Project Points:	
Wing Analysis Function:	700 points
Write-Up:	200 points
Total:	900 points

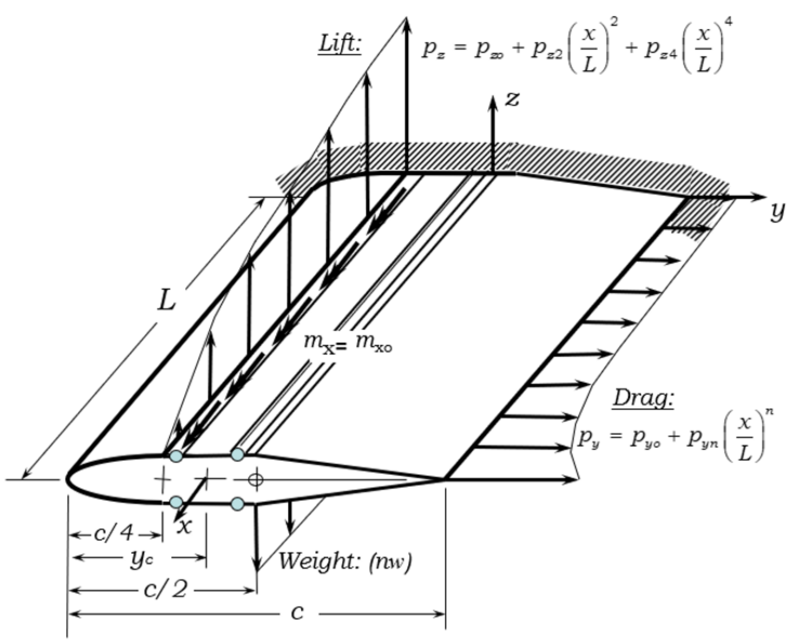
MATLAB Project: Single Cell Wing Bending, Torsion, and Shear Analysis
Due Date: Upload zip folder to Canvas by 11:58 PM, Thursday March 13, 2025

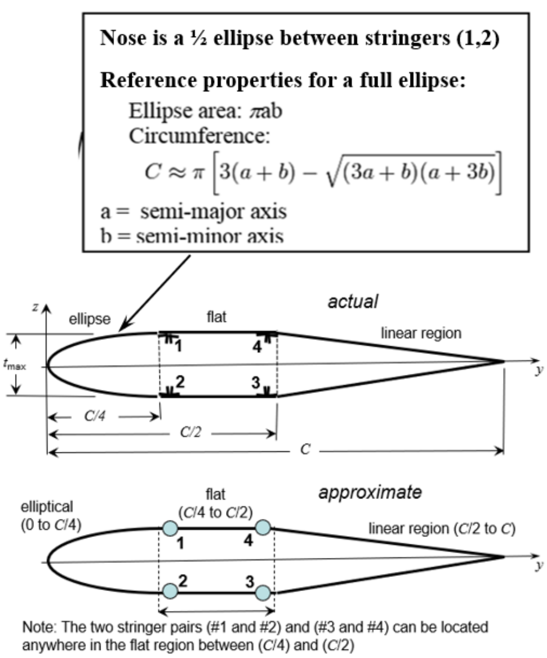
Files in CANVAS ZIP folder. Extract and put into your MATLAB folder:

Wing_Analysis_Main.p	Kosmatka Wing Analysis Main Code (encrypted p-version)
Wing_Analysis_Function.p	Kosmatka Wing Analysis Function (encrypted p-version)
Wing_Analysis_Function (Student Skeleton).m	Recommended Wing Function Starter Code
SE160A_2_Wing_Analysis_WriteUp_(v1).pdf	Project Write Up (Version 1)
SE160A_2_Wing_Analysis_Input.xlsx	Wing Analysis Excel Input File (Sample)
SE160A_2_Wing_Analysis_Input (Blank).xlsx	Wing Analysis Excel Input File (Blank)
SE160A_2_Wing_Analysis_Output.xlsx	Wing Analysis Excel Output File (Sample)
SE160A_2_Wing_Analysis_Output (Blank).xlsx	Wing Analysis Excel Output File (Blank)

On or before March 13th, 2025, you will load the following six files into a zip folder and then upload the zip folder to the SE-160A Canvas Gradescope Account.

SE160A_2_LastName_FirstName.zip	Zip Folder
Wing_Analysis_Function.m	Your Wing Analysis Function With Your Name and PID Inside
Wing_Analysis_LastName_FirstName.pdf	Your solutions to answered questions and design studies.
SE160A_2_Wing_Analysis_Input_AL.xlsx	Analysis Excel Input File (Design Study Aluminum)
SE160A_2_Wing_Analysis_Output_AL.xlsx	Analysis Excel Output File (Design Study Aluminum)
SE160A_2_Wing_Analysis_Input_CE.xlsx	Analysis Excel Input File (Design Study Carbon/Epoxy)
SE160A_2_Wing_Analysis_Output_CE.xlsx	Analysis Excel Output File (Design Study Carbon/Epoxy)





INTRODUCTION

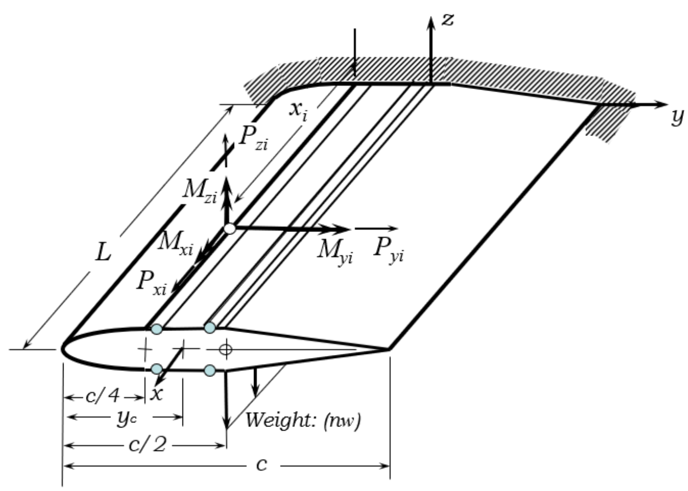
Maneuvering aircraft (and other aerospace vehicles) produce loads on their lifting and control surfaces. These loads vary greatly for different maneuvering conditions. In this MATLAB project, you will analyze a simplified one-cell uniform wing subjected to one-dimensional distributed aerodynamic loads (lift, drag, and moment) and in-flight maneuvering (*nw*) loads, as well as concentrated loads applied at two different locations along the wing length. The project details include:

- The wing is limited to a one cell symmetric cross-section composed of a uniform skin and four stringers.

- The *skin* is defined using a single material and constant thickness. The skin shape is a ½ ellipse from the leading edge to the quarter chord, then horizontal flat panels from the quarter-chord (C/4) to the mid-chord (C/2), and finally sloped flat panels from the mid-chord (C/2) to the trailing edge.

- The four *stringers* are defined as two matched pairs (stringers #1 and #2) and (stringers #3 and #4). Both pairs are located in the airfoil horizontal flat region between the quarter-chord (c/4) and the mid-chord. Looking at the first pair of stringers (#1, #2), both stringers have identical section properties and the same location (y_i/c) along the airfoil, where (#1) is on the upper surface and (#2) is on the lower surface. Similarly, the second pair of stringers (#3, #4) have identical section properties and the same location along the airfoil, where (#3) is on the lower surface, and (#4) is on the upper surface. Changing the stringer locations or the stringer section properties using this matched pair approach will ensure that the resulting section properties remain symmetric about the y-axis (i.e. $z_c = I_{yz} = 0$). The materials and cross-section shape for each stringer pair is unique (metallic or composite).

- The *concentrated loads* can be used to simulate point loads acting on wing. For example, a hanging instrument pod, a fuel store, or landing gear. The load location is uniquely specified using (x_i) along the wing length and at the quarter-chord (c/4) along the airfoil. The magnitudes of each load, moment, and torque are MATLAB input variables.



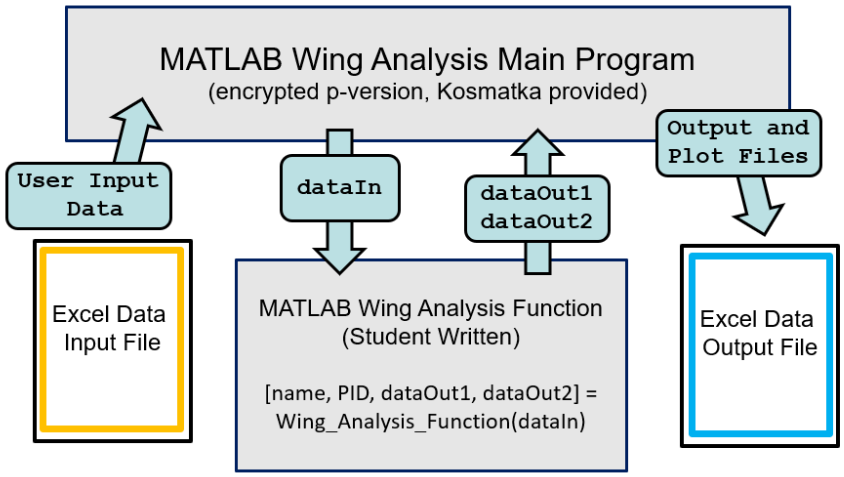
- The *aerodynamic loads (lift, drag, and moment)*, all acting at the quarter chord, are approximated as a 4th order polynomial for lift, rth order polynomial for drag, and a linear varying twisting moment. All of the aerodynamic coefficients are MATLAB input variables.

- The *distributed wing weight* (*nw*) is assumed to act along the mid-chord. The weight coefficients and load factors are MATLAB input parameters.

The project is developed using a Wing Analysis Main Program and a student programmed Wing Analysis Function. See below. The Main Program opens an Excel input file, an Excel output file, and the Wing Analysis Function. The main program, which is written in a MATLAB encrypted p-version, takes care of all of the data input and transferring the input data to the Wing Analysis Function. Next the Wing Analysis Function performs all the data analysis and organizes the data for output and plotting. The Main Program writes the calculated data to the Excel output file, as well as generating all of the needed plots and writing them to specific locations in the

Excel output file. Thus, the student only needs to focus on performing the calculations and not worry about the cumbersome book-keeping effort of reading the input data from Excel and writing/plotting the calculated data to an Excel output file.

Note: All integration (or derivatives) must be performed outside of the MATLAB function and then programmed into the MATLAB function. You cannot use the MATLAB symbolic integration or numerical integration.



The eight goals of this project include the calculation of:

- the modulus-weighted centroid (y_c, z_c) and the modulus-weight cross-section properties (EA, EI_{yy}, EI_{zz}, EI_{yz}) for the stringers only
- the torsion constant (GJ) and the shear center location (y_{sc}, z_{sc}) using the stringers and skin definition.
- the internal load distribution of the wing due to the concentrated loads, the aerodynamic loading, and the maneuvering wing weight loads.
- the axial, shear, bending moment, and torque diagrams.
- the stringer stress distribution (σ_{xx}) and margins of safety (MS)
- the skin shear stress distribution (τ_{xs}) and margins of safety (MS)
- the wing bending displacement and bending slope distribution from the applied concentrated forces, aerodynamic loads, and maneuvering loads.
- the wing twist distribution about the modulus-weighted centroidal axis and the shear center.

FUNCTION INPUT

```
Input Data Array: dataIn (57)

        dataIn(01):     Number of Output Plot Data Points
        dataIn(02):     Wing Length (inch)
        dataIn(03):     Wing Chord (inch)
        dataIn(04):     Maximum Wing Thickness (t_max/c) (%)
        dataIn(05):     Secondary Structure Added Wing Weight (%)

        dataIn(06):     Wing Skin Thickness (inch)
        dataIn(07):     Wing Skin Weight Density (lb/in^3)
        dataIn(08):     Skin Material Shear Modulus (Msi)
        dataIn(09):     Skin Material Yield Strength - Shear (Ksi)
        dataIn(10):     Skin Material Ultimate Strength - Shear (Ksi)

        dataIn(11):     Stringer Pair (#1,#2) y-location (y/c)
        dataIn(12):     Stringer Pair (#1,#2) Cross-Section Area (inch^2)
        dataIn(13):     Stringer Pair (#1,#2) Iyy Inertia about y-axis (inch^4)
        dataIn(14):     Stringer Pair (#1,#2) Izz Inertia about z-axis (inch^4)
        dataIn(15):     Stringer Pair (#1,#2) Iyz Product of Inertia (inch^4)
        dataIn(16):     Stringer Pair (#1,#2) Weight Density (lb/in^3)
        dataIn(17):     Stringer Pair (#1,#2) Material Young’s Modulus (Msi)
        dataIn(18):     Stringer Pair (#1,#2) Yield Strength - Tension (Ksi)
        dataIn(19):     Stringer Pair (#1,#2) Ultimate Strength - Tension (Ksi)
        dataIn(20):     Stringer Pair (#1,#2) Yield Strength - Compression (Ksi)
        dataIn(21):     Stringer Pair (#1,#2) Ultimate Strength - Compression (Ksi)

        dataIn(22):     Stringer Pair (#3,#4) y-location (y/c)
        dataIn(23):     Stringer Pair (#3,#4) Cross-Section Area (inch^2)
        dataIn(24):     Stringer Pair (#3,#4) Iyy Inertia about y-axis (inch^4)
        dataIn(25):     Stringer Pair (#3,#4) Izz Inertia about z-axis (inch^4)
        dataIn(26):     Stringer Pair (#3,#4) Iyz Product of Inertia (inch^4)
        dataIn(27):     Stringer Pair (#3,#4) Weight Density (lb/in^3)
        dataIn(28):     Stringer Pair (#3,#4) Material Young’s Modulus (Msi)
        dataIn(29):     Stringer Pair (#3,#4) Yield Strength - Tension (Ksi)
        dataIn(30):     Stringer Pair (#3,#4) Ultimate Strength - Tension (Ksi)
        dataIn(31):     Stringer Pair (#3,#4) Yield Strength - Compression (Ksi)
        dataIn(32):     Stringer Pair (#3,#4) Ultimate Strength - Compression (Ksi)

        dataIn(33):     Safety Factor - Yield
        dataIn(34):     Safety Factor - Ultimate

        dataIn(35):     First Load Location (x/L)
        dataIn(36):     Concentrated Force - X Direction (lb)
        dataIn(37):     Concentrated Force - Y Direction (lb)
        dataIn(38):     Concentrated Force - Z Direction (lb)
        dataIn(39):     Concentrated Torque - About X Direction (lb-in)
        dataIn(40):     Concentrated Moment - About Y Direction (lb-in)
        dataIn(41):     Concentrated Moment - About Z Direction (lb-in)
        dataIn(42):     Second Load Location (x/L)
        dataIn(43):     Concentrated Force - X Direction (lb)
        dataIn(44):     Concentrated Force - Y Direction (lb)
        dataIn(45):     Concentrated Force - Z Direction (lb)
        dataIn(46):     Concentrated Torque - About X Direction (lb-in)
```

```
  dataIn(47):     Concentrated Moment - About Y Direction (lb-in)
  dataIn(48):     Concentrated Moment - About Z Direction (lb-in)

  dataIn(49):     Aircraft Load Factor
  dataIn(50):     Drag Distribution - Constant (lb/in)
  dataIn(51):     Drag Distribution - rth order (lb/in)
  dataIn(52):     Drag Distribution - polynomial order
  dataIn(53):     Lift Distribution - Constant (lb/in)
  dataIn(54):     Lift Distribution - 2nd Order (lb/in)
  dataIn(55):     Lift Distribution - 4th Order (lb/in)
  dataIn(56):     Twist Moment Distribution - Constant (lb-in/in)
  dataIn(57):     Twist Moment Distribution - 1st Order (lb-in/in)
```

FUNCTION OUTPUT

```
Name:             Name of student author

PID:              UCSD Student ID number

dataOut1(51):     Packed calculated output variable data
  dataOut1(01):   Modulus Weighted Centroid y-direction (inch)
  dataOut1(02):   Modulus Weighted Centroid z-direction (inch)
  dataOut1(03):   Cross-Section Weight rhoA (lb/in)
  dataOut1(04):   Axial   Stiffness EA   (lb)
  dataOut1(05):   Bending Stiffness EIyy (lb-in^2)
  dataOut1(06):   Bending Stiffness EIzz (lb-in^2)
  dataOut1(07):   Bending Stiffness EIyz (lb-in^2)
  dataOut1(08):   Torsion Stiffness GJ   (lb-in^2)
  dataOut1(09):   Shear Center, y-direction (inch)
  dataOut1(10):   Shear Center, z-direction (inch)
  dataOut1(11):   Total Half-Span Wing Weight including added weight factor (lb)

  dataOut1(12):   Root Internal Force - X Direction (lb)
  dataOut1(13):   Root Internal Force - Y Direction (lb)
  dataOut1(14):   Root Internal Force - Z Direction (lb)
  dataOut1(15):   Root Internal Moment - about X Direction (lb-in)
  dataOut1(16):   Root Internal Moment - about Y Direction (lb-in)
  dataOut1(17):   Root Internal Moment - about Z Direction (lb-in)

  dataOut1(18):   Stringer (#1) Calculated Axial Stress (lb/in^2)
  dataOut1(19):   Stringer (#1) Allowable Stress - Tension (lb/in^2)
  dataOut1(20):   Stringer (#1) Allowable Stress - Compression (lb/in^2)
  dataOut1(21):   Stringer (#1) Margin of Safety

  dataOut1(22):   Stringer (#2) Calculated Axial Stress (lb/in^2)
  dataOut1(23):   Stringer (#2) Allowable Stress - Tension (lb/in^2)
  dataOut1(24):   Stringer (#2) Allowable Stress - Compression (lb/in^2)
  dataOut1(25):   Stringer (#2) Margin of Safety

  dataOut1(26):   Stringer (#3) Calculated Axial Stress (lb/in^2)
  dataOut1(27):   Stringer (#3) Allowable Stress - Tension (lb/in^2)
  dataOut1(28):   Stringer (#3) Allowable Stress - Compression (lb/in^2)
  dataOut1(29):   Stringer (#3) Margin of Safety
```

```
    dataOut1(30):   Stringer (#4) Calculated Axial Stress (lb/in^2)
    dataOut1(31):   Stringer (#4) Allowable Stress - Tension (lb/in^2)
    dataOut1(32):   Stringer (#4) Allowable Stress - Compression (lb/in^2)
    dataOut1(33):   Stringer (#4) Margin of Safety

    dataOut1(34):   Skin Panel (1.2) Calculated Shear Stress (lb/in^2)
    dataOut1(35):   Skin Panel (1.2) Allowable Stress - Shear (lb/in^2)
    dataOut1(36):   Skin Panel (1.2) Margin of Safety
    dataOut1(37):   Skin Panel (2.3) Calculated Shear Stress (lb/in^2)
    dataOut1(38):   Skin Panel (2.3) Allowable Stress - Shear (lb/in^2)
    dataOut1(39):   Skin Panel (2.3) Margin of Safety
    dataOut1(40):   Skin Panel (3.4) Calculated Shear Stress (lb/in^2)
    dataOut1(41):   Skin Panel (3.4) Allowable Stress - Shear (lb/in^2)
    dataOut1(42):   Skin Panel (3.4) Margin of Safety
    dataOut1(43):   Skin Panel (4.1) Calculated Shear Stress (lb/in^2)
    dataOut1(44):   Skin Panel (4.1) Allowable Stress - Shear (lb/in^2)
    dataOut1(45):   Skin Panel (4.1) Margin of Safety

    dataOut1(46):   Tip Displacement  - X Direction (inch)
    dataOut1(47):   Tip Displacement  - Y Direction (inch)
    dataOut1(48):   Tip Displacement  - Z Direction (inch)
    dataOut1(49):   Tip Twist (degree)
    dataOut1(50):   Tip Bending Slope (dv/dx) (inch/inch)
    dataOut1(51):   Tip Bending Slope (dw/dx) (inch/inch)

 dataOut2(nplot,27):  Packed calculated output plot data
    column( 1):     X direction coordinate (inch)
    column( 2):     Applied distributed drag force (lb/in)
    column( 3):     Applied distributed lift force (lb/in)
    column( 4):     Applied distributed torque (lb-in/in)
    column( 5):     Internal shear force  -   Vx (lb)
    column( 6):     Internal shear force  -   Vy (lb)
    column( 7):     Internal shear force  -   Vz (lb)
    column( 8):     Internal axial torque -   Mx (lb-in)
    column( 9):     Internal bending moment - My (lb-in)
    column(10):     Internal bending moment - Mz (lb-in)
    column(11):     Stringer (#1) Axial Stress (lb/in^2)
    column(12):     Stringer (#2) Axial Stress (lb/in^2)
    column(13):     Stringer (#3) Axial Stress (lb/in^2)
    column(14):     Stringer (#4) Axial Stress (lb/in^2)
    column(15):     Skin Panel (1.2) Shear Stress (lb/in^2)
    column(16):     Skin Panel (2.3) Shear Stress (lb/in^2)
    column(17):     Skin Panel (3.4) Shear Stress (lb/in^2)
    column(18):     Skin Panel (4.1) Shear Stress (lb/in^2)
    column(19):     Displacement - X Direction (inch)
    column(20):     Displacement - Y Direction (inch)
    column(21):     Displacement - z Direction (inch)
    column(22):     Total Twist (Twsit1 + Twist2 + Twist3)  (degree)
    column(23):     Bending Slope (dv/dx) (inch/inch)
    column(24):     Bending Slope (dw/dx) (inch/inch)
    column(25):     Twist1 (Twist from Vy only)  (degree)
    column(26):     Twist2 (Twist from Vz only)  (degree)
    column(27):     Twist3 (Twist from Mx only)  (degree)
```

APPENDIX A:

Running the MATLAB P-code with YOUR Function

1) In your MATLAB directory include the following files:

```
SE160A_2_Wing_Analysis_Input.xlsx            Excel Input File
SE160A_2_Wing_Analysis_Output.xlsx          Excel Output File
Wing_Analysis_Main.p                   Project Main Code p-code
Wing_Analysis_Function.m       (function that you will write)
```

2) Open MATLAB in that directory
3) Make sure both the Excel input and output files are closed
4) Run the p-code. Do not try and open a (*.p) code

```
>> run Wing_Analysis_Main.p
```

The main code will open the Excel input file, your Wing_Analysis_Function, and the Excel output file. Then the input data is passed to the function for calculation, followed by the calculated data being returned to the main program. Finally, the main program writes the data and plots to the Excel output file.

5) Open the Excel output file and review results.

Note: It is recommended that you start with the Wing_Analysis_Function (student skeleton).m. This example function has lots of useful code.

Note: If you want to check your results, replace your Wing_Analysis_Function.m with the provided Wing_Analysis_Function.p, which was provided in your zip folder. Using the above command, run the main program, which will now call the p-version function. The correct answers are written to the output file. Save the output file for comparing with your function results.

DESIGN STUDY

Now let's use your MATLAB code to perform a preliminary structural analysis on a single-cell wing of length (L = 240 inch) and chord (c = 48 inch), where we will look at the approximate cruise condition and the four corners of the V-n diagram (see Table 1) for both an aluminum and a composite wing design. You will use your code to determine the minimum stringer cross-section area and skin thickness for each wing design, so that the effective weight savings can be determined.

__*Note:*__ ***In order to receive full credit (200 points) for the design study, you must use your own code. If you use the professor supplied p-code or another student's code, then you will be graded on a 100-point scale. In addition, you must state in your write-up that "the calculation results were obtained using a provided code." It would be clear academic integrity violation to use results from another's code and trying to pass it off as your own results. If this is the case, you will receive a zero for the project and your case will be forwarded to the UCSD Academic Integrity Department.***

Table 1: Load case definition for cruise and the four corners of V-n diagram

		Load Cases					
Variable	**Description**	***Cruise***	***PHAA***	***PLAA***	***NHAA***	***NLAA***	
n	Load factor	1	3.8	3.8	-1.5	-1.5	
p_{yo}	Drag distribution (constant)	1.8	2.4	60	2.4	60	lb/inch
p_{yr}	Drag distribution (rth order)	0.4	0.4	6	0.4	6	lb/inch
r	Drag distribution (polynomial order)	10	10	10	10	10	
p_{zo}	Lift distribution (constant)	15	57	57	-22.5	-22.5	lb/inch
p_{z2}	Lift distribution (2nd order)	-4	-15.2	-15.2	6	6	lb/inch
p_{z4}	Lift distribution (4th order)	-1	-3.8	-3.8	1.5	1.5	lb/inch
m_{xo}	Twist moment distribution	50	15	200	15	200	lb-inch/inch
m_{x1}	Twist moment distribution (linear)	0	0	0	0	0	lb-inch/inch

Table 2: Aluminum and Graphite/Epoxy Properties

Property	Al 7075-T6	Carbon/epoxy	Units
ρ	0.10	0.056	lb./in^3
E	10	23.5	Msi
G	3.75	6	Msi
σ_{Ty}	37	263	Ksi
σ_{Tu}	43	263	Ksi
σ_{Cy}	-37	-263	Ksi
σ_{Cu}	-43	-263	Ksi
τ_y	24	88	Ksi
τ_u	28	88	Ksi

Step 1: Preliminary structural analysis of an aluminum wing

Perform a structural analysis of a single-cell aluminum wing for cruise and the four corners of the *V-n* diagram. The input data file is given at the end of this project statement. During this preliminary study, ignore stringer section properties (I_{yy}, I_{zz}, I_{yz}) and assume all four stringers have the same properties. Using your results from the five analysis cases, fill out the following table and calculate the total wing weight. Use this total wing weight in your study.

	units	Cruise	PHAA	PLAA	NHAA	NLAA
MS - stringer (minimum)						
MS – skin (minimum)						
Tip vertical displacement	(inch)					
Tip twist	(degree)					
Wing weight	(lb)					

Note: (1) the minimum stringer margin of safety (MS) is defined as the minimum value observed for all four stringers at the root (x = 0), (2) the minimum skin MS is defined as the minimum value observed for all four skin sections at the root (x = 0), and (3) the calculated wing weight is determined using:

$$W = (S\rho_{skin}t_{skin} + \sum_{i=1}^{4}\rho_i A_i)L\left(1+K_w\right)$$

where;

S	=	wing cross-section perimeter
ρ_{skin}	=	wing skin material density
t_{skin}	=	skin thickness
ρ_{i}	=	wing stringer material density
A_{i}	=	i^{th} stringer area
L	=	wing length (= b/2)
K_{w}	=	Additional wing weight from secondary or tertiary structures (typically 10-20%)

Note: It is recommended that you use the above equation to calculate the wing weight.

Step 2: Preliminary structural analysis of a carbon/epoxy wing

Perform a structural analysis of a single-cell carbon/epoxy wing for cruise and the four corners of the V-n diagram. The input data file is given at the end of this project statement. During this preliminary study, ignore stringer section properties (I_{yy}, I_{zz}, I_{yz}) and assume all four stringers have the same properties. Using your results from the five analysis cases, fill out the following table and calculate the wing weight assuming the density of carbon/epoxy is (ρ = 0.056 lb/in^3). If the displacements are unrealistically large (> 15 inch) , you may have to increase stringer areas.

	units	Cruise	PHAA	PLAA	NHAA	NLAA
MS - stringer (minimum)						
MS – skin (minimum)						
Tip vertical displacement	(inch)					
Tip twist	(degree)					
Wing weight	(lb)					

Step 3: Design analysis of an aluminum wing

Using the results from the table in Step 1, determine the minimum stringer areas (A_s) and minimum skin thickness (t_s) so that all the margins are safety for all five analysis cases are greater than or equal to zero. Rerun the five analysis cases from step (1) with the new section properties and fill out the table. (Hint: All MS > 0)

Stringer (#1) Area (minimum)	(inch^2)	
Stringer (#2) Area (minimum)	(inch^2)	
Stringer (#3) Area (minimum)	(inch^2)	
Stringer (#4) Area (minimum)	(inch^2)	
Skin Thickness (minimum)	(inch)	

	Units	Cruise	PHAA	PLAA	NHAA	NLAA
MS - stringer (minimum)						
MS – skin (minimum)						
Tip vertical displacement	(inch)					
Tip twist	(degree)					
Minimum Aluminum Wing Weight	(lb)					

Step 4: Design analysis of a carbon/epoxy wing

Using the results from the table in Step 2, determine the minimum composite stringer area (A_s) and minimum composite skin thickness (t_s) so that all the margins are safety for all five load cases are greater than or equal to zero. Rerun the five analysis cases from step 2 with the new section properties and fill out the table.

Stringer (#1) Area (minimum)	(inch^2)	
Stringer (#2) Area (minimum)	(inch^2)	
Stringer (#3) Area (minimum)	(inch^2)	
Stringer (#4) Area (minimum)	(inch^2)	
Skin Thickness (minimum)	(inch)	

	Units	Cruise	PHAA	PLAA	NHAA	NLAA
MS - stringer (minimum)						
MS – skin (minimum)						
Tip vertical displacement	(inch)					
Tip twist	(degree)					
Minimum Carbon/epoxy Wing Weight	(lb)					

Step 5: Comment on the weight savings between the aluminum wing and the composite wing.