IOWA STATE UNIVERSITY

AIRCRAFT SIZING AND INITIAL CONCEPTS

STRATOSHIELD

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Glossary

$C_{D_{\mathbf{max}}}$	Maximum value of the 2D drag coefficient. *(p. 6)*
$C_{L_{\mathbf{max}}}$	Maximum value of the 2D lift coefficient. *(p. 6)*
L	Characteristic length used in calculating Reynolds number. *(p. 5)*
M	Mach number. *(p. 8)*
Re	Reynolds number. *(p. 5, 8)*
T	Temperature. *(p. 5)*
$\alpha_{\mathbf{stall}}$	Angle of attack at which stall occurs. *(p. 6)*
$\frac{dC_L}{d\alpha}$	Slope of the lift coefficient vs the angle of attack. *(p. 6)*
$\left(\frac{L}{D}\right)_{\mathbf{max}}$	Maximum value of the lift-drag ratio. *(p. 6)*
μ	Fluid dynamic viscosity. *(p. 5)*
ρ	Fluid density. *(p. 5)*
b	Wingspan. *(p. 2, 11)*
b_t	Tail wingspan. *(p. 11)*
c	Chord length. *(p. 2, 11)*
c_t	Tail chord length. *(p. 11)*
c_v	Vertical stabilizer chord length. *(p. 11)*
h	Elevation or altitude. *(p. 5)*
h_v	Height of the vertical stabilizer. *(p. 11)*
l_t	Tail moment arm, *i.e.*, the distance from the leading edge of the wing to the tail. *(p. 11)*
v	Fluid velocity. *(p. 5)*

Acronyms

AGL	above ground level. *(p. 4)*
AOI	area of interest. *(p. 1)*
AR	aspect ratio. *(p. 2, 11)*
CAD	computer-aided design. *(p. 1, 18)*
CG	center of gravity. *(p. 13, 14)*
COTS	commercial off-the-shelf. *(p. 4, 15)*
DHS	Department of Homeland Security. *(p. 1)*
E	Eppler. *(p. iii, 3, 4, 6, 7, 9, 10)*
MATLAB	MATrix LABoratory. *(p. 5, 14)*
MH	Martin Hepperle. *(p. 3, 6)*
MSL	mean sea level. *(p. 5)*
NACA	National Advisory Committee for Aeronautics. *(p. iii, 3–6, 8–10, 18)*
RC	remote-controlled. *(p. 8, 15)*
SLF	steady level flight. *(p. 2)*
sUAS	small unmanned aerial systems. *(p. 1, 3, 4, 13)*
UAV	unmanned aerial vehicle. *(p. 1, 3, 8, 10, 15, 18)*

1 Introduction

The Department of Homeland Security (DHS) has contracted StratoShield to build an unmanned aerial vehicle (UAV) capable of identifying and neutralizing hostile small unmanned aerial systems (sUAS). To satisfy this mission, StratoShield is designing an aircraft equipped with technology to jam sUAS control signals. It will be capable of patrolling an area of interest (AOI) for up to 45 min while protecting a designated key area within the AOI. The primary objective is to identify, track, and disable sUAS threats through non-destructive electronic countermeasures.

This report includes initial estimates of the weight and size of StratoShield's UAV, code-named "Banshee". Additionally, several concept sketches and designs are discussed in Chapter 5. These measurements and decisions are not final; rather, they are intended to provide the computer-aided design (CAD) engineers and design team with a more reasonable starting point. Additionally, starting with a more refined design also aims to reduce the number of design cycles required to meet the system requirements.

2

Airfoil

2.1 Basic Airfoil Dimensions

The primary goal at this point in the design process is to maximize time aloft, also known as endurance. To help our aircraft "glide" more easily and reduce the thrust required to maintain steady level flight (SLF), we chose to start with a wingspan of 216 cm (85 in), just under the maximum wingspan dictated by the course requirements.

With the wingspan selected, we determined our chord length by assuming an aspect ratio (AR) of approximately 10. Then, by using the equation for AR for a rectangular wing, Equation 2.1,

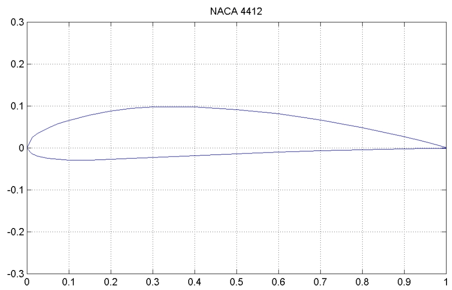
$$AR = \frac{b}{c} \tag{2.1}$$

where AR is the aspect ratio, b is the wingspan, and c is the wing chord length, we back calculated the chord length to be 20 cm (8 in).

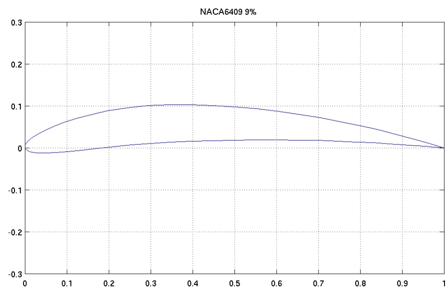
2.2 Airfoil Selection

2.2.1 Main Wing

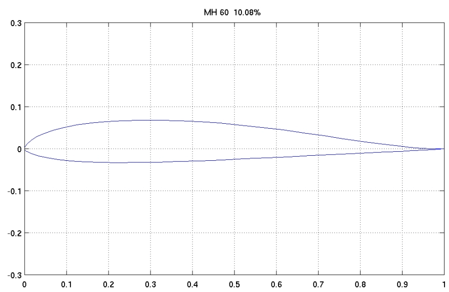
Based on our precursory research, we decided to investigate the suitability of three airfoil shapes, two National Advisory Committee for Aeronautics (NACA) airfoils and one Martin Hepperle (MH) airfoil, shown in Figure 2.1.



(a) *Plot of the NACA 4412 airfoil.*



(b) *Plot of the NACA 6409 airfoil.*



(c) *Plot of the MH 60 airfoil.*

Figure 2.1: *Plot of each of the airfoils we analyzed for the main wing. Graphs are courtesy of University of Illinois at Urbana-Champaign (2024).*

These airfoils were selected for further analysis because of their historical ability to provide high lift at low speeds. They were also selected for their manufacturing ease, attributed to their simpler shape, in contrast to other high-lift airfoils, which are often very thin and have steep positive camber.

2.2.2 Horizontal Stabilizer

For the horizontal stabilizer, we chose to analyze an Eppler (E) airfoil, specifically the E423 (see Figure 2.2). The E423 was specifically designed for low-speed aircraft, offering stability and control at slow loitering speeds—crucial for an sUAS. It is commonly used in gliders and UAV that require long endurance at low speeds.

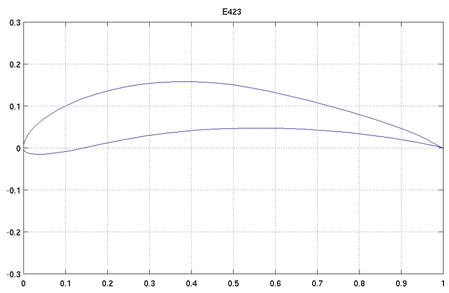
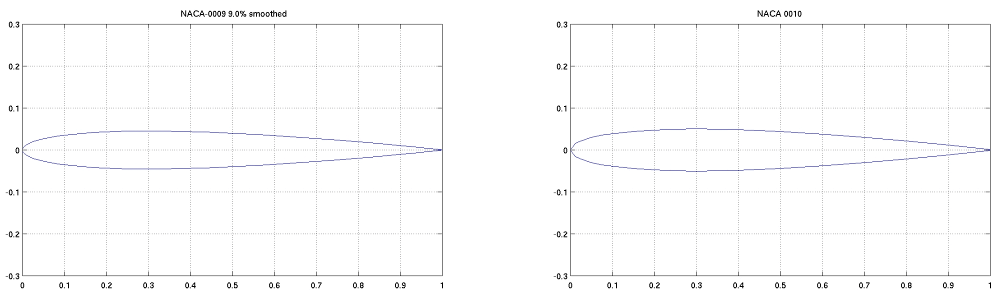


Figure 2.2: *A plot of the E423 airfoil (University of Illinois at Urbana-Champaign, 2024).*

2.2.3 Vertical Stabilizer

Vertical stabilizers are often symmetric airfoils since they have a neutral pitching moment and do not need to generate much lift to create a yawing moment. We chose to analyze the NACA0009 and the NACA0010, shown in Figure 2.3.



(a) *Plot of the NACA 0009 airfoil.* **(b)** *Plot of the NACA 0010 airfoil.*

Figure 2.3: *Plot of each of the airfoils we analyzed for the vertical stabilizer. Graphs are courtesy of University of Illinois at Urbana-Champaign (2024).*

2.3 Reynolds Number

To analyze the performance of these airfoils, we had to determine an initial Reynolds number. Since Reynolds number is dependent on atmospheric conditions, we first defined our expected operating conditions. The Banshee needs to loiter above the maximum altitude of commercial off-the-shelf (COTS) sUAS, which is typically 120 m (Khan, 2024). For the purpose of analysis, we chose a cruise altitude of 150 m above ground level (AGL). Given the average of elevation of Ames, IA, United States of America, we determined the cruise elevation to be:

$$h = 307\,\text{m} + 150\,\text{m}$$
$$h = 457\,\text{m}$$

where h is the elevation above mean sea level (MSL). Using a table of standard atmospheric conditions, we generated the cruise conditions tabulated in Table 2.1.

Table 2.1: *Standard atmospheric conditions where h is elevation above MSL, ρ is air density, T is air temperature, and μ is dynamic viscosity (Auld, 2024).*

h **(MSL)** [m]	$\rho \left[\frac{\text{kg}}{\text{m}^3}\right]$	T [K]	$\mu \left[\frac{\text{kg}}{\text{m s}}\right]$
457	1.185	285.35	1.775×10^{-5}

Using the equation for Reynolds number,

$$Re = \frac{\rho v L}{\mu} \tag{2.2}$$

where Re is Reynolds number, ρ is fluid density, v is fluid velocity, L is the characteristic length, and μ is the fluid dynamic viscosity, (in our analyses, the "fluid" is air and the characteristic length is the "chord" length) we calculated our Reynolds number as shown in Equation 2.3.

$$\begin{aligned} Re &= \frac{\left(1.185\,\frac{\text{kg}}{\text{m}^3}\right)\left(17.9\,\frac{\text{m}}{\text{s}}\right)(20\,\text{cm})}{\left(1.775 \times 10^{-5}\,\frac{\text{kg}}{\text{m s}}\right)} \\ &\approx 240\,000 \end{aligned} \tag{2.3}$$

Using a Reynolds number of 240 000, we simulated the airfoil performance using XFLR5, a free airfoil analysis tool, and imported the data into a MATrix LABoratory (MATLAB) script to perform further analysis (techwinder, 2024; MathWorks, 2024).

2.4 Initial Main Wing Analysis

To determine which of the three airfoils would be best for our mission, we first ran the airfoils shown in Figure 2.1 through XFLR5. The results of this analysis are shown in Figure 2.4.

The results from XLFR5 (Figure 2.4) were imported into the MATLAB script shown in Section A.2. From this script, the parameters tabulated in Table 2.2 were generated.

Based on the results in Table 2.2, we chose to move forward with the NACA 4412. It had the best lift-drag ratio and a high stall angle.

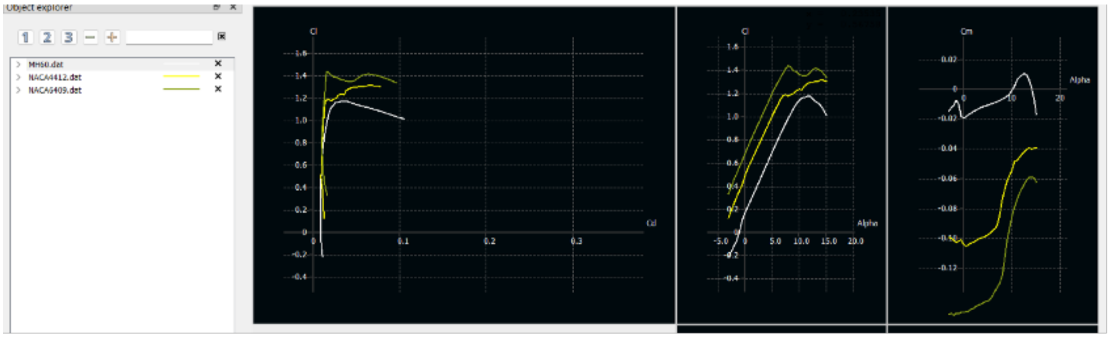


Figure 2.4: *The results from running our original three main airfoil choices through XLFR5 with the Reynolds number calculated in Section 2.3.*

Table 2.2: *The results of the initial main wing airfoil analysis, where $\frac{dC_L}{d\alpha}$ is the slope of the coefficient versus angle of attack curve, $C_{L_{max}}$ is the maximum value of the coefficient of lift curve, $C_{D_{max}}$ is the maximum value of the drag coefficient, and $\left(\frac{L}{D}\right)_{max}$ is the maximum value of the lift-drag ratio.*

Airfoil	$\frac{dC_L}{d\alpha}$	$C_{L_{\max}}$	$C_{D_{\max}}$	$\left(\frac{L}{D}\right)_{\max}$	α_{stall}
NACA 4412	0.1001	1.3127	0.0777	19.3466	14
NACA 6409	0.0990	1.4378	0.0962	17.9252	8
MH 60	0.1061	1.1728	0.1051	12.0892	11.5

2.5 Initial Stabilizer Analysis

Each of the stabilizers have a chord of approximately 15 cm; so, their corresponding Reynolds number is approximately 180 000 (see Equation 2.2). Plugging the airfoils shown in Figure 2.2 and Figure 2.3 into XFLR5, we obtained the results shown in Figure 2.5.

The NACA 0009 and NACA 0010 converged well, but the E423 did not. Running the E432 airfoil again with a higher mach and Reynolds number yielded better results, as shown in Figure 2.6.

From the data, we began to be concerned that the E423 would produce too much lift at higher speeds for a horizontal stabilizer, but chose to continue the analysis with the E423 as our horizontal stabilizer. For the two vertical stabilizers, the data at cruise speed was almost identical, but we decided to move forward with the NACA 0010 because its average pitching moment was closer to zero—and a neutral pitching moment is desirable for a vertical stabilizer.

2.6 Reynolds Number Ranges

At this point in the analysis, we have selected the following airfoils:

- Wing: NACA 4412
- Horizontal Stabilizer: E423
- Vertical Stabilizer: NACA 0010

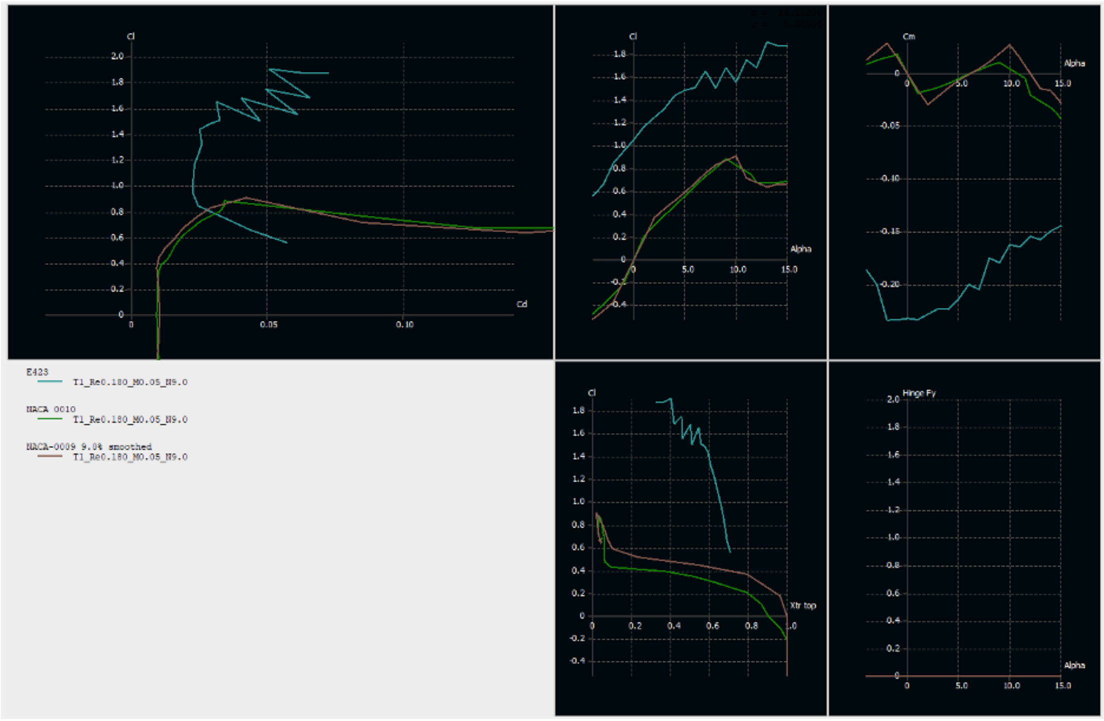


Figure 2.5: *The results from running our three stabilizer airfoils through XFLR5.*

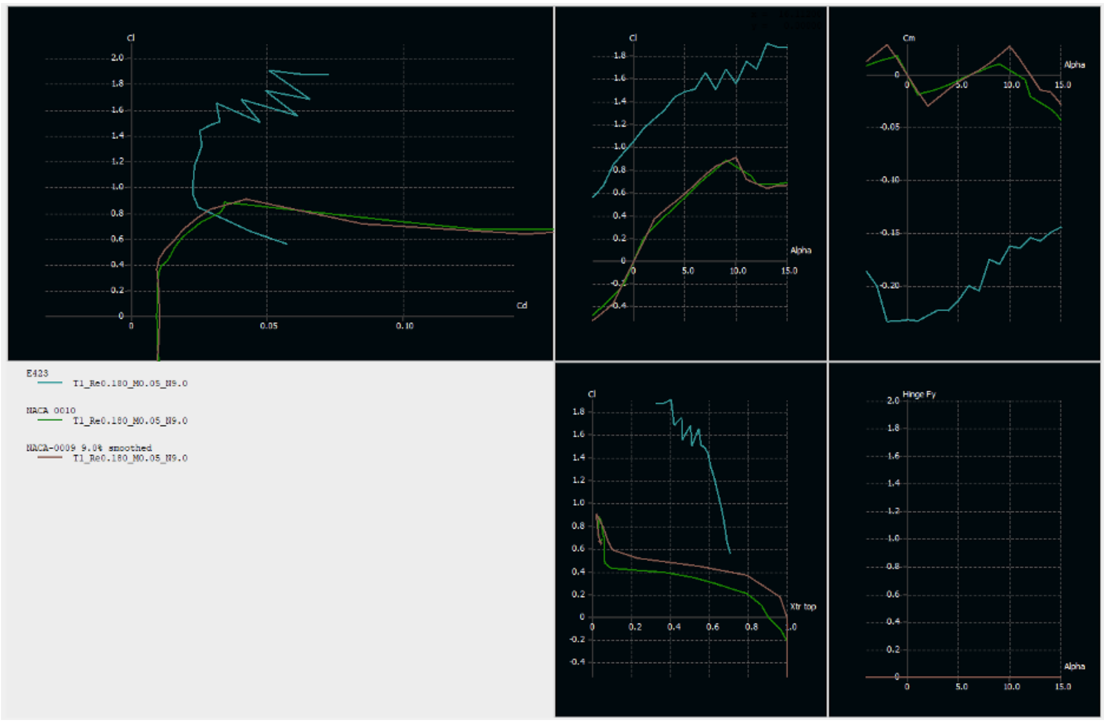


Figure 2.6: *The results from running the E423 stabilizer in XFLR5 with a higher Reynolds number.*

To further evaluate these choices, we ran XFLR5 analyses for the airfoils at a range of Reynolds number that our UAV would encounter during flight. To get a better sense of the airfoil performance, we chose a Mach number range of 0.040 to 0.080, which represents a reasonable range of speeds remote-controlled (RC) aircraft normally fly at during cruise. The corresponding Reynolds numbers this range produces are shown in Table 2.3.

Table 2.3: *Range of Reynolds numbers calculated at different Mach numbers M.*

M	Re **(wing)**	Re **(stabilizer)**
0.040	180 000	135 000
0.053	240 000	180 000
0.066	300 000	225 000
0.080	360 000	270 000

We also calculated Reynolds numbers at ground level, but the results of the ground analysis were nearly identical to the cruise altitude; so, going forward, we will only consider the cruise altitude analysis results.

The results of the XFLR5 analyses with a range of Reynolds numbers are shown in Figure 2.7, Figure 2.8, and Figure 2.9.

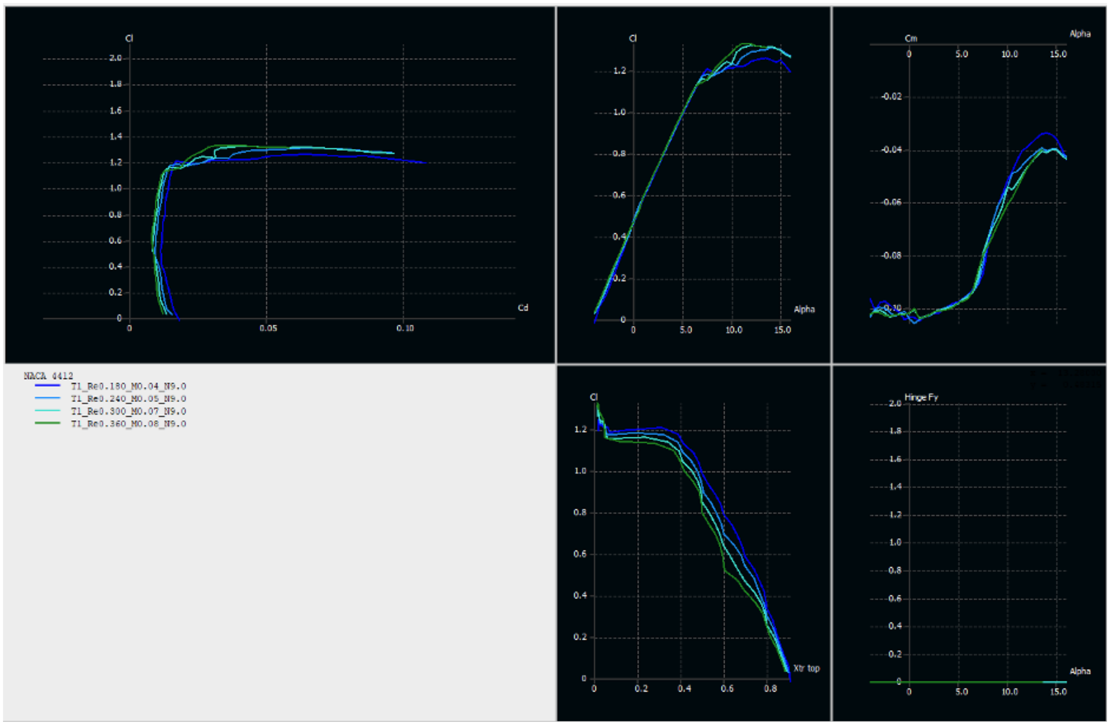


Figure 2.7: *The XFLR5 results from running a range of Reynolds numbers on the NACA 4412.*

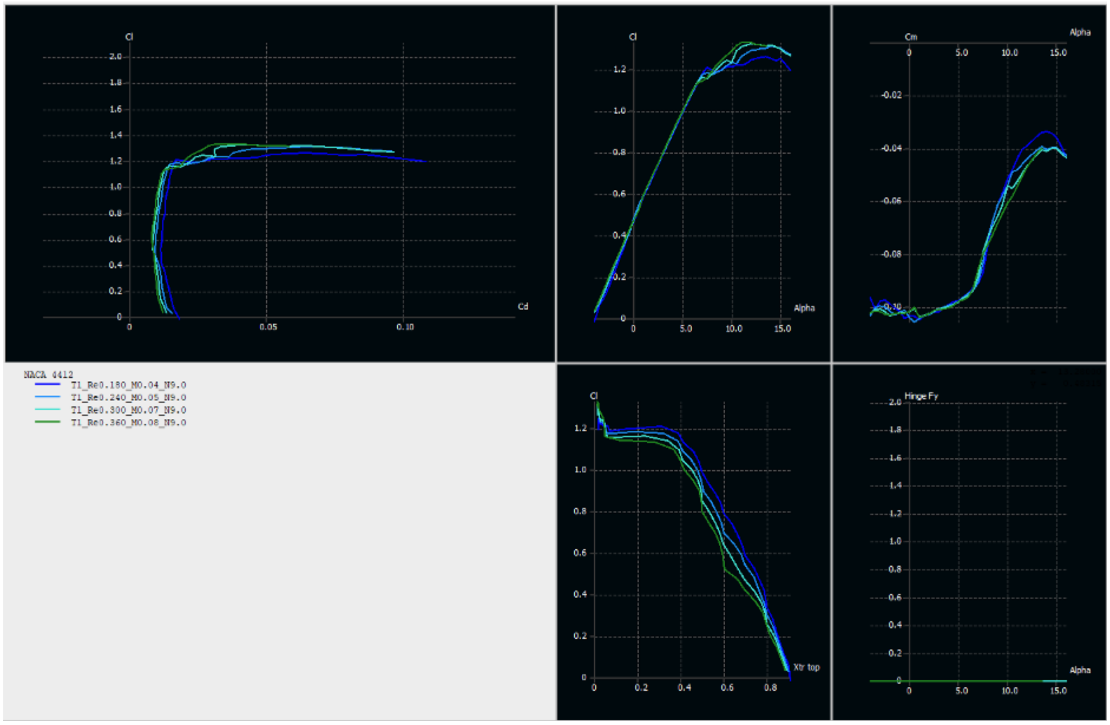


Figure 2.8: *The XFLR5 results from running a range of Reynolds numbers on the E423.*

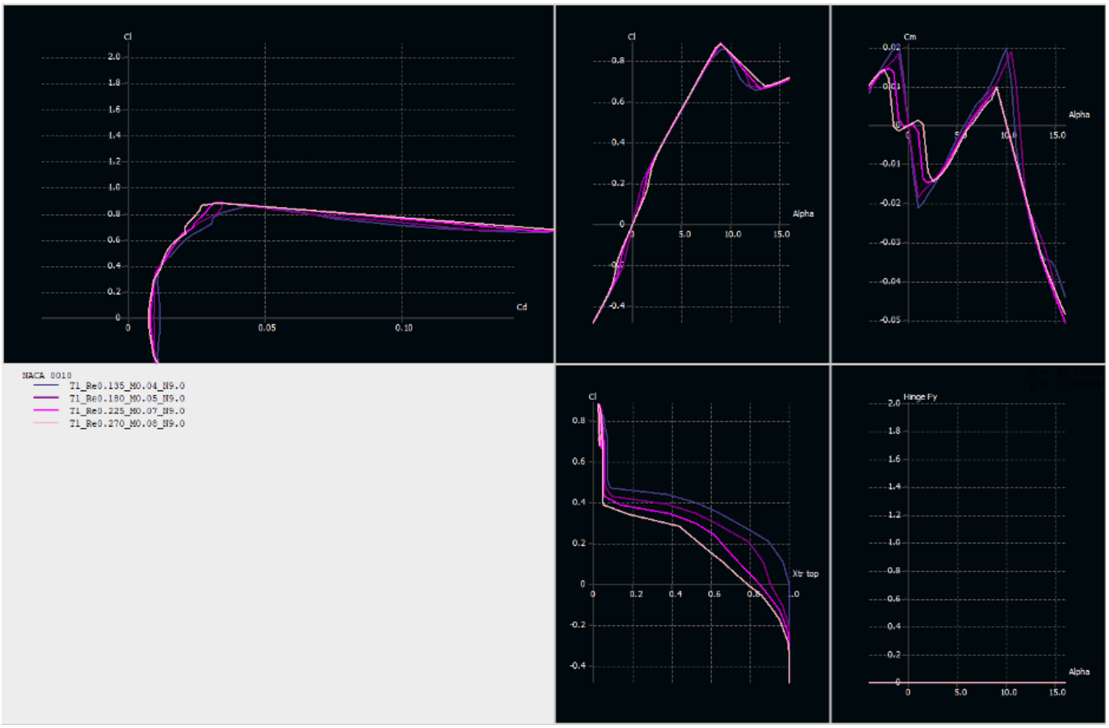


Figure 2.9: *The XFLR5 results from running a range of Reynolds numbers on the NACA 0010.*

From these XFLR5 results, we now had a better sense of the airfoils over a range of flight conditions. While we were initially drawn to the E423 for its high lift-drag ratio, it had widely varying values for its lift coefficient, which may make the UAV unstable if we were to use it for a horizontal stabilizer and fly at high speeds.

Our initial research did point us towards using a symmetric airfoil for the horizontal stabilizer, but we wanted to experiment with a cambered airfoil. Based on these discouraging XFLR5 results, we decided to fall back to the NACA 0010 as both the horizontal and vertical stabilizer. The NACA 4412 continued to seem like a good option for a main wing airfoil.

3

Sizing

3.1 Wing and Tail

The flight performance and structural worked together to generate the table of initial wing sizes shown in Table 3.1.

Table 3.1: *Initial wing sizing parameters.*

Name	Variable	Dimension
wingspan	b	216 cm
wing chord	c	20 cm
tailspan	b_t	38 cm
tail chord	c_t	15 cm
vertical stabilizer height	h_v	19 cm
vertical stabilizer chord	c_v	15 cm
tail moment arm	l_t	91 cm

The primary driver for these measurements is the requirement to achieve a loiter time of at least 45 min. Aircraft designed for long endurance typically feature extended wingspans and high AR, which informed our selection of b and c. The remaining dimensions were determined by referencing similar high-efficiency, glider-type aircraft.

3.2 Fuselage

For the Banshee to hold the payload components, batteries, and electronics required to complete its mission, we designed an initial fuselage to be 41 cm × 23 cm × 23 cm. More detailed volume and layout analysis will be required to determine whether the fuselage size needs to be increased or decreased. A sketch of the fuselage is shown in Figure 3.1.

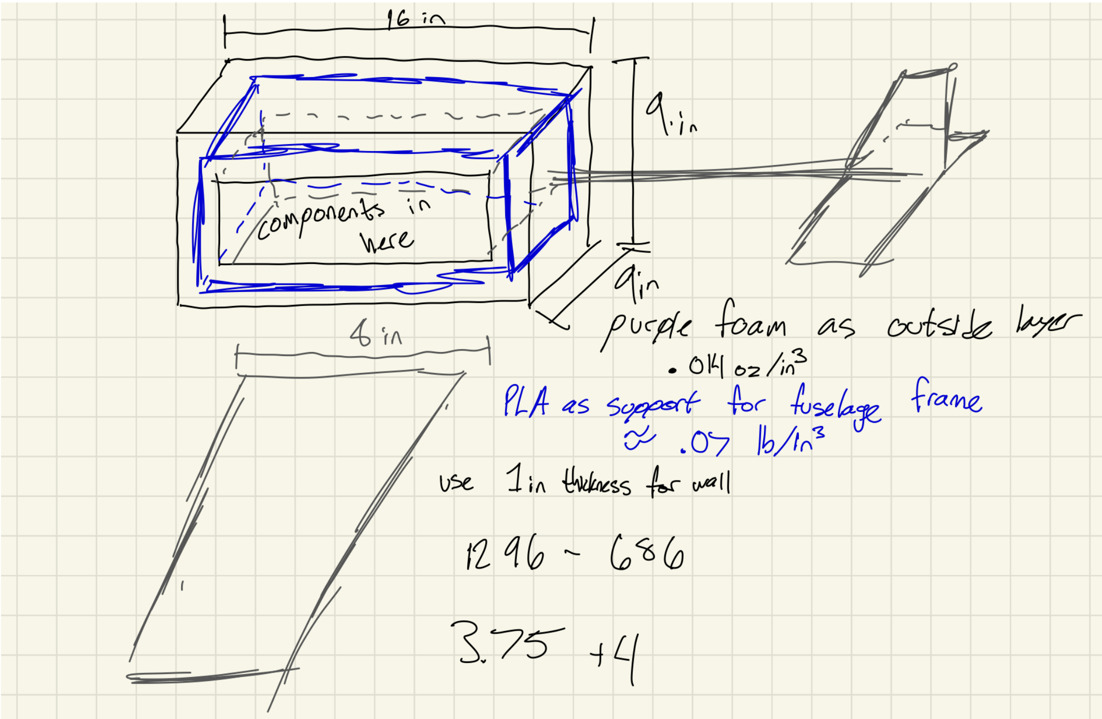


Figure 3.1: *A sketch of the fuselage frame and dimensions.*

4 Weight

4.1 Structures

To estimate the weight of the aircraft wings, we assumed the wings would be composed of carbon-fiber spars encased in "purple foam" (presumably a type of polystyrene foam). Using the densities and weights given by Grager (2024), we calculated and tabulated the estimated weights of different section of the Banshee, shown in Table 4.1.

Table 4.1: *Initial estimates of the different structural components of the Banshee.*

Component	Mass
wing foam	167 g
wing spars	193 g
tail foam	23 g
tail spar	65 g
boom	50 g
fuselage	488 g
	986 g

Table 4.1 only considers the mass of the Banshee's structural components.

4.2 Payload

According to our mission proposal, the Banshee must be equipped with components capable of detecting, tracking, and jamming a sUAS. Research is ongoing to determine what components we need to meet this requirement.

Based on some preliminary research, however, we have determined we will need a jamming payload and an antenna to propagate the jamming signals. The specifications of the jamming modules and antenna are tabulated in Table 4.2.

These jamming modules are lighter than the original payload we specified when we ran our weight and center of gravity (CG) calculations—in part because the original payload component also had tracking and detection capabilities. The payload mass we specified in our analysis was 1.275 kg.

Table 4.2: *Initial estimates of the Banshee's payload components.*

Component	Mass	Size (L x W x H)
Signal Jamming Module (2.4 GHz)	102 g	97 mm × 43 mm × 17 mm
Signal Jamming Module (5.8 GHz)	102 g	97 mm × 43 mm × 17 mm
Antenna	166 g	293 mm × 98 mm × 35 mm
	370 g	

4.3 Total Weight

Based on the total weights calculated in Table 4.1 and Table 4.2, the currently estimated structures and payload weight is 1.36 kg. The weight we used in our CG calculations was 2.26 kg.

We expect the actual weight to be closer to 10 kg once the weight of batteries, electronics, and other payload components are estimated. These trade studies will be carried in the coming weeks.

4.4 Center of Gravity

To provide the flight performance team with a sense of the aircraft balance, the structural engineering team conducted a rough CG estimate, considering only the approximate weights and locations of the payload and structural components discussed above. The MATLAB script used to estimate the CG can be found in Section A.1. The resulting CG position was approximately 36 cm. For reference, the trailing edge of the wing is at approximately 41

Obviously, there is a significant amount of mass missing from this analysis, but it served as a reasonable starting point.

5

Concept Skteches

5.1 Variant 1 - Original Design

Concept sketch variant one, shown in Figure 5.1, is our favorite design for our UAV. It is the simplest of the three designs and follows a common design pattern we noted from COTS RC aircraft and past senior design projects.

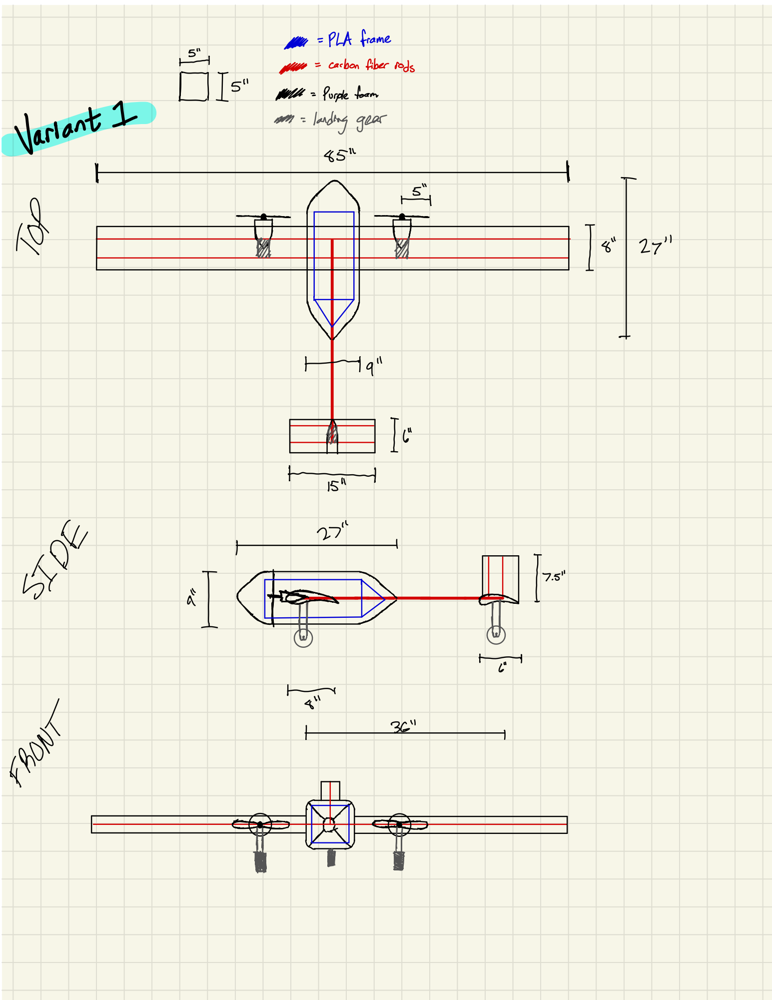


Figure 5.1

5.2 Variant 2 - Wider and Flatter Fuselage

Variant two, shown in Figure 5.2, features a wider and flatter to reduce drag and weight. This design would be dependent on the smaller fuselage still being able to hold the Banshee's required payload, batteries, and electronics.

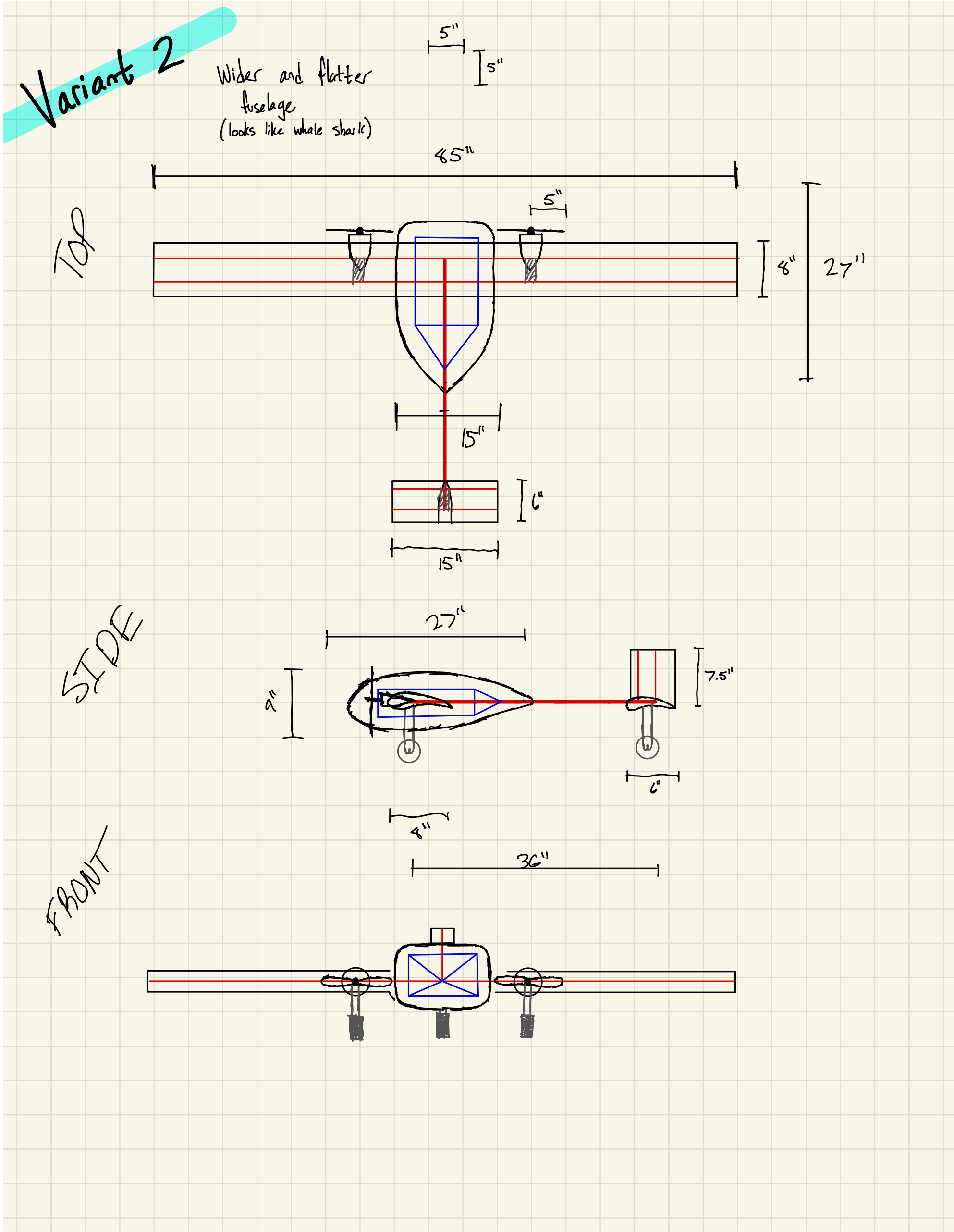


Figure 5.2

5.3 Variant 3 - Wing-Integrated Fuselage

The final variant, shown in Figure 5.3, is more unconventional and may be harder to manufacture and structurally analyze. It features a wing-integrated fuselage, which—besides looking cool—may significantly reduce drag.

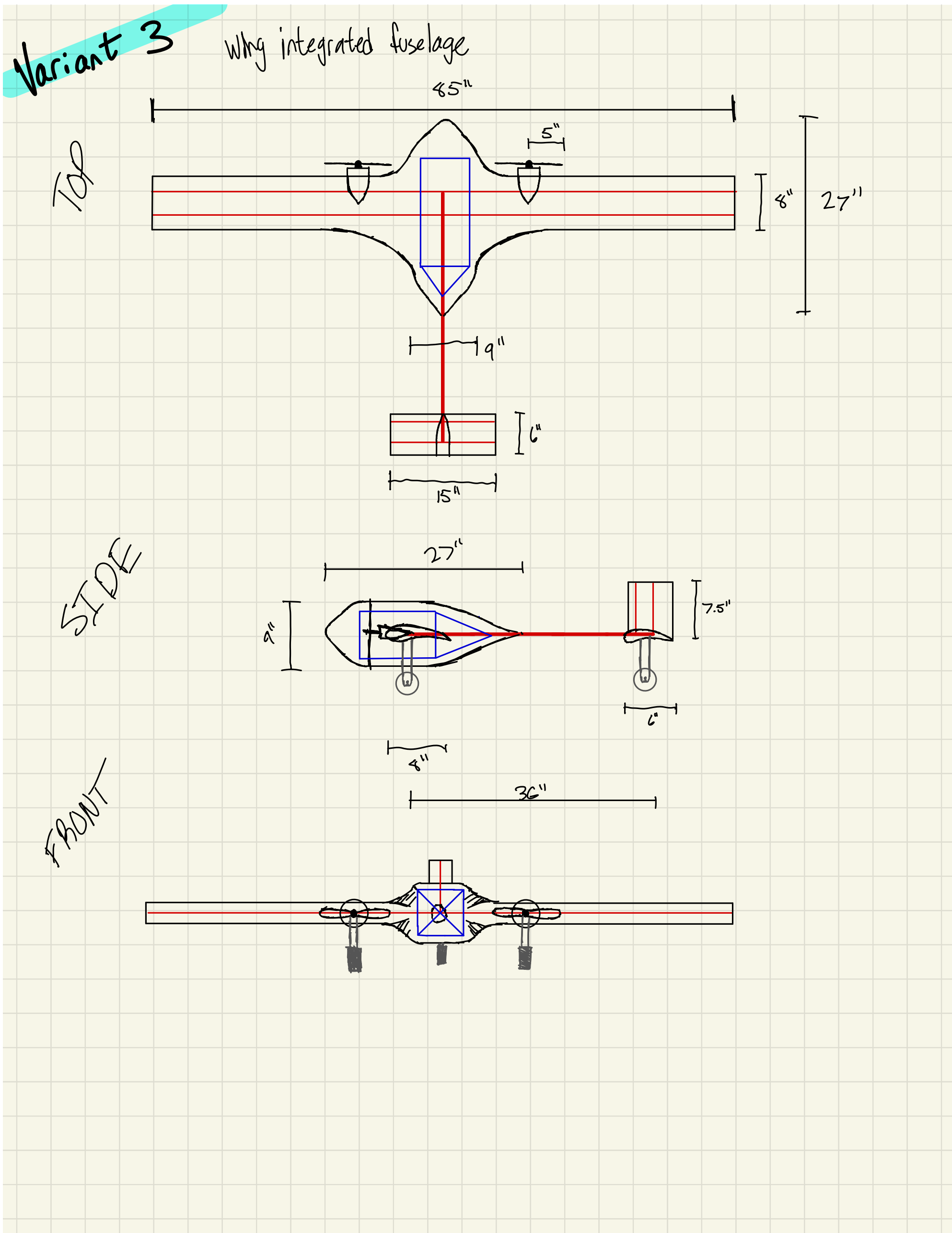


Figure 5.3

6
Conclusion

Based on the results of this initial weight and sizing, we have a much better sense of the shape and size of our UAV. We chose the NACA 4412 for our main wing airfoil and the NACA 0010 for the horizontal and vertical stabilizers. The weight and size of our aircraft leave plenty of headroom for batteries, motors, electronics, and additional payload components, which will be analyzed in closer detail in future trade studies. Additionally, we have three concept sketches that match our initial sizing estimates from which we can generate an initial CAD model.

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A

Appendix A

A.1 calculate_cg.m

```
% Calculate the CG.
%   CG Calculator for StratoShield, AERE 4610 Fall 2024. All measurements
%   of distance are in inches and weight is in pounds. The position of each
%   part is measured from the tip of the aircraft's nose.
clear; clc; close all;

%% Aircraft values

c_wing = 8; % [in]
c_tail = 6; % [in]
boom_length = 36; % [in]

%% Parts

% Wing foam
W_wing_foam = 0.368; % [lb]
x_wing_foam = 1.5 * c_wing; % [in]

% Wing spar (2 for the wing)
W_wing_spar = 0.2125; % lb
x_wing_spar = 1.5 * c_wing; % in

% Tail foam
W_tail_foam = 0.05; % lb
x_tail_foam = boom_length - c_tail / 2;

% Tail spar (6 for the tail)
W_tail_spar = 0.024; % lb
x_tail_spar = boom_length - c_tail / 2;

% Boom
W_boom = 0.11 / 36 * boom_length; % lb
```

```
x_boom = boom_length / 2; % in

% Fuselage
W_fuselage = 1.07625; % lb
x_fuselage = 27 / 2; % in

% Payload
W_payload = 2.8108938; % lb
x_payload = 27 / 2; % in

%% Calculations

W_total = W_wing_foam ...
    + 2 * W_wing_spar ...
    + W_tail_foam ...
    + 6 * W_tail_spar ...
    + W_boom ...
    + W_fuselage ...
    + W_payload;

CG = (W_wing_foam * x_wing_foam ...
    + 2 * W_wing_spar * x_wing_spar ...
    + W_tail_foam * x_tail_foam ...
    + 6 * W_tail_spar * x_tail_spar ...
    + W_boom * x_boom ...
    + W_fuselage * x_fuselage ...
    + W_payload * x_payload) ...
    / W_total;

%% Output

fprintf("Total weight = %g lb\n", W_total);
fprintf("CG position = %g in\n", CG);
```

A.2 initial_wing_analysis.m

```
clear; clc; clf;
% ========================================================================
%    _____________  ___  __________  _____ __  ______________    ____
%   / ___/_  __/ __ \/   |/_  __/ __ \/ ___// / / /  _/ ____/ /   / __ \
%   \__ \ / / / /_/ / /| | / / / / / /\__ \/ /_/ // // __/ / /   / / / /
%  ___/ // / / _, _/ ___ |/ / / /_/ /___/ / __  // // /___/ /___/ /_/ /
% /____//_/ /_/ |_/_/  |_/_/  \____//____/_/ /_/___/_____/_____/_____/

% ========================================================================
% Initial Sizing | Flight Performance Subteam

% Patricia Belinga (FP Engineer)
% Thomas Housley (FP Lead)
% Bradley Norwall (CAD Lead)

% ========================================================================

% title made using:
% https://patorjk.com/software/taag/#p=display&f=Slant&t=STRATOSHIELD



%% Reynolds Number Calculations

z = 1500; % [ft] (altitude 150m above Ames from SL reference)

% Standard Atmo values/ratios from Initial Airfoil Sizing source @ 1500 ft
T = 12.2+273.15; % [K]
p_p0 = .947;
rho_rho0 = 0.9568;
mu_mu0 = 0.992;

% sea level conditions for ratios
p0 = 14.7; % [psi]
rho0 = 0.002378; % [slug / ft^3]
mu0 = 3.737e-7; % [slug / ft * s]
Cruise.Vel = 58.6667; % [ft / s] = 40 mph
Cruise.Vel_SI = 17.8816; % [m / s]

% chord was used for Reynold's, span is just under max wingspan and kinda
% = 10 max Aspect Ratio recommendation from class THESE CHANGE BASED ON
% SIZING PARAMETERS FROM STRUCTURES TEAM ********************************
% IF THAT HAPPENS THEN NEW XFLR5 DATA NEEDS TO BE AQUIRED
chord = 8 / 12; % [ft]
span = 85 / 12; % [ft]
```

```
% now getting actual values
rho = rho_rho0 * rho0;
mu = mu_mu0 * mu0;

% Reynolds # at cruise speed (40mph)
Cruise.Re = (rho * Cruise.Vel * chord) / mu;

% using SI because mach is unitless, verified that it matches cruise in MPH
a = sqrt(T * 1.4 * 287);
Cruise.Mach = Cruise.Vel_SI / a;

%% Loading CSV XFLR5 data

% ========================================================================
% IF YOU ARE RUNNING THIS ON YOUR OWN COMPUTER, THE XFLR5 CSV FILES WILL
% PROBABLY HAVE A DIFFERENT PATH. IF THEY HAVE THE SAME FILENAME AS
% FORMATTED BELOW:

% -RIGHT CLICK ON THE FILE IN YOUR FILE EXPLORER
% -SELECT "COPY AS PATH" (CTRL+SHIFT+C)
% -PASTE PATH INTO directory VARIABLE BELOW
% -REMOVE THE FILENAME FROM THE PATH STRING YOU PASTED

% then you should be able to use this code
% ========================================================================

naca4412_filename = "NACA_4412_data.csv";
naca6409_filename = "NACA_6409_data.csv";
mh60_filename = "MH60_data.csv";
e423_filename = "E423_data.csv";

directory = "C:\Skewl\AerE4610\Airfoil XFLR5 initial sizing data\";

% creating path string for each airfoil

naca4412_path = strjoin(directory + naca4412_filename);
naca6409_path = strjoin(directory + naca6409_filename);
mh60_path = strjoin(directory + mh60_filename);
e423_path = strjoin(directory + e423_filename);

% loading variable columns into table so its easy to identify columns

naca4412_data = readtable(naca4412_path, ...
    "Delimiter", ",", "VariableNamingRule", "preserve");
naca6409_data = readtable(naca6409_path, ...
    "Delimiter", ",", "VariableNamingRule", "preserve");
```

```
mh60_data = readtable(mh60_path, ...
    "Delimiter", ",", "VariableNamingRule", "preserve");
e423_data = readtable(e423_path, ...
    "Delimiter", ",", "VariableNamingRule", "preserve");

% this is for loops later on dw
n_naca4412 = size(naca4412_data);
n_naca6409 = size(naca6409_data);
n_mh60 = size(mh60_data);
n_e423 = size(e423_data);


% recreating XFLR5 plots for viewing
% this can be done for tail airfoils too so I can add that code

figure(1)
plot(naca4412_data.CD, naca4412_data.CL)
hold on
plot(naca6409_data.CD, naca6409_data.CL)
plot(mh60_data.CD, mh60_data.CL)
% plot(e423_data.CD, e423_data.CL)
grid on
title("C_l vs. C_d")
legend("NACA 4412", "NACA 6409", "MH60")%,"E423 h-stabilizer")
xlabel("C_d")
ylabel("C_l")
hold off

% time for curve fitting (throwing up in my mouth rn)
% im sorry its hardcoded on the rows and columns
fit_naca4412 = polyfit( ...
    naca4412_data.alpha(5:22), naca4412_data.CL(5:22), 1);
fit_naca6409 = polyfit( ...
    naca6409_data.alpha(5:22), naca6409_data.CL(5:22), 1);
fit_mh60 = polyfit(mh60_data.alpha(5:22), mh60_data.CL(5:22), 1);
fit_e423 = polyfit(e423_data.alpha(5:22), e423_data.CL(5:22), 1);

figure(2)
plot(naca4412_data.alpha, naca4412_data.CL)
hold on
plot(naca6409_data.alpha, naca6409_data.CL)
plot(mh60_data.alpha, mh60_data.CL)
% plot(e423_data.alpha, e423_data.CL)

% plotting fits (throwing up in my mouth again)
plot(naca4412_data.alpha, ...
    fit_naca4412(1) .* naca4412_data.alpha + fit_naca4412(2), ...
```

```
    "LineStyle", "--", "Color", "black")
plot(naca6409_data.alpha, ...
    fit_naca6409(1) .* naca6409_data.alpha + fit_naca6409(2), ...
    "LineStyle", "--", "Color", "black")
plot(mh60_data.alpha, ...
    fit_mh60(1) .* mh60_data.alpha + fit_mh60(2), ...
    "LineStyle", "--", "Color", "black")
% plot(e423_data.alpha, ...
%     fit_e423(1) .* e423_data.alpha + fit_e423(2), ...
%     "LineStyle", "--", "Color", "black")

grid on
title("AoA vs. C_l")
xlabel("AoA (degrees)")
ylabel("C_l")
xline(0)
yline(0)
legend("NACA 4412", "NACA 6409", "MH60") % , "E423 h-stabilizer")
hold off

figure(3)
plot(naca4412_data.alpha, naca4412_data.Cm)
hold on
plot(naca6409_data.alpha, naca6409_data.Cm)
plot(mh60_data.alpha, mh60_data.Cm)
% plot(e423_data.alpha, e423_data.Cm)
grid on
title("AoA vs. C_m")
xlabel("AoA (degrees)")
ylabel("C_m")
xline(0)
yline(0)
xlim([-4,20])
legend("NACA 4412", "NACA 6409", "MH60") % , "E423 h-stabilizer")
hold off

% ok ze curves hath been createth
% i only did curves again so i could have the slopes for later hehe
% from the polyfits can then identify L/D

%% LIFT/DRAG CALCULATIONS (OPTIMIZATION)
% all of the (important) calculated data thats importnant is dumped into
% calc struct

% Cl/Alpha 2D slopes
calc.naca4412.Lalpha = fit_naca4412(1);
calc.naca6409.Lalpha = fit_naca6409(1);
```

```
calc.mh60.Lalpha = fit_mh60(1);
calc.e423.Lalpha = fit_e423(1);

% Cl^(3/2)/C_d ratio matrices
% aere 261 notes on max battery prop endurance (see teams folder w slides)
% calc.naca4412.LD_aloft_values = (naca4412_data.CL).^(3 / 2) ...
%     ./ naca4412_data.CD;
% calc.naca6409.LD_aloft_values = (naca6409_data.CL).^(3 / 2) ...
%     ./ naca6409_data.CD;
% calc.mh60.LD_aloft_values = (mh60_data.CL).^(3 / 2) ./ mh60_data.CD;
% calc.e423.LD_aloft_values = (e423_data.CL).^(3 / 2) ./ e423_data.CD;

% maximum Cl^(3/2)/C_d ratios
% calc.naca4412.LDmax_aloft = max(calc.naca4412.LD_aloft_values);
% calc.naca6409.LDmax_aloft = max(calc.naca6409.LD_aloft_values);
% calc.mh60.LDmax_aloft = max(calc.mh60.LD_aloft_values);
% calc.e423.LDmax_aloft = max(calc.e423.LD_aloft_values);

% C_l maxes and the row location of the max to later find corresponding
% stall angle
[calc.naca4412.CLmax, calc.naca4412.stall_index]= max(naca4412_data.CL);
[calc.naca6409.CLmax, calc.naca6409.stall_index] = max(naca6409_data.CL);
[calc.mh60.CLmax, calc.mh60.stall_index]= max(mh60_data.CL);
[calc.e423.CLmax, calc.e423.stall_index]= max(e423_data.CL);

% C_d maxes now
calc.naca4412.CDmax = max(naca4412_data.CD);
calc.naca6409.CDmax = max(naca6409_data.CD);
calc.mh60.CDmax = max(mh60_data.CD);
calc.e423.CDmax = max(e423_data.CD);


calc.naca4412.LD_aloft_max = (calc.naca4412.CLmax)^(3 / 2) ...
    / calc.naca4412.CDmax;
calc.naca6409.LD_aloft_max = (calc.naca6409.CLmax)^(3 / 2) ...
    / calc.naca6409.CDmax;
calc.mh60.LD_aloft_max = (calc.mh60.CLmax)^(3 / 2) / calc.mh60.CDmax;
calc.e423.LD_aloft_max = (calc.e423.CLmax)^(3 / 2) / calc.e423.CDmax;

% afformentioned alpha_stall
calc.naca4412.alpha_stall = naca4412_data.alpha(calc.naca4412.stall_index);
calc.naca6409.alpha_stall = naca6409_data.alpha(calc.naca6409.stall_index);
calc.mh60.alpha_stall = mh60_data.alpha(calc.mh60.stall_index);
calc.e423.alpha_stall = e423_data.alpha(calc.e423.stall_index);

% HOWEVER, THESE RESULTS ARE BASED ON CL_MAX
% can also get "linear stall alpha"
```

```
% this value is lower than other derived alpha_stall but may be good
% comparison?
% best way i can think of deriving it in a repeatable way is having a
% tolerance between linear fit line and actual XFLR5 data

tolerance = 0.1;

difference_naca4412 = ...
    (fit_naca4412(1) .* naca4412_data.alpha + fit_naca4412(2)) ...
    - naca4412_data.CL;
difference_naca6409 = ...
    (fit_naca6409(1) .* naca6409_data.alpha + fit_naca6409(2)) ...
    - naca6409_data.CL;
difference_mh60 = (fit_mh60(1) .* mh60_data.alpha + fit_mh60(2)) ...
    - mh60_data.CL;
difference_e423 = (fit_e423(1) .* e423_data.alpha + fit_e423(2)) ...
    - e423_data.CL;

calc.naca4412.linearstall = 0;
calc.naca6409.linearstall = 0;
calc.mh60.linearstall = 0;
calc.e423.linearstall = 0;
% gross but short nested loops sorry this code is O(n^100)
for i = 1:n_naca4412(1)
    if difference_naca4412(i) >  tolerance
        calc.naca4412.linearstall = naca4412_data.alpha(i);
        break;
    end
end
for i = 1:n_naca6409(1)
    if difference_naca6409(i) > tolerance
        calc.naca6409.linearstall = naca6409_data.alpha(i);
        break;
    end
end
for i = 1:n_mh60(1)
    if difference_mh60(i) > tolerance
        calc.mh60.linearstall = mh60_data.alpha(i);
        break;
    end
end
for i = 1:n_e423(1)
    if difference_e423(i) > tolerance
        calc.e423.linearstall = e423_data.alpha(i);
        break;
    end
end
```

```
% ok so basically after doing all that they are almost the same but having
% 2 is good ig? idk i am tired

% GLORIOUS DATA AQUIRED
% time for consideration

% i would start controls but need moments of inertia and I will throw up
% making 4 billion assumptions for that so no


disp("NACA 4412:"); disp(calc.naca4412)
disp("NACA 6409:"); disp(calc.naca6409)
disp("MH60:"); disp(calc.mh60)
disp("E423:"); disp(calc.e423)


%% Reynolds Number Range (RUN CLEAR, CLC BEFORE RUNNING)

% Need a range of Re values for flight at 30mph - 60mph on ground and
% cruise altitude

ground.z = 304; % [m] (~altitude on ground @ ames from SL) (1000ft)
cruise.z = 457; % [m] (altitude ~150m above ames from SL) (1500ft)

% Standard Atmo values & ratios from Initial Airfoil Sizing source @ cruise
cruise.T = 12.2 + 273.15; % [K]
cruise.p_p0 = 0.947;
cruise.rho_rho0 = 0.9568;
cruise.mu_mu0 = 0.992;

% Standard Atmo values & ratios from Initial Airfoil Sizing source @ ground
ground.T = 13.2 + 273.15; % [K]
ground.p_p0 = 0.9644;
ground.rho_rho0 = 0.9711;
ground.mu_mu0 = 0.9947;

% sea level conditions for ratios
p0 = 101325; % [Pa]
rho0 = 1.225; % [kg / m^3]
mu0 = 1.789e-5; % [kg / (m * s)]


vel = [30, 40, 50, 60] ./ 2.237; % mph --> m/s conversion

% chord was used for Reynold's, span is just under max wingspan and about
% equal 10 max Aspect Ratio recommendation from Class
```

```
% horizontal stabilizer chord was arbitrary choice and may be subject to
% change

chord_wing = 8 / 39.37; % in --> meters
span_wing = 85 / 39.37; % in --> meters

chord_hStab = 6 / 39.37; % in --> meters
span_hor = 15 / 39.37; % in --> meters


% now getting actual values
% cruise conditions
cruise.rho = cruise.rho_rho0 * rho0;
cruise.mu = cruise.mu_mu0 * mu0;

% ground conditions
ground.rho = ground.rho_rho0 * rho0;
ground.mu = ground.mu_mu0 * mu0;

% Reynolds #s
cruise.Re_wing = (cruise.rho .* vel .* chord_wing) ./ cruise.mu;
cruise.Re_stab = (cruise.rho .* vel .* chord_hStab) ./ cruise.mu;
ground.Re_wing = (ground.rho .* vel .* chord_wing) ./ ground.mu;

% finding machs
cruise.a = sqrt(cruise.T * 1.4 * 287);
cruise.mach = vel ./ cruise.a;
ground.a = sqrt(ground.T * 1.4 * 287);
ground.mach = vel ./ ground.a;

disp("Ground Conditions to be input to XFLR5")
disp(ground)
disp("Cruise Conditions to be input to XFLR5")
disp(cruise)
```