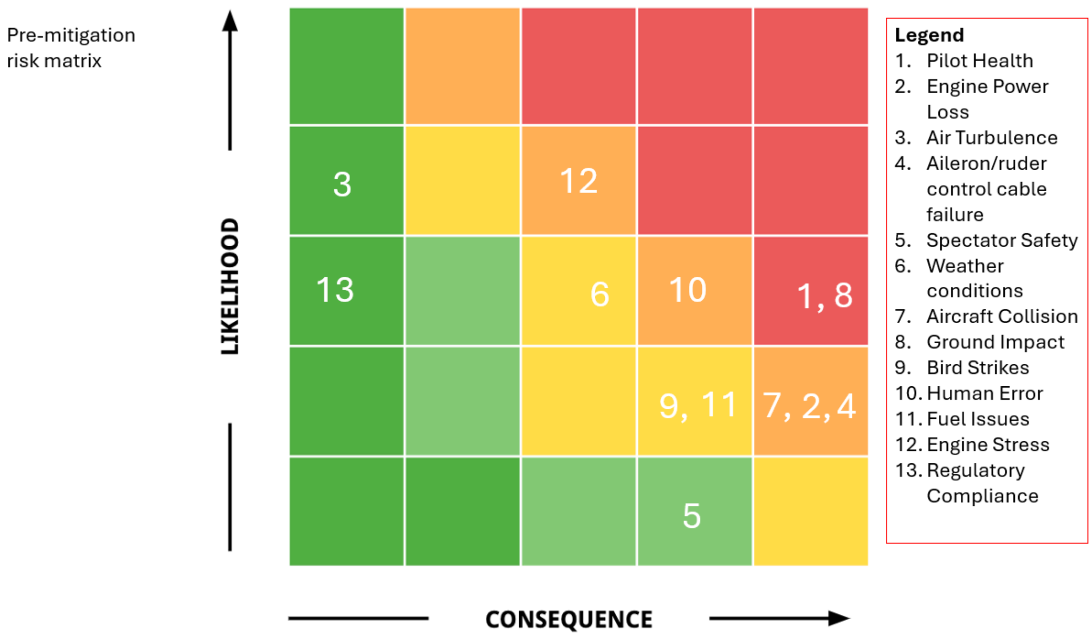

Figure 2.1: Pre-Mitigation Flight Risk Matrix

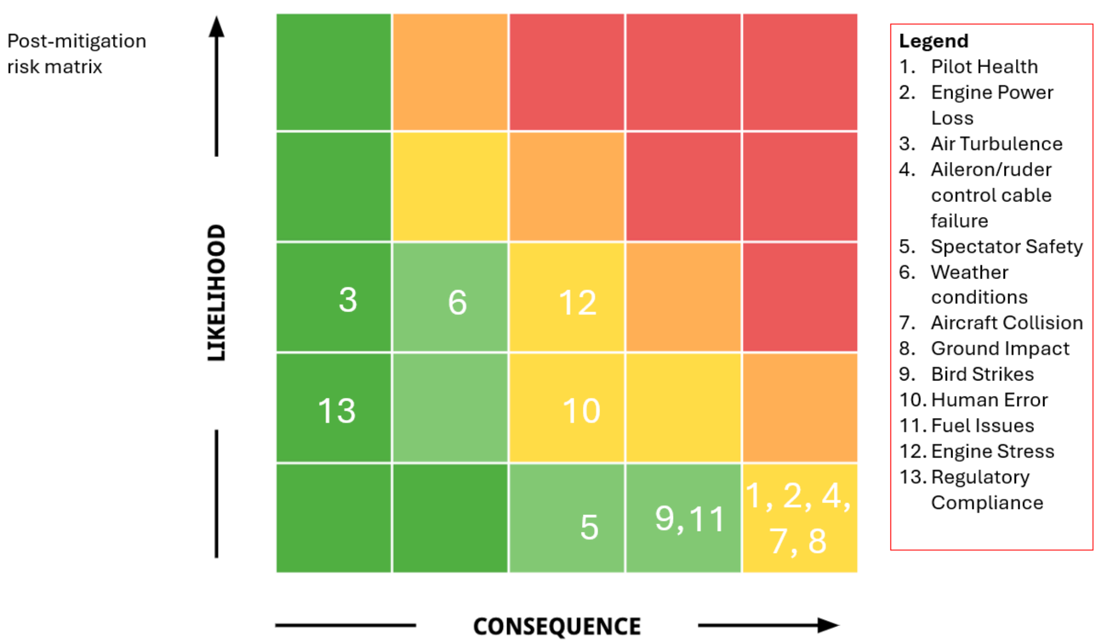

Figure 2.2: Post-Mitigation Flight Risk Matrix

3 ESTIMATING DESIGN PARAMETERS

In this section, a detailed description of the design phase of the aircraft that we designed will be covered. This includes a thorough description of the process, beginning with the descriptions of the aircraft used for initial reference, to initial estimations of the expected weights of the aircraft, the wing and tail design processes, takeoff and landing distances, and the airplane's $C_{D,0}$ value and drag polar.

3.1 EXISTING DESIGNS

Within this subsection, we will introduce three different types of reference aircraft similar to the final design we are aiming for. Not only will these reference aircraft be similar in physical characteristics but also will have similar mission profiles. The purpose of researching a variety of aircraft is to help influence minor and major design decisions down the road. Below will be a description of the three chosen reference aircraft our team decided to research, an image of the aircraft, and some aerodynamic limits supported by the aircraft.

Aircraft 1: Zivko Edge

The Zivko Edge 540 was designed as a lightweight aerobatic aircraft and was mainly used in the Red Bull Air Race. It is highly maneuverable thanks to its high roll rate and impressive climb rate. This combined with an impressive top speed makes it an ideal consideration for the USA 5000. Overall this is a great reference aircraft for our purposes and will serve as a great way to improve our design as the project progresses. Below in Figure 3.1 is an image of the aircraft and some design features that are useful. All data and images were found according to (Aero Corner, n.d.).

Figure 3.1: Zivko Edge 540

<u>Brief description of design features:</u>

- Can handle 10 continuous G's
- Top Speed: 265 mph
- Rate of Climb 3700 ft/min
- Empty weight: 1170 lbs
- Roll Rate: 420 degrees/sec
- Horsepower: 340 hp (single-piston engine)
- Max Weight: 1,800 lbs
- Wing Area: 98 ft^2
- Wing Span: 24 ft 5 in

Aircraft 2: Pitts S-2C Series

The Pitts S-2C Series aircraft, while slightly slower and less maneuverable than the Zivko Edge 540, is significantly easier to fly making it ideal for less experienced pilots. This aircraft is known for its very impressive aerobatics and many appearances at airshows around the world. Although it is a bi-plane with many features that will not come up in our design process it is still a great reference to help us be more creative in our decisions. Below in Figure 3.2 an image of the aircraft and some design features are given. All values and images below were found according to (Haun, 2016).

Figure 3.2: Pitts S-2C Series

<u>Brief description of design features:</u>

- Wing Span: 20 feet
- Wing area: 127.5 square feet
- Top speed: 184 mph
- empty weight = 1,155 pounds
- Max Gross Weight: 1700 pounds
- Rate of Climb: 2900 ft/min
- Horsepower: 260 hp (single-piston engine)
- Max G force: 6 Gs
- Roll Rate: 300 ft/s

Aircraft 3: Extra 330SC

The Extra 330SC, similar to the Zivko Edge, is a highly maneuverable Aerobatic Aircraft designed to compete in aerobatic competitions such as the Red Bull Air Race. Its combination of top speed and maneuverability makes it ideal for competing in the USA 5000. This aircraft will provide our team with many possible design features and could help us improve our design. Overall it is a great aircraft and is a great reference for future choices. Below in Figure 3.3 is an image and a description of the features of the aircraft. All values and images of this aircraft were found according to (Britt Lincoln, n.d.) and (www.aopa.org, 2019)

Figure 3.3: Extra 330SC

<u>Brief description of design features:</u>

- Max G force: +- 10 Gs
- Top Speed: 253 mph
- Horsepower: 315 hp (piston engine)
- Wing Area: 105,6 ft^2
- Empty Weight: 1.291 lbs
- roll rate: 420 degrees/sec
- Climb Rate: 3,200 ft/min

3.2 ESTIMATING GROSS TAKEOFF WEIGHT

The mission assigned to the aircraft for this competition is to fly the assigned course 25 times to complete a 5000-nautical mile air race as quickly as possible while equipped with a 34-gallon fuel tank. Each lap begins when the aircraft takes off from the runway at sea level, climbs to the first cruising altitude at 50 meters above mean sea level (MSL), and flies at this altitude for 49.5 nautical miles. This cruise period concludes with a distance of 1 nautical mile in which the aircraft must climb 740 meters to an altitude of 790 meters MSL for its second cruise phase. This cruise is also 49.5 nautical miles long before the aircraft reaches the far end of the course and must turn around towards the start line again. The aircraft must then return to the start line along the same path. Once this out-and-back flight is completed, the aircraft has the option of landing and refueling, or immediately reversing course and continuing along the race course again to complete another lap.

The aircraft that we designed aimed to complete this task by completing as many simultaneous laps as possible while flying at speeds as near to the limitation of Mach 0.4 set in the competition guidelines as possible. We used the values in the aircraft sizing code to verify our expected fuel consumption against the number of laps that we would have to fly, and we determined that two laps would be the most laps we could complete before needing to refuel in order to save time. Based on this discovery, which is covered later in this section, we updated our mission diagram to its final form, which is visible in Figure 3.4.

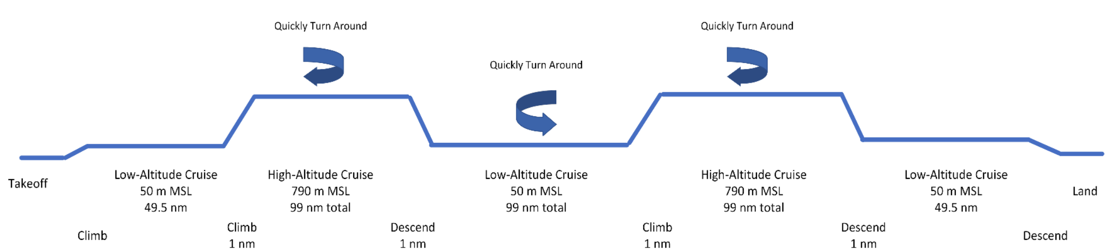

Figure 3.4: Mission Diagram for Race Aircraft

To determine certain characteristics of the aircraft, such as its fuel consumption and weight, we utilized a sizing code based on equations from Chapter 3 of the Raymer textbook

(Raymer, 2006). We chose not to implement a looping system to determine the maximum number of laps possible on the given fuel load, so we calculated the maximum mass of fuel available based on the maximum fuel tank capacity of 34 gallons and the fuel density of 100LL aviation fuel. This provided an answer of a maximum fuel load of 92.69 kg of fuel available to burn during each flight before refueling.

We chose to use Metric units for a majority of the sizing code, although values for cruise fractions were English because of the fact that our range units were in nautical miles and we had constants in these equations in English units. We chose to use a weight of 81 kg for the pilot because this is the average person's weight in North America (Walpole et al., 2012). We used the target flight speed of Mach 0.4 because this was the competition limit we hoped to be flying at. For other values, such as the constants being used in the calculations for empty weight and the specific fuel-to-weight ratio for cruise flight, values chosen to represent these constants were based on similar aircraft types, largely from Chapter 3 of the Raymer textbook (Raymer, 2006).

Our sizing code iterated through the equations we were using each time, first plugging an initial guess value and later using the last calculated value for the aircraft's maximum takeoff weight to find an updated maximum takeoff weight. These iterations were completed until the most recent calculated value was within one kg of the previously calculated value, suggesting that this value is now close enough to the true value to be valuable.

To find the takeoff weights of the aircraft assuming that it completes a certain number of laps, we set the code to calculate the maximum takeoff weight for the aircraft for the number of laps entered as the variable numLaps. The results from different numbers of laps are included in Table 3.1. Based on the results, the greatest number of laps that can be completed without exceeding the maximum fuel capacity was determined to be 2 laps.

Number of Laps	Mass of 34 Gallons of Fuel (kg)	Mass of Fuel Consumed (kg)	Max Takeoff Mass (kg)	Max Takeoff Mass (lbs)
1	92.687	48.453	540.71	1191.73
2	**92.687**	**87.810**	**699.40**	**1541.47**
3	92.687	155.079	968.27	2134.06

Table 3.1: Sizing Code Results

3.3 WING LOADING

The requirements outlined in this project that affected the wing loading on the aircraft were the weight of the aircraft, the efficiency of the aircraft, the maximum/minimum takeoff and landing distances, and the ability to fly in certain weather. These requirements are important because they are all dependent on the weight of the aircraft, which directly affects the wing loading. After finding the maximum takeoff weight and the wing area of the Zivko Edge 540, we divided its maximum takeoff weight by the total wing area to find a wing loading to be 18.37 lb/ft^2. This value is acceptable since it fits within the range of light, short-range civil aircraft provided in Chapter 6 of the Nicolai (2010) textbook, since it suggested a range from 10-30. Based on resources that suggested that higher wing loading helps to increase the maximum speed of the aircraft due to lower drag, we chose to then increase this wing loading value by 2 to 20.37 lb/ft^2 to hopefully increase the top speed relative to the Edge 540 (Cavagnaro, 2019). Higher wing loadings offer the ability to reach higher top speeds due to the fact that a smaller wing will produce less drag due to having a smaller area. This value is also close to the Zivko Edge 540 reference aircraft's wing loading while at empty mass, which is 12.4 lb/ft^2 (Wikiwand, n.d.) and is just over 18 lb/ft^2 according to the article written by Cavagnaro (2019).

3.4 WING SIZE AND AIRFOIL CHARACTERISTICS

While determining the wing size that our aircraft should have, we chose to continue using the Zivko Edge 540 for inspiration and to confirm that the values we were calculating were reasonable for an aircraft of this type. We chose this aircraft as many aspects of this competition are similar to the Red Bull Air race in which this aircraft competed. This plane was built for high-speed maneuvering which our aircraft will have to undergo as well as being capable of a speed of over 250 mph. While this is lower than our stated maximum possible, this was achieved with a lower-power engine than what we are allowed to use.

To determine the wing size of the aircraft, we first examined the limitations posed on our design by the racing organization of the total sum of linear dimensions to be less than 20 meters. We used our expected wing loading for our aircraft. Using the value of 20.37 lb/ft^2, we calculated the wing area from the maximum takeoff weight and wing loading values by plugging them into the equation $S = W_0 / Wing\ Load$. Solving for S, we get a wing surface area of 75.67 ft^2.

We then found the aspect ratio of the Zivko Edge 540, which we calculated to be 6.125 by applying the equation $AR = \frac{wing\ span^2}{wing\ area}$ (Aero Corner). Using this equation once again, with the aspect ratio from the Edge 540 and the wing area as determined above, we calculated a wing span of 21.53 feet. Now, reordering this equation and exchanging the aspect ratio for its other definition of wingspan divided by chord length, we found the equation for the chord length to be c = b / AR. This allowed us to calculate a chord length of 3.515 feet.

Our primary parameters for choosing an appropriate airfoil were a high maximum C_L, a high maximum C_L/C_D, and good lift characteristics at a low angle of attack. In addition to these, different locations of characteristics of the airfoil shape can lead to different general flight characteristics from the airfoil. Firstly, we chose to try to find a NACA airfoil family which most aligned with the characteristics we wanted for our aircraft design. Using a document put together by Stanford University, we judged our aircraft to most align with the performance characteristics associated with the NACA 6 series of airfoils. This was because the 6-series airfoils were designed to increase the laminar regions of the airflow around the airfoil, decreasing drag. They are also optimized for high speeds and have been used in piston-powered fighter aircraft, which we determined were somewhat similar to the design application we desired. The airfoil characteristics referenced can be seen in Figure 3.5 (The NACA Airfoil Series, n.d.).

Family	Advantages	Disadvantages	Applications
4-Digit	1. Good stall characteristics 2. Small center of pressure movement across large speed range 3. Roughness has little effect	1. Low maximum lift coefficient 2. Relatively high drag 3. High pitching moment	1. General aviation 2. Horizontal tails ***Symmetrical:*** 3. Supersonic jets 4. Helicopter blades 5. Shrouds 6. Missile/rocket fins
5-Digit	1. Higher maximum lift coefficient 2. Low pitching moment 3. Roughness has little effect	1. Poor stall behavior 2. Relatively high drag	1. General aviation 2. Piston-powered bombers, transports 3. Commuters 4. Business jets
16-Series	1. Avoids low pressure peaks 2. Low drag at high speed	1. Relatively low lift	1. Aircraft propellers 2. Ship propellers
6-Series	1. High maximum lift coefficient 2. Very low drag over a small range of operating conditions 3. Optimized for high speed	1. High drag outside of the optimum range of operating conditions 2. High pitching moment 3. Poor stall behavior 4. Very susceptible to roughness	1. Piston-powered fighters 2. Business jets 3. Jet trainers 4. Supersonic jets
7-Series	1. Very low drag over a small range of operating conditions 2. Low pitching moment	1. Reduced maximum lift coefficient 2. High drag outside of the optimum range of operating conditions 3. Poor stall behavior 4. Very susceptible to roughness	Seldom used
8-Series	Unknown	Unknown	Very seldom used

Table 3.2: NACA Airfoil Families and Characteristics

As we were considering the thickness of the airfoil that we wanted to use, we used information from Chapter 7 of *Fundamentals of Aircraft and Airship Design* to determine that for high C_L max values, thicknesses between 12 and 16 percent of the chord length produce highly effective and efficient results. Because we would want the overall weight of our wings to remain small, we chose to use an airfoil with a thickness of 12% of the chord length. Based on information later in Chapter 7 regarding the different locations of the maximum thickness of the airfoil, the textbook recommends having the maximum thickness position relative to the chord further back so that the flow can remain laminar over the wing. Although this is important, based on Figure F.3 in Appendix F in the Nicolai textbook, it appeared that 63 series airfoils produced less drag at higher lift values than the 66 series airfoils, even though the 66 series produces less drag at lower C_L values, as seen in Figure 3.6. Because of this, we determined that allowing the maximum thickness to remain further forward wouldn't be detrimental to our design, and we selected the 63 series of airfoils. Regarding airfoil camber, positive camber increases the C_L at specific angles of attack, shifting the $C_{L\alpha}$ graph to the left from where it would be for a symmetrical airfoil. Because of this, and based on the information in Figure 7.6 in the Nicolai textbook, we decided to primarily look at cambered wings around the 4% values. From this set of choices, we were leading immediately towards the NACA 63-412 airfoil to use for our aircraft, which can be found pictured in Figure 3.7. In addition to these design choices made up to this point, we cross-checked our assumptions against the table of airfoil information in Appendix F of the Nicolai (2010) textbook, where we found NACA 63-412 to have nearly the highest maximum $C_{L,\,max}$ of any airfoil in Table F.1 on page 718, a table of many airfoils of different families.

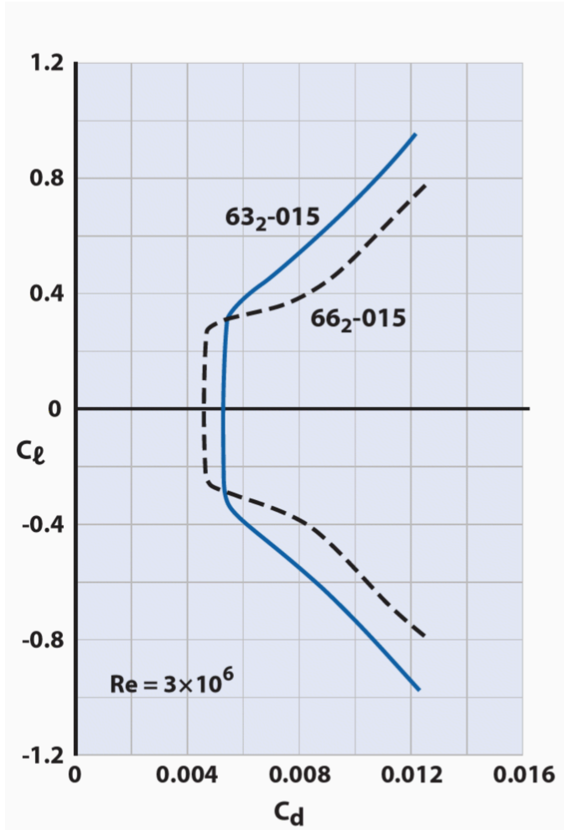

Figure 3.5: NACA Airfoil C_L vs C_d Comparison

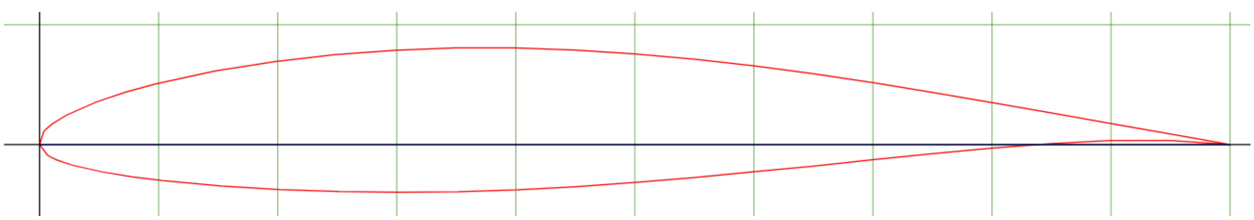

Figure 3.6: NACA 63-412 Airfoil

From: (*NACA 63-412 AIRFOIL (n63412-il),* n.d.)

For our tail airfoils, we decided to use a symmetrical airfoil because this allows the airfoil to not apply any unwanted movement on the aircraft in level flight. Reducing the amount of elevator deflection in level flight also decreases the drag that the aircraft will experience, as the airflow around the tail surfaces is disturbed less. To fulfill this requirement, we looked at the performance and characteristics of different NACA symmetrical airfoils, and based on their performance characteristics at low angles of attack, we chose the NACA 0012 airfoil for our tail surfaces. The NACA 0012 airfoil can be seen in Figure 3.8.

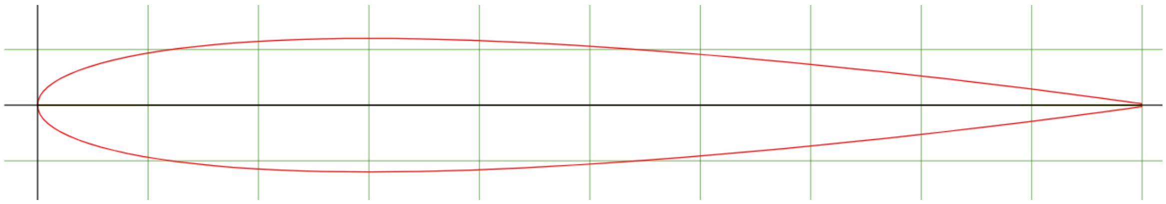

Figure 3.7: NACA 0012 Airfoil
From: (*NACA 0012 AIRFOILS (n0012-il)*, n.d.)

3.5 TAIL SIZING

To determine the configuration of the aircraft's horizontal and vertical tails, as a team, we took advantage of the tail volume coefficient approach. This approach is described within Chapter 11 of the Nicolai (2010) textbook and will be used as a sufficient way to determine our preliminary tail sizing. To begin using this approach our team first looked at designing the horizontal tail for longitudinal stability and estimating a horizontal tail volume based on Table 11.8 in Nicolai's (2010) Chapter 11. This table states the typical values of volume coefficients for different types of aircraft. The best and most similar class of aircraft on this table compared to our reference aircraft the Zivko Edge is "General aviation (one engine propeller)". The corresponding horizontal tail volume (C_{HT}) is 0.7. From this estimation, our team then found the horizontal tail area (S_{HT}) by using the trainer design tool (rcplanes.online, n.d.). This tool gave us an approximation of the horizontal tail area being about 20% of the wing reference area (S_{ref}). From our calculated wing reference area, 75.67 $ft^{2,}$ and 20% of this value gives a (S_{HT}) of 15.135 ft^2. Finally, after making these design decisions based on certain reference material we can use the equation given in Nicolai Chapter 11 to determine the distance from the horizontal tail to the center of gravity of our aircraft or C.G. which is denoted by l_{HT}. The equation is given below in (Equation 3.1).

$$C_{HT} = (l_{HT} * S_{HT})/(c * S_{ref}) \qquad \text{(Equation 3.1)}$$

(C_{HT}) is the volume coefficient of the aircraft's horizontal tail. (S_{HT}) is the area of the horizontal tail. (c) is the wing mean aerodynamic chord. This is found by using the calculated wing chord for the aircraft found earlier, which is 3.515 ft. The (l_{HT}) variable is the distance from the initial estimate of the c.g location. Solving for this value and plugging in the other estimations we get a final value of (l_{HT}) to be 13.41 ft. To give an idea of where all these values are being implemented, a sketch is provided below in Figure 3.9 demonstrating how the horizontal tail configuration was designed.

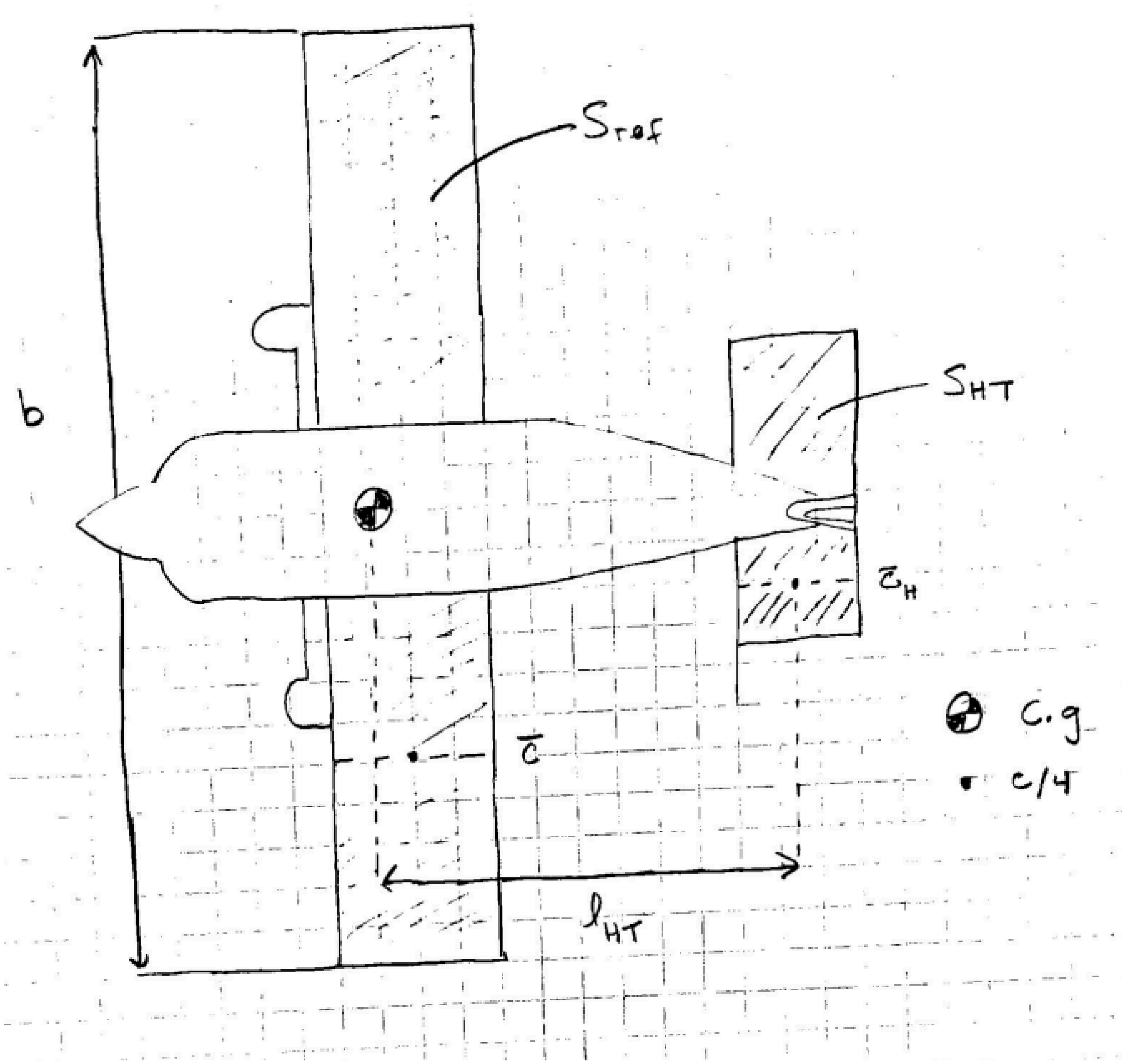

Figure 3.8: Initial Horizontal Tail Configuration

On the other hand, The vertical tail is primarily used to provide directional control of the aircraft and is a very important design aspect. To determine the location of the vertical tail on the fuselage our team followed the same approach as the horizontal tail. To determine the volume coefficient (C_{VT}) we used the same table in Nicolai Chapter 11 and found that for the "General aviation (one engine propeller)" a value of 0.032 was sufficient for the vertical tail volume coefficient. Next, we found a value for the vertical tail wing area. To determine this our team also used the design tool from (rcplanes.online, n.d.) which showed that the vertical stabilizer area is 33% of the horizontal stabilizer area or the wing area of the horizontal tail area. Taking 33% of S_{HT}, we found S_{VT} to be 5.031 ft^2. With this estimation, our team used the following formula to calculate the distance from the CG to the vertical tail or the l_{VT}

$$C_{HT} = (l_{VT} * S_{VT})/(b * S_{ref}) \qquad \text{(Equation 3.2)}$$

All terms in this equation have the same definition as for the horizontal tail formula and the S_{ref} has the same value as before. Though the new term b is the wingspan of the aircraft in ft. Using our reference aircraft of the Zivko Edge 540 we found that the wingspan is 24 ft 5 in. When plugging in all estimated numbers, we found l_{VT} to be 11.8405 ft. A sketch of this design location of the vertical tail is given below in Figure 3.10.

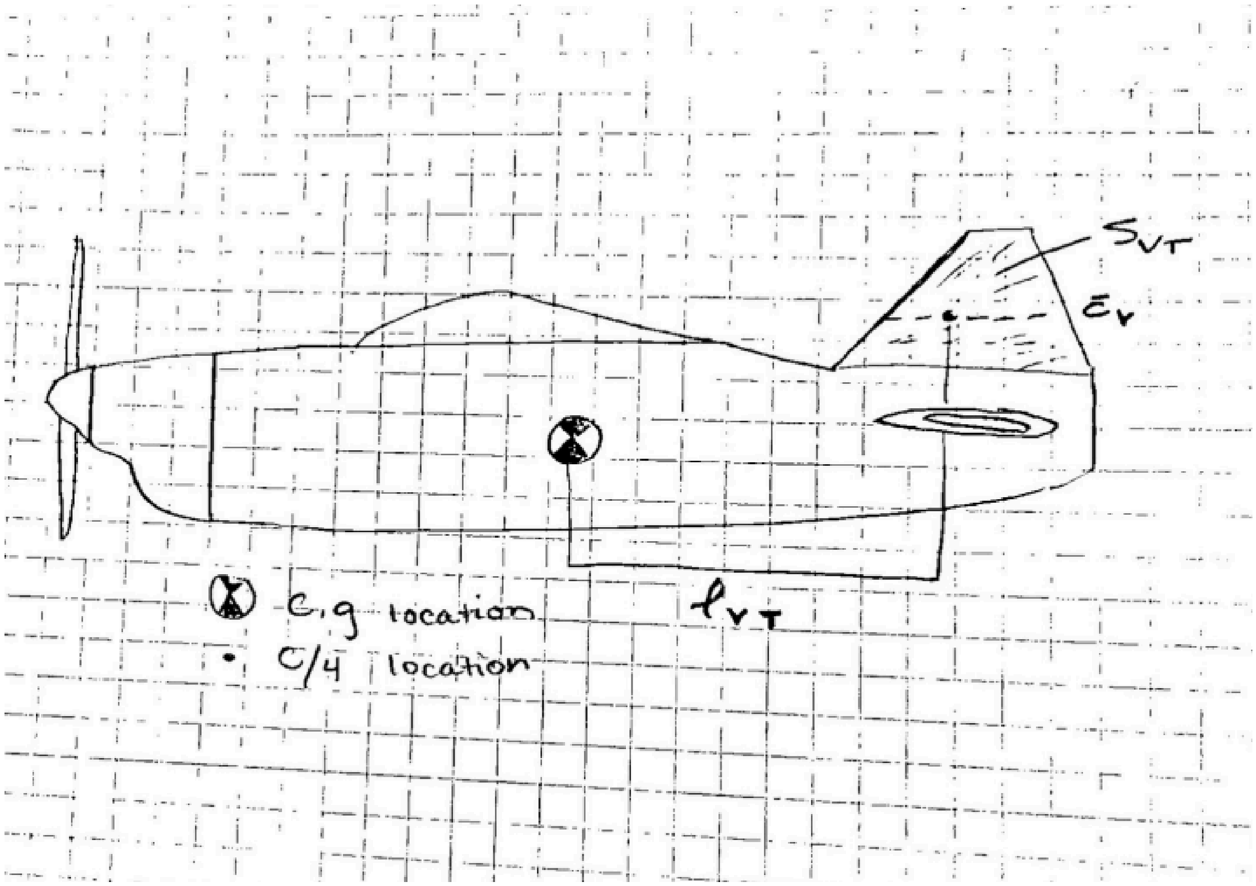

Figure 3.9 Initial Vertical Tail Configuration

3.6 TAKEOFF AND LANDING DISTANCES

In order to calculate the maximum takeoff distance we first have to determine the liftoff total weight. The takeoff weight is equal to the Gross takeoff weight calculated through the aircraft sizing code in section 3.2. We find this value to be 1191.73 lbs. Additionally, we also calculated our Cl_max and wing area values. In order to find the thrust, we take our maximum sea level power and divide it by our takeoff speed. The equation for takeoff speed is $V_{takeoff} = 1.2 * V_{stall}$ and $V_{stall} = \sqrt{\frac{2*W_{takeoff}}{\rho_h Cl_{max} S}}$. σ is equal to the ratio of current air density to sea-level density. Then we use the takeoff distance equation:

$$S_{TO} = 20.9 \frac{W/S}{\sigma C_{L_{max}} (T/W)} + 69.6 \sqrt{\frac{W/S}{\sigma C_{L_{max}}}}$$ (Equation 3.3)

Using this equation we find that the total takeoff distance is 247.2663 ft (75.3668 m). This distance is considerably shorter than the runway length (1500 m) meaning that the aircraft is capable of taking off fully loaded.

The equation for the landing distance is similar however, since we used a significant amount of fuel, we find that our landing weight is equal to W_empty/(Wlioter/W_landing) which we calculated as being 992.352 lbs.

$$S_L = 79.4 \frac{W/S}{\sigma C_{L\max}} + 50/\tan\theta_{\text{app}}$$

(Equation 3.4)

The calculated landing distance is equal to 1294.3 ft (394.5026 m). This value is acceptable as it is considerably below the maximum length of the runway (1500 m), meaning that the aircraft can land on it.

3.7 AIRPLANE $C_{D,0}$ AND DRAG POLAR DETERMINATION

To understand the characteristics exhibited by our aircraft during flight, we needed to calculate the $C_{D,0}$ of our aircraft while accounting for the wings, fuselage, tail, and landing gear. In order to save weight and increase the range of our aircraft at high speeds, we selected the fixed landing gear option for our aircraft, meaning that landing gear will have a reasonably large effect on the drag coefficient of the aircraft.

First of all, we calculated the $C_{D,0}$ for the wings, using Equation 13.13 from the Nicolai (2010) textbook, listed as Equation 3.4 in our report. Using our wing chord of 3.515 feet, the air density at 50 meters above sea level, our estimated flight speed of 267 miles per hour, and the dynamic viscosity of air at this altitude, we calculated that our aircraft would have a Reynolds number of 8,740,000. Based on the values of C_f shown in Figure 2.6 from the Nicolai (2010) textbook, which is shown here in Figure 3.11, we determined that the C_f value for our aircraft was 0.0008. Plugging this value, the maximum thickness to chord ratio of our wing, estimated wetted wing area, wing reference area, and the R-value for air, we calculated that our value of $(C_{D,0})_w$ value was 0.0059.

$$\left(C_{D0}\right)_W = C_f\left[1 + L\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4\right] R \frac{S_{\text{wet}}}{S_{\text{ref}}} \qquad (13.13)$$

(Equation 3.5)

In order to calculate the drag value associated with the tail, we used Equation 3.4 again and Equation 13.13 from the Nicolai (2010) textbook. Since our tail uses a NACA 0012 airfoil with a reference area of 15.135 ft^2 and a chord length of 1.57 feet, we determined the drag due to the tail airfoils to be 0.0178.

Next, to calculate the drag force acting on the fuselage of the aircraft, we continued referencing Chapter 13 of the Nicolai textbook. According to the textbook, on aircraft similar to the one we have designed, skin drag is the primary source of drag caused by the fuselage (Nicolai, 2010). Using data from our reference aircraft and our own design, we were able to determine the skin drag we could expect to see from our aircraft. Using the same value of C_f we found previously, 0.0008, and using a body length value of 6.27 meters, just over 20.5 feet, from the Edge 540 and our own design of the same length, a fuselage diameter of 4.3 feet, a wetted area for the body surface of 248 ft^2, and a largest cross section value of 15.5 square feet, we plugged these into Equation 3.5 (13.22 from the Nicolai (2010) textbook) to get a value for $(C_{Df})_B$. Completing this calculation gives a drag coefficient value of 0.0583.

$$\left(C_{Df}\right)_B = C_f\left[1+\frac{60}{\left(\ell_B/d\right)^3}+0.0025\left(\frac{\ell_B}{d}\right)\right]\frac{S_S}{S_B} \qquad (13.22)$$

(Equation 3.6)

After computing this value, we computed the value of C_{Db}, which is the drag created by a blunt surface at the rear end of a body. Since the aircraft fuselage we designed was entirely tapered out and concludes with the rear of the tail surface, it was determined that the value for d_b, the diameter of the blunt surface, was zero. With this being the case, any values that we would enter into Equation 3.6 and Equation 13.23 from the Nicolai textbook, would still lead to a final calculated value of C_{Db} being zero.

$$C_{Db} = 0.029\left(d_b/d\right)^3 \Big/ \sqrt{\left(C_{Df}\right)_B} \qquad (13.23)$$

(Equation 3.7)

The sum of the drag coefficient for the entire body is the sum of the last two drag components which we had previously calculated, which can be seen in Equation 3.6. Because C_{Db} is zero, this means that the total body drag number is equal to the previously calculated value for $(C_{Df})_B$, or 0.0583

$$\left(C_{D0}\right)_B = \left(C_{Df}\right)_B + C_{Db}$$

(Equation 3.8)

Another factor of the aircraft drag that needed to be taken into account was the drag caused by the shape of the canopy on top of the fuselage. In Chapter 13 of the Nicolai (2010) textbook, in Figure 13.16, there is a description of values for different pointed bodies with parallel center sections. This is included in Figure 3.11, which is what we chose to generally design our canopy. We designed the canopy with the length of its tapering rear portion being twice as large as the tapering of the front segment of the canopy. We also designed our center linear portion to be around the same length as the length of the front portion. With our forward tapering portion being around 1.3 meters long with the diameter of the canopy section being 1 meter, we were able to calculate the drag due to the canopy to be 0.0296.

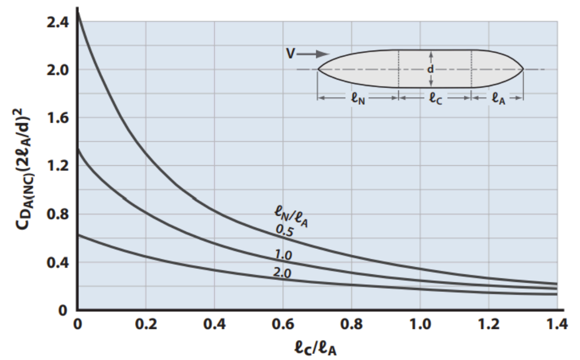

Figure 13.16 Interference drag for pointed bodies with parallel center section.

Figure 3.10 Interference drag for pointed bodies with parallel center section

Now that each of these values has been calculated, we can begin calculating the total value of the $C_{D,0}$ of the aircraft from these values. Firstly, using equation 13.28 from the Nicolai (2010) textbook, which is Equation 3.8, we can calculate the value of $(C_{D0})_{WB}$. With the $(C_{D0})_B$ value from previous calculations, the largest fuselage cross-section, wing reference area, $(C_{D0})_W$ value, and other values are included in ΔC_{D0}. In this ΔC_{D0}, we included the drag coefficient of our landing gear, which we based off of known values from similar aircraft, concluding that we would use 0.005 for the gear. The values from the canopy section are grouped into the main fuselage $(C_{D0})_B$ value. Plugging all of these values into the equation for $(C_{D0})_{WB}$, we can find its value to be 0.0229.

$$\left(C_{D0}\right)_{WB} = \left(C_{D0}\right)_B \frac{S_B}{S_{\text{Ref}}} + \left(C_{D0}\right)_W + \Delta C_{D0} \qquad (13.28)$$

(Equation 3.9)

$$(C_{D0})_{WB} = (0.0583 + 0.0293) \cdot \frac{15.5}{75.67} + 0.005 = 0.0229$$

Now, this value and others can be entered into the next equation, Equation 3.9, which is Equation 13.31 from the Nicolai textbook (2010). The $(C_{D0})_{WB}$ value is 0.0115, the $(C_{D0})_{Wing}$ value used is 0.0059, S_{VT} and S_{HT} are 10 square feet and 15.135 square feet respectively, S_{ref} is 75.67 ft^2, and $S_{nacelle}$ is zero because we do not have any engines using nacelles. Collectively, applying these and previously calculated values into the equation, we arrive at the final C_{D0} value for the aircraft, which is calculated at 0.031.

$$\left(C_{D0}\right)_{a/c} = \left(C_{D0}\right)_{W/B} + \left(C_{D0}\right)_{wing} \frac{S_{VT} + S_{HT}}{S_{ref}} + \left(C_{D0}\right)_{fuse} S_{nacelle} / S_{ref} \qquad (13.31)$$

(Equation 3.10)

$$C_{D0} = 0.0229 + (0.0059 + 0.0178) \cdot \frac{10+15.246}{75.67} = 0.0309$$

From this calculated value of C_{D0}, we were then able to plot the drag polar of our aircraft based on the other already determined aircraft design characteristics. By inputting our wing aspect ratio, a set of C_L values from analysis of our airfoils, and the Oswald efficiency factor of our wing shape, we were able to create the drag polar plot seen in Figure 3.12.

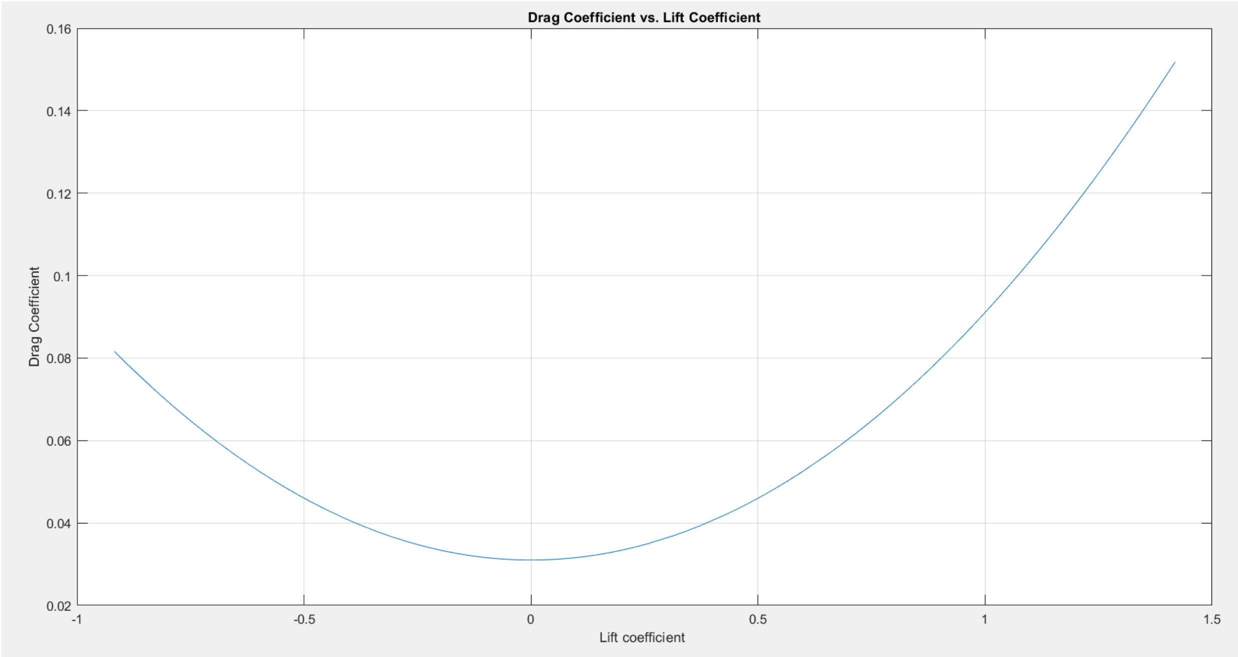

Figure 3.11: Drag Polar Plot

In this same process, we also created a plot of the drag coefficient vs flight velocity. This only required additional inputs of our aircraft weight, and a range of speed values to plot the C_D over. This plot can be found as Figure 3.13.

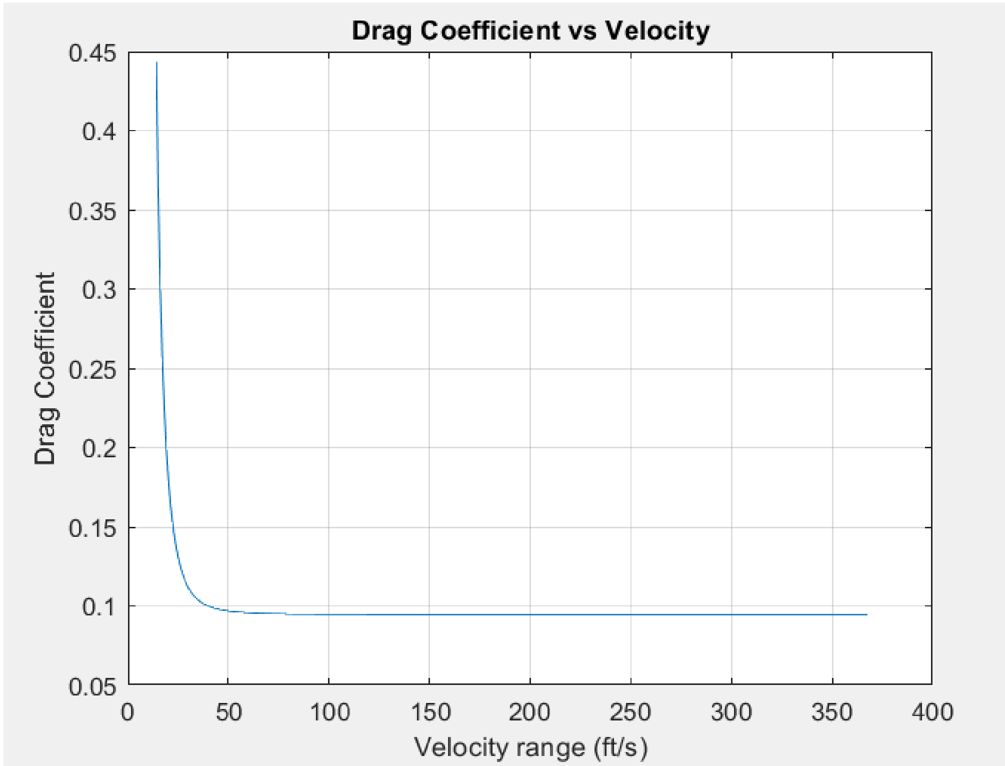

Figure 3.12: Drag Coefficient vs. Velocity Plot

4 DETAILED CONCEPT, SELECTION, AND PERFORMANCE ANALYSIS

In Section Four, the concept generation, selection, and refinement process is described, with the decision processes that led to the final aircraft design being described in detail. The section also includes a performance analysis of the final aircraft design, describing important characteristics such as race speed, range, and projected race completion time.

4.1 CONCEPT GENERATION

After defining the major design features of the aircraft and looking at other aircraft for inspiration, we were able to begin concept generation for our plane. This included deciding what kind of wings to use and where they will be placed, how many engines it will have and where they will be placed, and what kind of landing gear it will have among other features. We came up with three concepts, the first having a high wing layout and fixed landing gear, the second having a low wing design with a fixed landing gear, and the third having a low wing design with retractable landing gear. These concepts are shown below in Figure 4.1 and Figure 4.3 . An explanation of each one is also given.

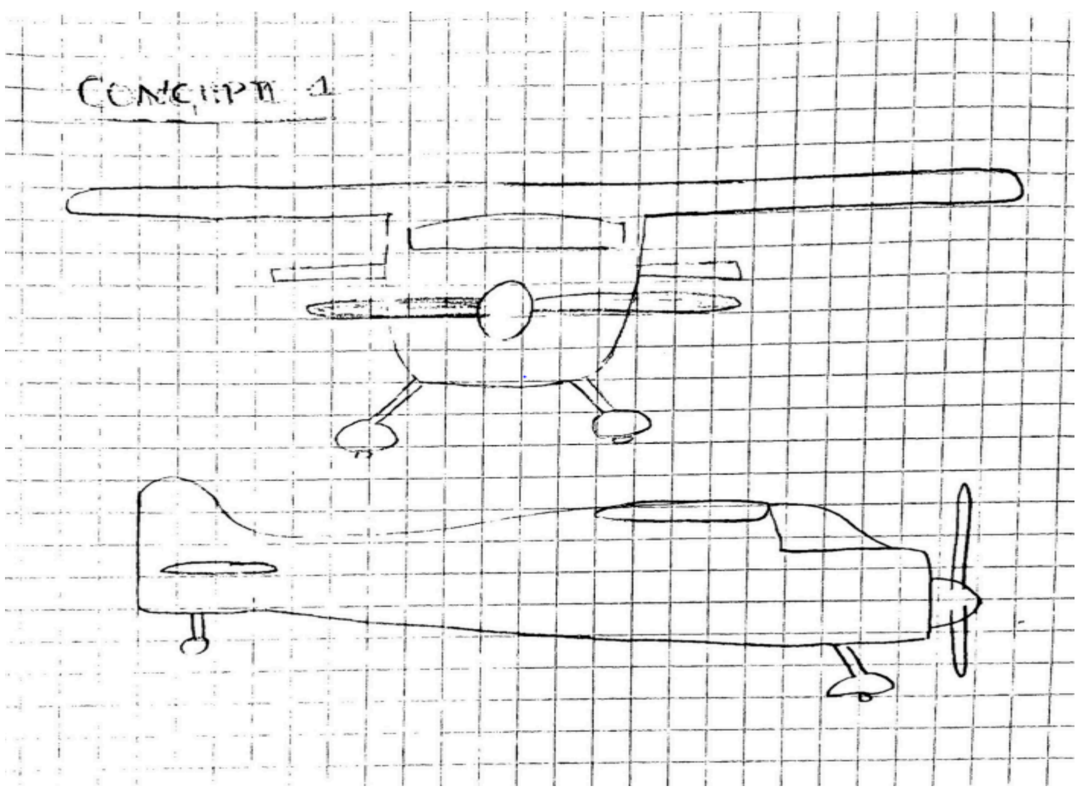

Figure 4.1 Airplane Concept 1

Concept one incorporates a high wing layout, fixed landing gear, and one engine mounted in the front of the aircraft below the wings. We chose this concept because there are multiple advantages of having a high-wing layout. One of those advantages is that the fuselage of the aircraft acts as a pendulum to increase roll stability, in other words, the center of gravity sits below the wing (Herbert, 2019). Not only will this increase stability but also help the pilot stay on track. However, a disadvantage of the high wing layout could be that it will increase profile drag and interference drag which affect the cruise performance of the aircraft (Herbert, 2019). Our team chose to keep the concepts to one engine due to the fact there is a horsepower limit and also because adding another engine would cause our weight and drag to increase. The one engine and high-wing aircraft were a great initial concept to get ideas flowing within our team but overall have less aerodynamic efficiencies than the low-wing layout that is described in concept two below in Figure 4.2.

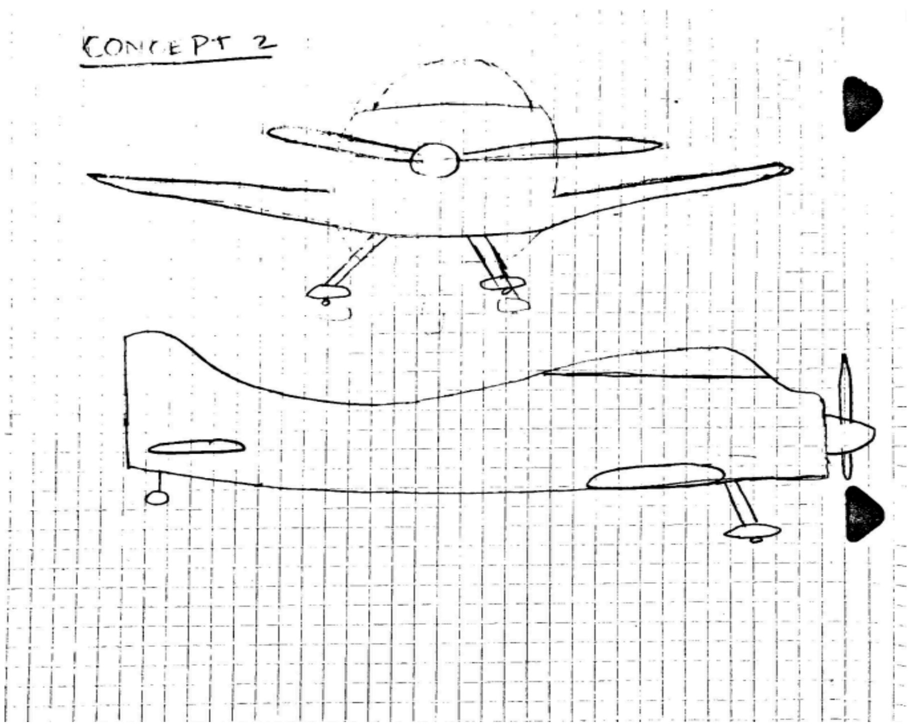

Figure 4.2 Airplane Concept 2

For this concept, our team has chosen to go with a low-wing layout with fixed-pitch landing gear and one engine mounted to the front of the engine. This layout has many benefits and gave many important ideas for our final design. As said before the low-wing design will increase all aerodynamic efficiencies compared to the high-wing layout. This includes greater cruise performance which is a big part of the race due to the long stretches cruise. For the fixed landing gear, our team decided that having constant drag throughout the race would not harm the overall design of the aircraft enough to justify the additional weight. It also helps with determining the other design parameters such as the range of takeoff, landing, and cruise. Our last design concept would be the low-wing layout with one engine and retractable landing gear. This choice was made to see if decreasing drag during the cruise would decrease race time. Overall these three concepts gave great oversight into what our team had to do to determine the most efficient aircraft design.

4.2 CONCEPT SELECTION

Detailed analysis of the three concepts led to the choice of a low-wing aircraft with fixed landing gear in a taildragger configuration. The fixed landing gear was chosen, even though the retractable landing gear had a lower drag coefficient, because the weight of the plane increased too significantly with the retractable landing gear, causing the range to reduce enough to reduce the number of laps that we projected the aircraft to complete before refueling. The range is important in deciding the best option for the plane because a higher range means that the plane will have to refuel less during the race. Refueling causes more time lost in the race so we want to have the least possible refuels during the race for our plane. Along with the range and weight of the aircraft, we also considered other aerodynamic factors such as the total drag on the aircraft. With a low-wing design, the drag that occurs during the cruise is lower than that of the other wing layouts discussed in section 4.1 of this report. This will have an extreme effect on race time and other factors mentioned before such as range. With there being constraints on the aircraft dimensions, such as the sum of linear dimensions (wingspan + fuselage length + airplane's height with extended landing gear) < 20 m, concept two seemed like the best choice. This is because the tail dragger landing configuration allows for less vertical height. Overall, the selection of concept two was based on three main criteria, being minimizing the total drag on the aircraft, maximizing the range of the aircraft, and constraints put on the aircraft by the organization that created the USA 5000 air race.

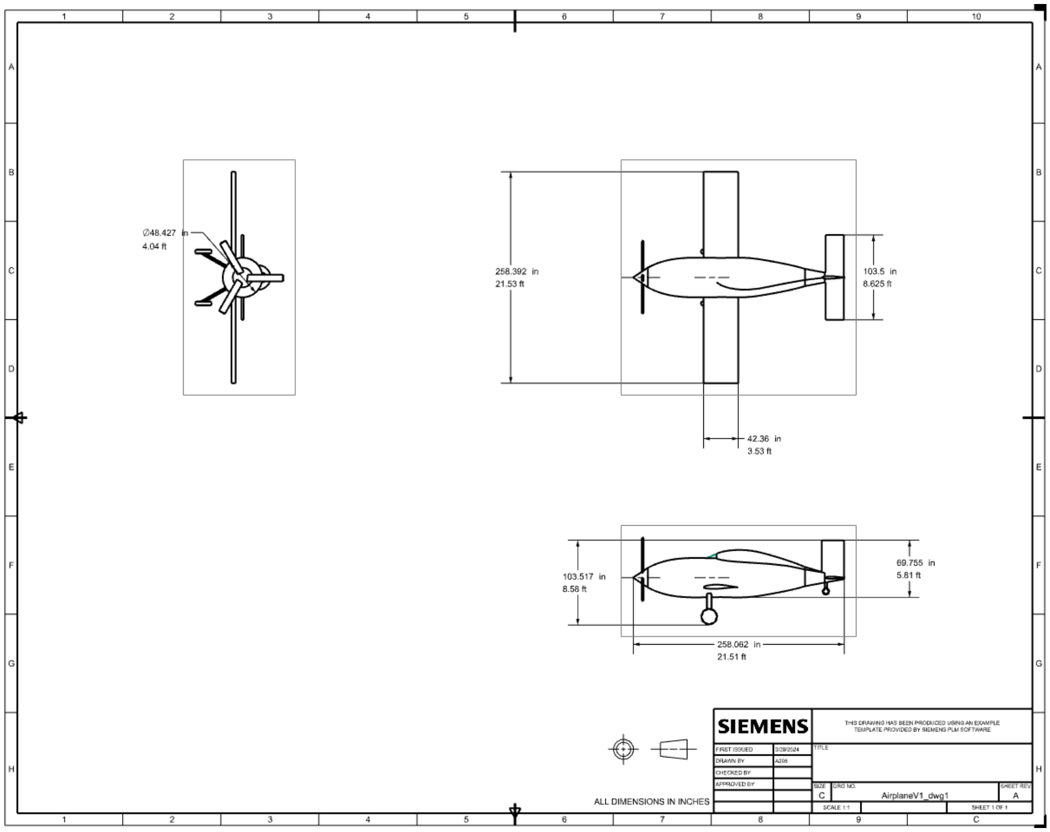

Figure 4.3: Drawing with Dimensions of the Aircraft Design

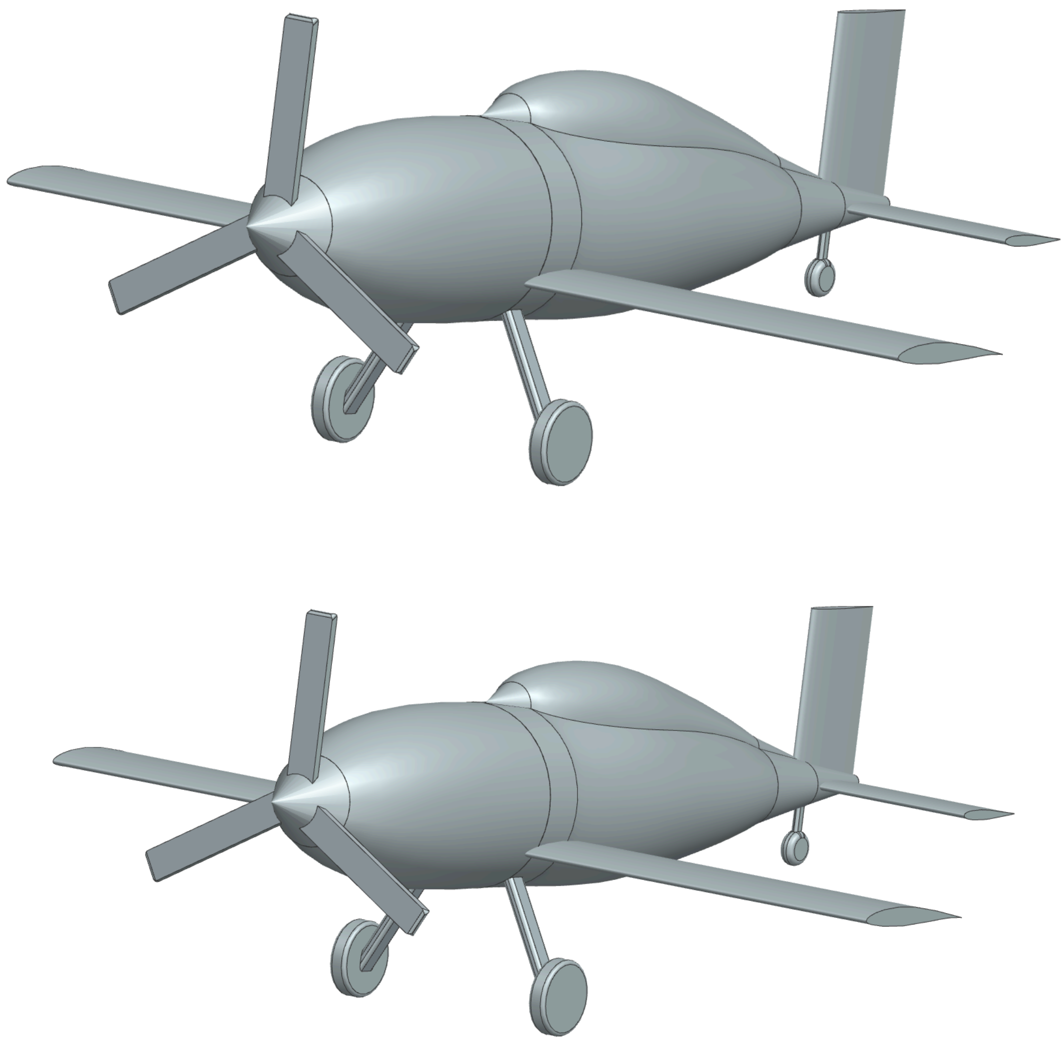

Figure 4.4: 3D View of the Aircraft Design

Parameter type	Parameter	Value	Rationale
Wing	Aspect ratio	6.125	Large aspect ratio selected for reduction of induced drag
	Wingspan	21.53 ft	
	Wing Area	75.67 ft^2	
	Winglets present (Y/N)	No	Reduce complexity, weight, and cost of development

Landing gear	Type	Fixed	Reduced weight allowed for better efficiency and more laps run before refueling
Weight	Empty weight	1174.52 lbs	
	MTOW	1541.47 lbs	
Airframe dimensions		21.51 ft long 8.58 ft tall 4 ft fuselage diameter	
Engine	Power	350 hp	Maximum allowed for competition
Speed	Max speed	267 mph	

Table 4.1: Aircraft Design Results from A205

Parameter type	Parameter	Value
Wing	Aspect ratio	6.125
	Wingspan	24.417 ft
	Wing Area	98 ft^2
	Winglets present (Y/N)	No
Landing gear	Type	Fixed
Weight	Empty weight	1170 lbs
	MTOW	1800 lbs
Airframe dimensions		20.75 ft long 9.16 ft tall
Engine	Power	340 hp
Speed	Max speed	265 mph

Table 4.2: Aircraft Design Characteristics from Edge 540

(flugzeuginfo.net, n.d.)

As can be seen in Tables 4.1 and 4.2, the aircraft that we have designed is similar in certain aspects to the Zivko Edge 540, the reference aircraft we based certain design decisions off of. We were able to create a slightly smaller and faster aircraft than the Edge 540, though, due to our design choices. This additional speed was the main goal of our design process, since this competition is about top speed rather than maneuverability like the Edge 540's intended purpose

4.3 CONCEPT REFINEMENT

For the wing shapes, we first outlined the needs of our aircraft to determine what kind of handling characteristics would be appropriate for the race. Our needs indicated that the plane required an airfoil with high wing loading enabling us to maximize cruise efficiency over long periods of time. However, this came at the cost of maneuverability during the climb and turn portions of the circuit. A high wing loading would mean that our aircraft would need a high coefficient of lift. Additionally, since the aircraft will primarily be operating in a small window of operating speed, our team decided that a 6 series airfoil would be appropriate as it combines a high lift coefficient with low drag over a small range of conditions, and is optimized for high speed.

Due to the design limitation of no more than 350 horsepower, we began to examine the feasibility of different engine types. We began the selection process by examining engine efficiency versus speed to determine which type of engine would be best suited for the speeds this aircraft will fly at. Turbojets and turbofans were quickly eliminated as propellers are better suited to speeds under Mach 0.5 due to their better efficiency. Furthermore, since piston engines are more fuel efficient than turboprops, and we do not need the extra power that a turboprop would be able to provide, we came to the conclusion that a piston propeller engine would suit this airplane.

For the landing gear decision, the fixed landing gear was deemed best for this mission. We decided the plane with the highest maximum range would perform the best in the race because it would have to refuel less during the race, so we used that as the basis of our calculations. Using the the coefficients of drag of both the fixed and retractable landing gear found in PM8, which were 0.031 and 0.028, respectively, we used the range equation see ection 4.4, and plugged our values in with the fixed landing gear having the regular plane mass at 1541.47 lb and the retractable landing gear being 100 kg heavier at 1761.932 lb.

The maximum range outcomes were R_{fixed} = 554.484 nautical miles for the fixed gear plane vs. $R_{not\ fixed}$= 439.86 nmi for the retractable landing gear plane. Because the fixed landing gear option had a higher value for maximum range, we went with fixed landing gear on our plane.

4.4 PERFORMANCE ANALYSIS

For the maximum range calculation, the formula is as follows:

$$R_{max} = \frac{\eta}{c_t} * \left(\frac{C_l}{C_d}\right)_{max} * ln\left(\frac{m_1}{m_2}\right) \qquad \text{(Equation 4.1)}$$

Where η is the propellor efficiency, C_t is the specific fuel consumption in 1/ft, m_1 is the weight of the plane at the beginning of the cruise (1472.8 lb), and m_2 is the plane weight at the end of the cruise (1343.8 lb), with only enough fuel to make the landing to refuel. To find the specific fuel consumption, we went to the datasheets (Lycoming.com) (Figure 4.5) of the Lycoming engine we chose and assumed that it would be acting at maximum power during the cruise portion of the race. We found the fuel consumption at maximum power and then converted that value into 1/s (Equation 4.2) to get specific fuel consumption.

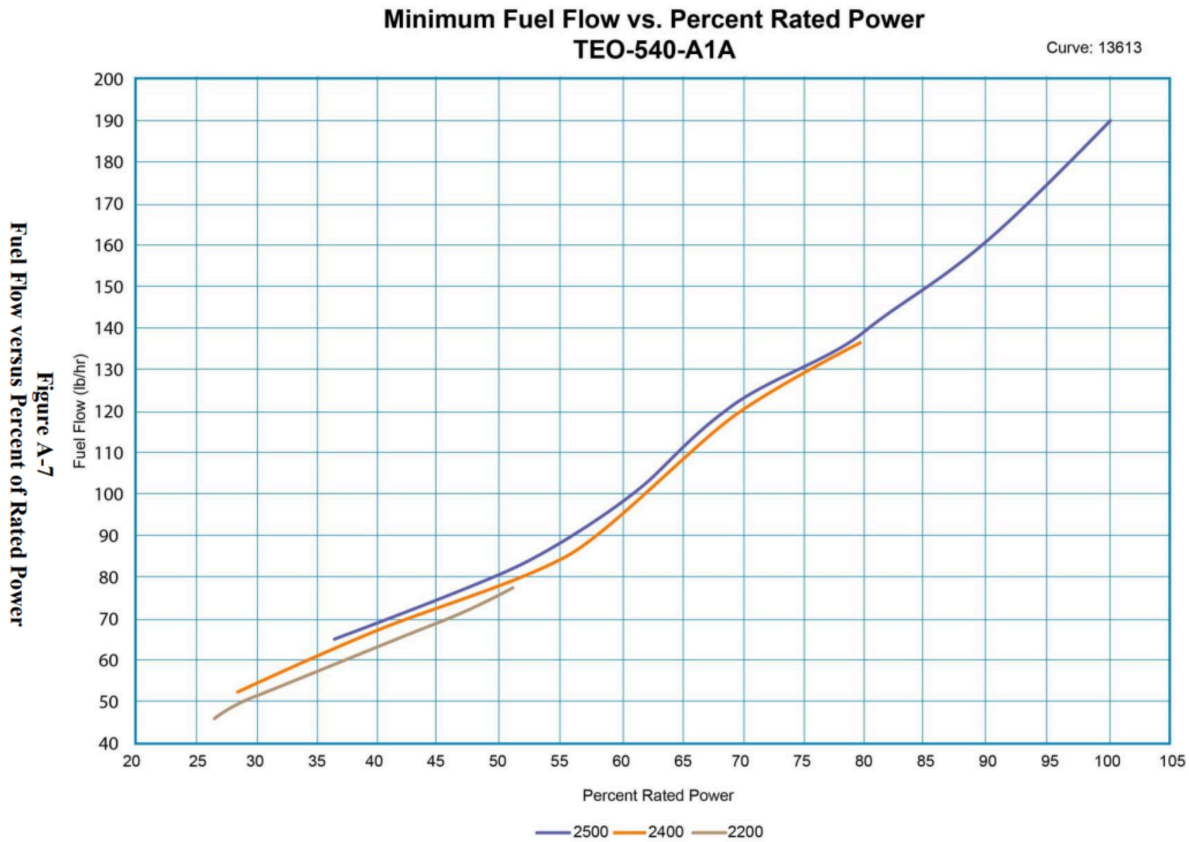

Figure 4.5: Lycoming Manual Fuel Flow Rate

$$c = 0.45 \frac{\text{lb}}{(\text{hp})(\text{h})} \frac{1\ \text{hp}}{550\ \text{ft} \cdot \text{lb/s}} \frac{1\ \text{h}}{3600\ \text{s}} = 2.27 \times 10^{-7}\ \text{ft}^{-1}$$

(Equation 4. 2) (Anderson)

As you can see, at maximum power, the fuel flow rate is 190 l,b/hr, and our maximum power is 350 hp. Using those values and the equation above, our final specific fuel consumption value was 2.741 * 10^-07.

To find Cl/Cd max, the (Equation 4.2) is as follows:

$$\left(\frac{C_l}{C_d}\right)_{max} = \frac{1}{2}\sqrt{\frac{1}{K^*C_{d,0}}} \qquad \text{(Equation 4.2)}$$

Our K value and Cd,0 have been calculated in the past PMs, being 0.0601 and 0.031 respectively. Plugging in all of our numbers and converting to nautical miles, the final maximum range was 554.484 nautical miles.

The maximum endurance calculation is as follows:

$$E_{max} = \frac{\eta}{c_t}\sqrt{\frac{2\rho S}{g}} * \left(\frac{C_l^{3/2}}{C_d}\right)_{max} * \left(\frac{1}{\sqrt{m_2}} - \frac{1}{\sqrt{m_1}}\right) \qquad \text{(Equation 4.3)}$$

Where

$$C_{l,\,E\,max} = \sqrt{\frac{3C_{d,0}}{K}} \text{ and } C_{d,\,E\,max} = 4C_{d,0} \qquad \text{(Equation 4.4 and 4.5)}$$

Using our values from before, we find that $\left(\frac{C_l^{3/2}}{C_d}\right)_{max}$ is 11.1945, and assume our air density is at the maximum elevation of our flight in the race (800 m), density is approximately 0.00232 lbf/ft^3, our wing area is 75.67 ft^2, m_1 is 1472.8 lb and m_2 is 1343.8 lb. This gives us a maximum endurance of 4420.231344 seconds or around 73.6705224 minutes.

Minimum thrust can be calculated with the (Equation 4.6)as follows:

$$T_{min} = 2mg\sqrt{K * C_{d,0}} \qquad \text{(Equation 4.6)}$$

From earlier solutions, we know that K = 0.0601, and Cd,0 = 0.031. We can assume the weight is at maximum or the takeoff weight because that will give us the most breathing room for the minimum thrust required in real life, so the weight is 1541.5 lbs. The air density and wing surface area is the same as in the previous solution. Solving the equation gives us a minimum thrust of 133.0257 lbs total.

Minimum Power can be calculated as follows:

$$P_{min} = \frac{4}{3}\left[\frac{2(mg)^3}{\rho S}\sqrt{3K^3C_{d,0}}\right]^{\frac{1}{2}} \quad \text{(Equation 4.7)}$$

We can use the same variables from the previous section as well as make the same assumptions to get a minimum power required at the maximum altitude of the race is 1870.5 lb ft/s or 34.0093 hp.

Of the engines that we considered for our plane, both the Lycoming iE2 350 HP TEO-540-A1A and Continental TSIO-550-E meet the minimum power requirement of 34.0093 hp because they both have up to 350 horsepower.

In order to calculate the speed at which our aircraft would be cruising during the race, while at full power, we created a graph showing the power available and power required at different airspeeds at an altitude near our highest altitude of flying during the race. The power required was calculated using the (Equation 4.8):

$$P_{available} = \eta\left(\frac{\rho_h}{\rho_0}\right)^{m_{ad}} \cdot P_{0,\,max} \quad \text{(Equation 4.8)}$$

The calculations for the power required at different velocities were based on the (Equation 4.9):

$$P_{required} = \frac{1}{2}\rho V^3 SC_{D_0} + \frac{2(mg)^2}{\rho VS}\left(\frac{1}{\pi e(AR)}\right) \quad \text{(Equation 4.9)}$$

The plot of these two different equations is visible in Figure 4.6, and the intersection between two lines is the maximum attainable speed in level flight. The results output by

MATLAB suggested that our intersection point at 1500 feet was 267 miles per hour as our maximum attainable speed in steady, level, unaccelerated flight.

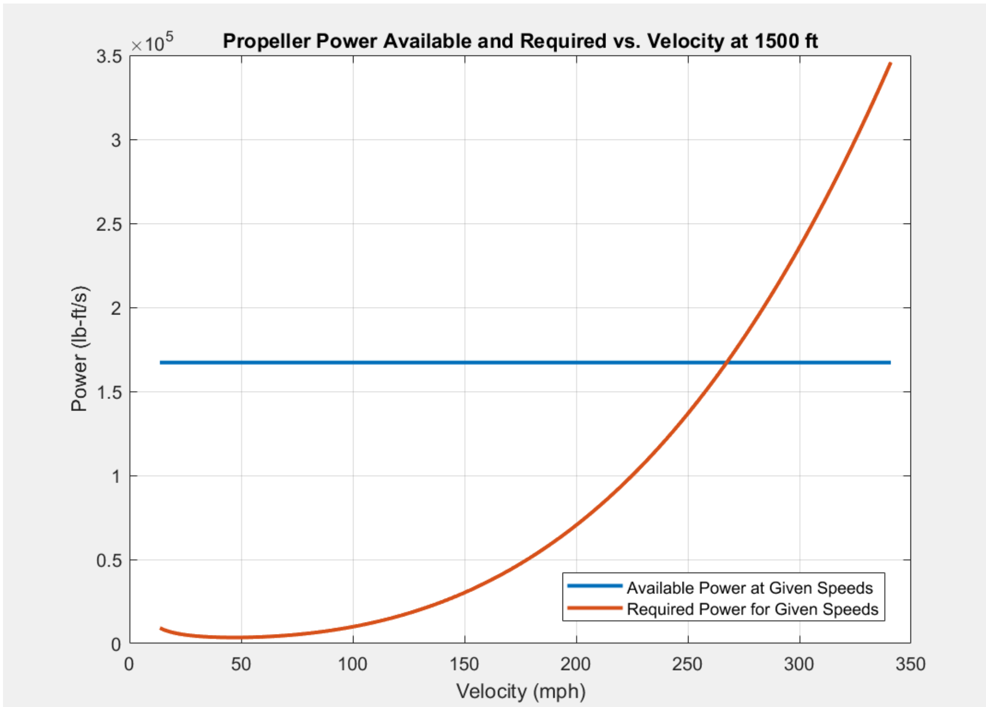

Figure 4.6: Propeller Power Available and Required vs. Velocity at 1500 ft

To find the service ceiling, the (Equation 4.10) was used.

$$\rho_{ceiling} = \left[\frac{4\rho_0^{\ m_{ad}}}{3P_{0,max}\eta} \left(\frac{2(mg)^3}{\rho} \sqrt{3K^3 C_{D,0}} \right)^{\frac{1}{2}} \right]^{\frac{2}{2m_{aad}+1}} \quad \text{(Equation 4.10)}$$

With values of 0.6 for mad (assumed) rho0 being 2.3769 * 10^-3, and P0, max being 350 hp or 192500 lbf ft/s, and keeping the rest of the values the same as prior to this equation, we found the air density at the flight ceiling to be 3.133 * 10^-4 lbf/ft^3, which corresponds to an altitude of around 53,000 ft.

To find the minimum velocity of the plane, the following (Equation 4.11) was used:

$$V_{max} = \left(\frac{2m_{low}g}{\rho_{ceiling} * S} * \sqrt{\frac{K}{3C_{d,0}}} \right)^{1/2} \qquad \text{(Equation 4.11)}$$

With the flight ceiling air density found before to be 3.133 * 10^-4 lbf/ft^3 and weight of the plane being the lightest weight possible which was 1472.8 lb at the end of the cruise portion of the flight, we found the maximum velocity to be 259.3628 ft/s or 176.838 mph.

To find the minimum velocity, the following (Equation 4.12) can be used:

$$V_{min} = \left(\frac{2m_{high}g}{\rho_{low\ altitude} * S} * \sqrt{\frac{K}{3C_{d,0}}} \right)^{1/2} \qquad \text{(Equation 4.12)}$$

Using the air density at the low altitude cruise portion as the density is higher, and using the mass at the beginning of the cruise portion of the race as it has the most fuel at that point, the minimum velocity of the plane is 101.0174 ft/s or 68.876 mph

To find the maximum flight path angle, we can use the following (Equation 4.13):

$$\gamma_{max} = \frac{\left(\frac{\rho_h}{\rho_0}\right)^{m_{ad}} \eta P_{0,max} - \frac{1}{2}\rho_h S C_{D,0} V_{max}^{\ 2} - \frac{2K(m_{low\ fuel}g)^2}{\rho_h S V_{max}^{\ 2}}}{m_{low\ fuel} g V_{max}} \qquad \text{(Equation 4.13)}$$

In this equation we assumed that the plane was flying at maximum velocity, and had low mass but enough mass to make the climb. Typically the plane would climb at a higher mass but this is the maximum flight path angle so a lower mass is preferred. For the mass we assumed it to be 1400 lbs, and our Vmax was 259.36 ft/s from before. This gave us a maximum flight path angle of 0.4407 rad or 25.25 degrees.

For the rate of climb, the (Equation 4.14) is as follows.

$$RoC_{max} = V_{max} sin(\gamma_{max}) \quad \text{(Equation 4.14)}$$

The maximum rate of climb found was 110.6426 ft/s, given the maximum velocity being 259.36 ft/s and the maximum flight path angle being 25.25 degrees.

Once we determined the optimal engine type, we began to examine different piston engines to determine which ones would suit our needs the best. We were able to narrow the decision down into two options, both of which are able to produce 350 hp. The Lycoming iE2 TEO-540-A1A and the Continental TSIO-550-E. To determine which engine was the best fit, we used a decision matrix to compare the specifications and performance of the two engines. The design requirements included fuel efficiency, size, and reliability. Fuel efficiency and size were the top priorities as they would determine our range and top speed. Maintenance was also a factor as the plane would be expected to fly at top speed for long periods of time without breaking. From considering these factors, we calculated that the optimal engine would be the Lycoming iE2 TEO-540-A1A due to its superior fuel efficiency and longer intervals between engine overhauls.

Weight/Importance:
15: Must have
10: Good to have
5: Nice to have

Benchmarking:
5: Best, 1: Worst

DESIGN REQUIREMENTS (Criteria & Constraints)	Weight/Importance	METRICS: Dry Weight	Fuel Consumption	Maintinance Intervals	Volume	BENCHMARKING (Unweighted): iE2 350 HP TEO-540-A1A	Continental TSIO-550-E		BENCHMARKING (Weighted): iE2 350 HP TEO-540-A1A	Continental TSIO-550-E	
1 Fuel Efficiency	15		X		x	4	3		60	45	
2 Relaiblilty	5			x		4	3		20	15	
3 Size	10	x			x	1.6	3		16	30	
ENGINEERING TARGETS -->											
	Units	lbs	lb/hr	hr	in^3						
WEIGHTED TOTAL -->									96	90	

DEVELOPMENT OF METRIC SCORING (5=Best, 1=Worst)

Metrics

Dry Weight	DATA (lbs)	Score	Description of Score/Meaning
	x>500	1	A score of 1 is the worst score with the clarity of instruction being really low.
	450<x<500	2	A score of 2 above average weight
	450>x>400 lbs	3	A score of 3 average weight
	400>x>350 lbs	4	A score of 4 is good with a below average weight
	x<350 lbs	5	A score of 5 means a very low weight

Fuel Consumption	DATA (lb/hr)	Score	Description of Score/Meaning
	x>300	1	A score of 1 is the worst score with a high fuel consumption
	250<x<300	2	A score of 2 is a little better with an above average fuel consumption
	200<x<250	3	A score of 3 is the average fuel consumption
	150<x<200	4	A score of 4 is below average fuel consumption
	x<150	5	A score of 5 is low fuel consumption to produce 350 hp

Maintinance Intervals	DATA(hr)	Score	Description of Score/Meaning
	1400	1	A score of 1 is undesirable which means that the device does not fit in with the Purdue Aesthetics.
	1600	2	A score of 2 is better and means that the device somewhat fits in with the Purdue Aesthetics.
	1800	3	A score of 3 means that the device fits in with the Purdue Aesthetics.
	2000	4	A score of 4 means that the device fits in well with the Purdue Aesthetics.
	2200	5	A score of 5 is the best and means that the device not only fits in but compliments Purdue Aestheti

Volume	DATA (in^3)	Score	
	x>65000	1	A score of 1 is a large engine that would be bulky
	60000<x<6500	2	A score of 2 is an engine of abover average size for its type
	55000<x<6000	3	A score of 3 is an average engine size
	50000<x<5500	4	A score of 4
	x<50000	5	A score of 5

CRITERIA ASSOCIATED WITH MULTIPLE METRICS

Criterion (related to multiple metrics)

Size	350 HP TEO-540-A1	Continental TSIO-550-E		Weighting Formula: 70% Weight, 30% Volume
Weight	1	3		Justification: If engine is too heavy or bulky, it could lead to decreased efficiency in flight or a higher drag resulting lower top speed.
Volume	3	3		
Weighted Score:	1.6	3		

Fuel Efficiency	350 HP TEO-540-A1	Continental TSIO-550-E		Weighting Formula: 100% Fuel Consumption
Fuel Consumption	4	3		Justification: An engine with high fuel consumption will result in a shorter range, therefore it is important to consider this when determining efficiency
Weighted Score:	4	3	0	

Reliability	350 HP TEO-540-A1	Continental TSIO-550-E		Weighting Formula: 100% Engine Overhaul Interval
Engine Overhaul Interval	4	3		Justification: A shorter engine overhaul interval meanst that the engine may be less durable or reliable meaning that it is more likely to fail under sustained high loads
Weighted Score:	4	3		

Figure 4.7: Weighted Design Matrix

Based on all of these calculations, we are then able to calculate an overall race time for the aircraft. We estimate that an initial takeoff and climb to 50 meters of altitude will take 1 minute, assuming the aircraft is lined up on the runway waiting to take off when time is started. In cruise scenarios, we expect our aircraft to be able to fly at 267 miles per hour, which means that it will take 25.6 minutes to cover the 49.5 nautical miles of each level cruising leg. The climb segment requires the aircraft to achieve a flight path of 21.78 degrees, which when entered into Equation 4.13, and then solved for V gives a speed of 148 knots airspeed and 137.4 mph horizontally. This means that the aircraft can complete the climb without issue in 0.50 minutes. During the descent, we assume the aircraft is able to reach the maximum allowable airspeed during the competition, Mach 0.4. At this speed, the time to complete the descent is 0.24 minutes.

Since we expect to be able to complete 2 laps before needing to refuel 32.16 gallons of fuel for most of the race, we calculated the stopped time to be 14.73 minutes for stops of this type based on Equation 4.15. Before the 25th lap, the aircraft will need to stop and refuel again, but since there is only one lap remaining in the race, only 16.08 gallons of fuel will need to be added. Using Equation 4.15, the stop time calculated for this stop is 12.36 minutes.

$$Refuel\,Time \;=\; [10 \,+\, 5(\frac{V}{34})] \qquad \text{(Equation 4.15)}$$

Based on these numbers, we have calculated the total race time of our aircraft to be 24 hours and 46 minutes, with 13 refueling stops.

5 CONCLUSIONS

Within this section, you can expect to read about our evaluation of the project as a whole and how many challenges within the project led to great lessons learned. You will also find a discussion about what we would do next as a team if we were to continue our work into another semester.

5.1 DESIGN EVALUATION

We believe that our design will function as intended. As we progressed through the project, we found that many of our predicted characteristics from our calculations were similar to those of real aircraft with similar roles. For example, the takeoff and landing distance calculations were very similar to the Zivko Edge 540 with our calculated landing distance being 75m and the Zivko Edge was 90 (Aero Corner). However there are some flaws with this design that could lead to issues in the race. The primary drawback of our design is a slow rate of climb. This will hamper the performance of the aircraft for the section where it has to pull a rapid climb. This could lead to lost time and result in a longer time to complete the race.

If we had to do this project again, we would take more time to learn about tail design to come up with a more optimal design as it currently uses many of the same parameters as the main wing. Additionally, we would also like to learn more about wing optimization beyond choosing an airfoil and learn more about how the position of the maximum thickness and camber of the airfoil affect the performance of the aircraft. This would allow us to better optimize our design and potentially offset some of the drawbacks of our wing shape and size.

Despite the risks still present, our aircraft was able to meet most of the requirements set by our stakeholders. We are able to measure the pilot's weight to ensure that they comply with our specification of being under 150 lbs and 6ft tall. After calculating our top speed, it is in fact below the race organization's requirement of Mach 0.4 as it is in fact Mach 0.35. Additionally, the engine we selected meets the maximum allowed power output of 350 horsepower. We are unable to currently test if several of these airplanes will be able to lap within 5-6 seconds of each other without actual flight testing. We were able to meet the requirement of keeping the engine running within specifications from the manufacturer as the engine's ECU is capable of measuring and displaying all of the relevant information for the continued running of the aircraft (Lycoming TEO-540-A1A Operations Manual). This will enable ground crews to notify the pilot of issues related to the engine and allow the pilot to safely perform an emergency landing. Additionally, G-force sensors will be able to inform the pilot and race officials if the plane and pilot are being overstressed and can take action to avoid the aircraft breaking or the pilot overexerting themselves. We were not able to meet the goal of flying within 0.05 of the maximum possible speed of Mach 0.4. For the Race team, we were able to meet the requirements of having a fuel tank capable of holding 34 gallons and a verified altimeter capable of reading the altitude correctly and the pilot will be able to interpret the data. For the part suppliers, we chose components such as the engine that are available to purchase off the shelf ready for installation. We used a standard airfoil design that is commonly used enabling easier manufacturing. We are able to verify the production of CO2 by the engine through sensors connected to the ECU that can measure the amount of CO2.

5.2 NEXT STEPS

If our team were to continue the project into another semester there would be a considerable amount of iterations to maximize certain efficiencies such as the takeoff weight and total drag during cruise. Given more time, our team would take a careful look at design decisions and see if any options seem like they could be improved. We would put more time into finding how to improve overall race time because as a team we felt as if we were only trying to design a plane with the minimum amount of efficiency to make it through the race. Given that we would continue our work, I am positive that our aircraft would be modified to a time more pleasing to the race team and stakeholders. Overall we would take our time perfecting each important design decision whether that be the airfoil or even going back to the basics and finding a better reference plane.

5.3 LESSONS LEARNED

Throughout the semester, we believe our ability to work as a team has improved immensely and our understanding of the engineering design process has developed more and more every day. We were tasked with designing an aircraft with varying assumptions and many underlying constraints. From initializing communication between group members to finalizing this report we faced many challenges.

One of the most important lessons we learned from these many challenges was being able to track certain decisions each member was making toward the design of the aircraft. Whether it was deciding to use a certain aerodynamic characteristic from one of the reference aircraft or using a certain equation from a textbook, the ability to keep all decisions clear between teammates is something our team learned during and after this long process. Something we would definitely change about our initial approach to the problem at hand. The lesson we learned from this challenge could be carried on to any project we are assigned in future years whether it be here at Purdue University or in industry

Another lesson learned from the many challenges we faced was time management. Now this may be a given with the magnitude of the project but the team feels as if this skill is always tested when assigned a major project or assignment. It is imperative to understand everyone's time constraints on the team early in the process. Time and time again we found ourselves in a predicament where we would have to work quickly to accomplish milestones. This would cause a ripple effect in the previous lesson described above. We found the way to improve this struggle was to divide tasks evenly and ask questions. Asking, what seems like a stupid question could end up saving thirty minutes to an hour of work. From experience, our team found going to office hours or asking the teaching team questions saved us from many hours of repeated failure.

In conclusion, when going through a process like this, you have to expect difficulties. It is never going to come easy and there will always be a new problem to face. Our team agreed that from little to no experience with aircraft design we accomplished a lot. From this came the knowledge to understand what worked and what did not work. We are only starting our journeys in the world of aerospace engineering and this project gave great insight into what all goes into a design project.

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APPENDIX: MATLAB CODE

Aircraft Sizing Code: Corresponding to Section 3.2

```
% AAE 251 Spring 2024
% PM 4
% Project Plane Weight Calculation
% Authors: Henry Hellmann, Dominic Mastromatteo
fprintf(' This code assumes a cruise level \nof 50 and 790 m and L/D ratio of 11 during flight. It also assumes \na drop time of 13.7 seconds , a low altitude cruise Vel. at 217 mph, \ncruise range of 100 nmi between laps. It also assumes 178 lb pilot. \nThe takeoff weight will be from takeoff to refueling, and will complete 25 refuels.\n\n')
%% Initialization
difference = 10;
% initial difference for final loop
z = 0;
% initialized value for while loop
unit = "English";
numLaps = 1;
range = (numLaps * 200 * 1.15078 * 5280);
mi100 = (100 * 1.15078 * 5280);
% 100 nm to feet, since each lap, 100 nm are flown at each altitude
Mach = 0.4;
V_LA = Mach * 633.8507 * 1.46667;
% Velocity at low altitude, given speed of sound at 100 m
V_HA = Mach * 628.0410 * 1.46667;
% Velocity at high altitude, given speed of sound at 800 m
Fuel_density = 720.1568;
% fuel density in kg/cu meters
Fuel_tank = 0.128704;
% fuel tank volume cubic meters
Fuel_weight = Fuel_density * Fuel_tank;
% MAX Total fuel weight kg
C_cruise = (0.4 * V_HA / (550 * 0.8)) / 3600;
% specific fuel to weight ratio for cruise
crew_weight = 81;
% payload and crew weight
Payload = crew_weight;
L_D_flight = 13.5;
% L/D ratio for cruise portion of flight
A = 0.81;
% English A constant
C = -0.01;
% C constant
K_vx = 1;
% Variable sweep constant
```

```
%% Calculations
W1_W0 = 0.97;
% Warmup/takeoff weight ratio
W2_W1 = 0.985;
% Climb weight ratio
W3_W2_LA = exp((-mi100 * C_cruise) / (V_LA * L_D_flight));
% Cruise weight ratio, Low Altitude, for one lap only
W3_W2_HA = exp((-mi100 * C_cruise) / (V_HA * L_D_flight));
% Cruise weight ratio, High Altitude, for one lap only
W5_W4 = 0.995;
% Land weight ratio
lap = W3_W2_LA * W2_W1 * W3_W2_HA;
refuel = W5_W4;
W7_W0 = W1_W0 * W2_W1 * (lap)^numLaps * (refuel);                    % total weight ratio
Wf_W0 = 1.06*(1 - W7_W0);
% fuel weight ratio
W0_guess = 900;
% initial weight guess
while abs(difference) > 1
% iterating while loop to find takeoff weight
    z = z+1;
    We_W0 = A * (W0_guess ^ C) * K_vx;
    W0_calculated = Payload / (1 - Wf_W0 - We_W0);
    difference = W0_calculated - W0_guess;
    fprintf("Calculated weight: %f kg (iteration %d)\n", W0_calculated, z)
    % printed weight calculation for each iteration, kg
    fprintf(" - Fuel mass consumed (with max available = %f): %f\n", Fuel_weight, Wf_W0 * W0_calculated)
    W0_guess = W0_calculated;
end
fprintf("Final weight: %f Pounds\n", W0_calculated * 2.204)
% Final calculated weight, pounds
```

CODE: SECTION 3.6

```
% AAE 251 Spring 2023
% PM6
% Takeoff and Landing Distance
% Authors: Robert Hays
%
%% Initialization
W_gross_1 = 1191.73; %lbs
W_gross_2 = 992.352;
density = 2.3769*10^-3; %slug/ft^3
sigma = 1; %since launching a sea level density airport == density sea level
Cl_max = 1.4; %approx Cl max from graph
```

```
theta = 3; %landing approach angle in degrees
S = 133.14957; %ft^2 wing area
power = 350; %horse power
%% Calculations
V_To_1 = 1.2 * (2*W_gross_1 / (density*Cl_max*S))^0.5;
Thrust_1 = power * 550 / V_To_1;
S_To_1 = 20.9*(W_gross_1 / S)/(sigma * Cl_max * (Thrust_1 / W_gross_1)) + 69*((W_gross_1/S) / (sigma * Cl_max))^0.5
S_L = (79.4*((W_gross_2-193.59)/S) / (sigma*Cl_max)) + 50/tand(theta)
```

3.7 Drag Polar MATLAB Code

Code:
```
% AAE 251 Spring 2024
% PM7 Code
% This code graphs Cd vs Cl and Cd vs V
% Authors: Henry Hellmann
%% Initialization
data = readmatrix("PM7 data.xlsx");  % data from airfoiltools.com based on Naca 63-412 airfoil
alpha = data(:,1);                % alpha values from data
CL = data(:,2);                   % Cl values from data
Cl_max = 2.41;                    % Cl max from PM5
W_Gross = 1382;                   % weight in pounds
p = 0.0765;                       % density at sea level in lb/ft^3
S = 75.23;                        % surface area feet squared
Cd0 = 0.0945;                     % Calculated Cd0 from PM7
AR = 6.125;                       % aspect ratio from PM5
%% Calculations
% Cd calculation from Cl
e = 1.78 * ( 1 - 0.045 * AR ^ 0.68) - 0.64;
K = 1/(pi * AR * e);
Cd = Cd0 + K*CL.^2;
% Cd calculations from different velocities
V_stall = (2*W_Gross / (p*Cl_max*S))^0.5;
V = V_stall:1:368.133;                 % velocity in ft/s from v stall to max cruise speed in ft/s
Cl1 = (W_Gross .* 2) ./ (p .* V.^2 .* S);
Cd1 = Cd0 + K*(Cl1.^2);
%% Plots
% Cd vs Cl plot
figure
plot(CL,Cd)
xlabel('Lift coefficient ')
ylabel('Drag Coefficient')
grid on
title('Drag Coefficient vs. Lift Coefficient')
% Cd vs. Velocity plot
figure
plot(V,Cd1)
```

```
xlabel('Velocity range (ft/s)')
ylabel('Drag Coefficient')
grid on
title('Drag Coefficient vs Velocity')
```

3.7 Drag Coefficient vs. Velocity Code

```
% AAE 251 Spring 2024
% PM7 Code
% This code graphs Cd vs Cl and Cd vs V
% Authors: Henry Hellmann
%% Initialization
data = readmatrix("PM7 data.xlsx");  % data from airfoiltools.com based on Naca 63-412 airfoil
alpha = data(:,1);                % alpha values from data
CL = data(:,2);                   % Cl values from data
Cl_max = 2.41;                    % Cl max from PM5
W_Gross = 1382;                   % weight in pounds
p = 0.0765;                       % density at sea level in lb/ft^3
S = 75.23;                        % surface area feet squared
Cd0 = 0.0945;                     % Calculated Cd0 from PM7
AR = 6.125;                       % aspect ratio from PM5
%% Calculations
% Cd calculation from Cl
e = 1.78 * ( 1 - 0.045 * AR ^ 0.68) - 0.64;
K = 1/(pi * AR * e);
Cd = Cd0 + K*CL.^2;
% Cd calculations from different velocities
V_stall = (2*W_Gross / (p*Cl_max*S))^0.5;
V = V_stall:1:368.133;                  % velocity in ft/s from v stall to max cruise speed in ft/s
Cl1 = (W_Gross .* 2) ./ (p .* V.^2 .* S);
Cd1 = Cd0 + K*(Cl1.^2);
%% Plots
% Cd vs Cl plot
figure
plot(CL,Cd)
xlabel('Lift coefficient ')
ylabel('Drag Coefficient')
grid on
title('Drag Coefficient vs. Lift Coefficient')
% Cd vs. Velocity plot
figure
plot(V,Cd1)
xlabel('Velocity range (ft/s)')
ylabel('Drag Coefficient')
grid on
title('Drag Coefficient vs Velocity')
```

4.4 Propeller Power Available and Required vs. Velocity MATLAB Code

```
% AAE 251 Spring 2024
% Thrust and Power Curves
% Authors: Dominic Mastromatteo
%% Initialization
clear all
clc
% aircraft properties
propMass = 1541/32.17;        % slugs - aircraft with full fuel tank
propS = 75.23;          % ft^2 - wing surface area
propSpeedRange = 20:500;   % f/s - free stream velocity
propCD0 = 0.031;
propK = 1/(pi * 0.7 * 6.125);          % 1/(pi*AR*e)
propP0Max = 350*550;      % ft-lb/s
propMAD = 0.6;
propPropEfficiency = 0.87;
% other constants
rho0 = 2.3769*10^-3;
rhoH = 2.2743*10^-3;          % slug/ft^3 - air density at required altitude
%% Calculations
% get available power values from the function
propPowAvail = powerAvailable(rhoH, propMAD, propP0Max, propPropEfficiency, propSpeedRange, rho0);
% get required power values to maintain SLUF from the function
propPowReq = propPowerRequired(propMass, rhoH, propS, propK, propCD0, propSpeedRange);
%% Formatted Text and Figure Displays
% create the plot of the power required at an altitude
p1 = plot(propSpeedRange, propPowAvail, propSpeedRange, propPowReq);
p1(1).LineWidth = 2;
p1(2).LineWidth = 2;
title("Propeller Power Available and Required vs. Velocity at 1500 ft")
xlabel("Velocity (ft/s)")
ylabel("Power (lb-ft/s)")
legend("Available Power at Given Speeds", "Required Power for Given Speeds")
grid on
figure()
p1 = plot(propSpeedRange*0.6818, propPowAvail, propSpeedRange*0.6818, propPowReq);
p1(1).LineWidth = 2;
p1(2).LineWidth = 2;
title("Propeller Power Available and Required vs. Velocity at 1500 ft")
xlabel("Velocity (mph)")
ylabel("Power (lb-ft/s)")
legend("Available Power at Given Speeds", "Required Power for Given Speeds")
grid on
```

```
% AAE 251 Spring 2024
% Power Available Calculations for Prop Aircraft
% Authors: Dominic Mastromatteo
function [P] = powerAvailable(rhoH, mAD, P0max, propEfficiency, velo, rho0)
P = [];
for i = 1:length(velo)
  P = [P, (rhoH / rho0)^mAD * P0max * propEfficiency];
end
```

```
% AAE 251 Spring 2024
% Power Required for Propeller Aircraft for different Velocities
% Authors: Dominic Mastromatteo
function [P] = propPowerRequired(m, rho, S, K, Cd0, velo)
g = 9.81;                    % m/s^2
P = [];
Pmin = 4/3 * ((2 * (m * g)^3)/(rho * S) * sqrt(3 * K^3 * Cd0))^1/2;
for i = 1:length(velo)
  term1 = (1/2) * rho * S * velo(i)^3 * Cd0;
  term2 = 2 * K * ((m * g)^2 / (rho * S * velo(i)));
  P = [P, term1 + term2];
end
```

Code, section 4.4

```
% AAE 251 Spring 2024
% Final Report
% Performance Specs Calculations
% Authors: Henry Hellmann
%% Initialzation
clear
eta = 0.87;          % propeller efficiency
rho_max = 2.2079E-3;    % Max altitude air density lb/ft^3
g = 32.2;            % imperial gravity
W_max = 1761.932;      % maximum weight (gross takeoff weight at 2 laps per refuel)
AR = 6.125;           % Aspect Ratio
S = 75.67;            % wing area (ft sq)
Cd0 = 0.031;          % Zero lift drag coeff
mad = 0.6;            % mad
rho0 = 2.3769E-3;      % Sea level air density lb/ft^3
fuel_weight = 204.34;   % fuel weight lbs
%% Calculations
% K calculation
e = 1.78 * (1 - 0.045 * AR ^ 0.68) - 0.64;
K = (pi*e*AR)^-1;
```

```
P0max = 350 * 550; % Max power
% mass before and after cruise
m1 = .97*.985* W_max;
m2 = (W_max - fuel_weight) / .995;
% SFC calc
ct = (1.90/(350 * 550 * 3600)) * 100;
% Max range calc
clcd_max = 0.5 * (sqrt((K*Cd0)^-1));
R_max = (eta/ct)*(clcd_max)*log(m1/m2);
Cl = sqrt(3*Cd0 / K);
% Max endurance Calc
Cd = 4*Cd0;
cl32cd = Cl^1.5 / Cd;
E_max = (eta/ct) * sqrt(rho_max * 2 * S / g) * (m2^(-.5) - m1^(-.5)) * cl32cd;
% Min thrust and power required
T_min = 2*W_max*sqrt(K*Cd0);
P_min = (4/3)*((2*W_max^3/(rho_max*S))*sqrt(3*K^3*Cd0))^.5;
% Flight ceiling
rho_ceiling = (((4*rho0^mad)/(3*P0max*eta))*((2*W_max^3/S)*sqrt(3*K^3*Cd0))^.5)^(2/(2*mad + 1));
% V max, min, ft/s
V_max = ((2*992.354/(rho_ceiling*S))*sqrt(K/(3*Cd0)))^.5;
V_min = ((2*1138.64/(2.37E-3*S))*sqrt(K/(3*Cd0)))^.5;
% Max flight path angle and Rate of climb
gamma_max = (rho_max / rho0)^mad * eta*P0max - .5 * rho_max * S * Cd0 * V_max^2 - (2*K*1400^2 / (rho_max * S * V_max^2));
gamma_max = gamma_max / (V_max*1400);
Roc = V_max*sin(gamma_max);
```