Project (#5): Wing Bending, Torsion, and Shear Analysis
Due Date: Upload zip folder to TED by 11:58 PM, Friday March 13, 2020

Files in your MATLAB Folder:

Download from TED:	**SE160A_5_Wing_Bending_Input.xlsx**	
Download from TED:	**SE160A_5_Wing_Bending_Output.xlsx**	
Download from TED:	**SE160A_5_Wing_Bending.p**	
Your created (m) file:	**For Undergraduate Students:**	**[SE160A_5_LastName_FirstName.m]**
	For Graduate Students:	**[SE260A_5_LastName_FirstName.m]**

Problem Answers are saved in a (pdf):
For Undergraduate Students: [SE160A_5_LastName_FirstName.pdf]
For Graduate Students: [SE260A_5_LastName_FirstName.pdf]

Upload your (m) file and (pdf) file into a (zip) folder of the same name:
For Undergraduate Students: [SE160A_5_LastName_FirstName.zip]
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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 uniform wing subjected to one-dimensional aerodynamic loads (lift, drag, and moment) and in-flight maneuvering loads. The wing is limited to a one cell symmetric cross-section with a maximum of 4 stringers, where the stringers are located at the leading and trailing edges, and one stringer is located symmetrically above and below the chord-line. The materials for each stringer and skin panel are unique (metallic or composite) and the resulting cross-section is structurally symmetric ($EI_{yz} = 0$). The *aerodynamic loads*, all acting at the quarter chord, are approximated as a 4^{th} order polynomial for lift, n^{th} order polynomial for drag, and a constant twisting moment. In addition, a distributed wing weight is defined that acts along the mid-chord. (See figure below). The aerodynamic and weight polynomial coefficients are input parameters.

The five goals of this project include the calculation of:

- the cross-section section properties (stringers only), torsion constant (skin only), and shear center location
- the internal load distribution of the wing due to aerodynamic and maneuvering wing weight loads.
- 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 twisting distributions.

INPUT (See Excel Input File for format restrictions)

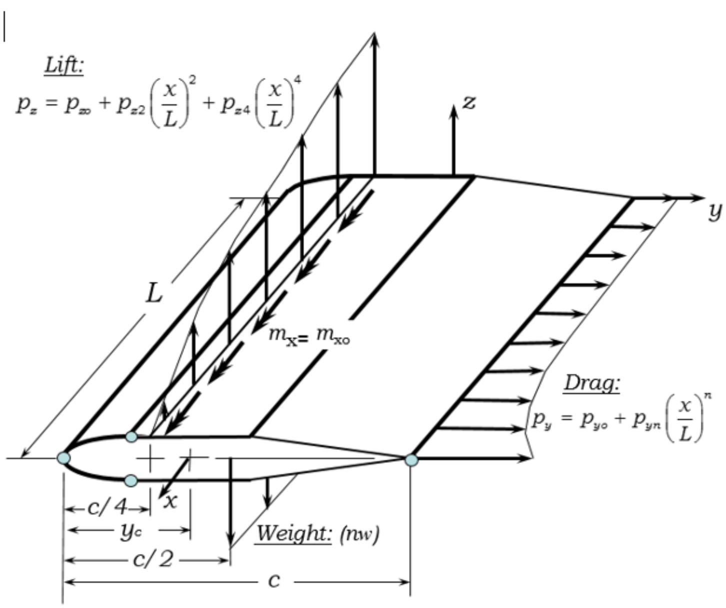
Title	Title of project
Wing Stringer Definition	The (y,z) location for the (4) stringers within the cross-section. The origin can be any arbitrary point. Each stringer is defined in terms of geometric and material properties.
Wing Skin Definition	The thickness and material details for the four skin panels that form the airfoil section. The two leading skins that connect stringers (1 and 2) and (1 and 4) form an elliptical shape, whereas the two remaining skins connect stringers (2 and 3) and (3 and 4) and include a horizontal portion to the mid-chord and a straight section to the trailing edge.

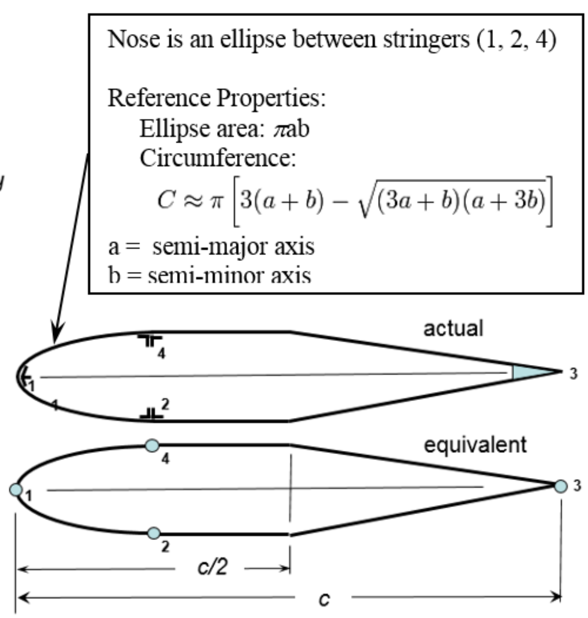
Wing Size and Weight	The wing length (1/2 wing span) along with the distributed weight ($w = w_o$ lb/in) assumed to act along mid-chord are given. In addition, the maneuvering load factor (n) along with the factor of safety definitions for all materials are defined.
Wing Aero Definition	The polynomial coefficients for the aerodynamic lift, drag, and moment are provided.

- Lift distribution: $p_z = p_{zo} + p_{z2}(x/L)^2 + p_{z4}(x/L)^4$ lb/in acting at ($c/4$)
- Drag distribution: $p_y = p_{yo} + p_{yn}(x/L)^n$ lb/in acting at ($c/4$)
- Aerodynamic moment $m_x = m_{xo}$ lb-in/in acting about ($c/4$)

OUTPUT (See Excel Input File for format restrictions)

Student Name, ID	Name of Student and Student UCSD ID number
Title, Echo	Title of project, Echo all input data
Cross-Section Properties	Determine the modulus weighted centroid and then determine the section properties about the modulus weighted centroid. Calculate the torsion constant (GJ). Calculate the shear center location (e_y, e_z) measured from the modulus-weighted centroid.
Internal Stress Resultants	Determine the six stress resultants at the wing root (x = 0). These six stress resultant functions are higher order polynomials that need to be derived.
Stringer Stress Analysis	Using the cross-section properties, and the root stress resultants, determine the root (x = 0) axial stress in each stringer. Determine the margin of safety.
Skin Stress Analysis	Using the skin thickness, and the root stress resultants, determine the root (x = 0) shear stress in each skin panel. Determine the margin of safety.
Tip Displacement/Twist	Using the complete stress resultant functions along with the material section properties, carry out the integration to find the wing tip displacement and twist.





ANALYTICAL STUDY (Approximately 1-hour)

Now let's use your MATLAB code to perform a preliminary structural analysis on a single-cell wing, where we look at the approximate cruise condition and 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. We will limit our design studies to insure cross-sections remain symmetric ($EI_{yz} = z_c = 0$). Thus, any change you recommend to a skin thickness or stringer cross-section area will be made to all stringers and all skins. Obviously, further weight savings can be obtained if the individual stringers or skins were unique but this would lead to an unsymmetric cross-section and a more complicated programming exercise.

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

	Load Cases					
	Cruise	***PHAA***	***PLAA***	***NHAA***	***NLAA***	
Wing Weight (w)	3	3	3	3	3	lb/inch
Load Factor (n)	1	3.8	3.8	-1.5	-1.5	
Lift Distribution (pzo)	15	57	57	-22.5	-22.5	lb/in
Lift Distribution (pz2)	-4	-15.2	-15.2	6	6	lb/in
Lift Distribution (pz4)	-1	-3.8	-3.8	1.5	1.5	lb/in
Drag Distribution at Root (pyo)	1.8	2.4	60	2.4	60	lb/in
Drag Distribution coefficient (pyn)	0.4	0.4	6	0.4	6	lb/in
Drag Distribution polynomial order (n)	10	10	10	10	10	
Aerodynamic moment (mxo)	50	15	200	15	200	lb-in/in

Table 2: Aluminum and Graphite/Epoxy Properties

Property	Al 7075-&6	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}). Using your results from the five analysis cases, fill out the following table and calculate the wing weight.

	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:

$$W = \rho L(4A_s + St_s)$$

where;

- ρ = wing material density (for Aluminum, ρ = 0.1 lb/in^3)
- L = wing length (250 inch)
- A_s = stringer area (inch^2)
- S = wing cross-section perimeter (98.49 inch)
- t_s = skin thickness (inch)

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. Neglect stringer section 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).

	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 area (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 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 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.

MATLAB Project (#5) - Wing Bending and Torsion Analysis

SE-160A Aerospace Structural Analysis, University of California, San Diego (Copyright J.B. Kosmatka, 2020)

Version: **Winter, 2020 (v1)**

Project Title: **Design Study Step 1: Single Cell Aluminum (6061-T6) Wing. Flight Condition: Cruise**

Variable	Description	Value	Units
iInput	Input Units	1	1 = US, 2 = SI
iOutput	Output Units	1	1 = US, 2 = SI

Units Reference		
	US	*SI*
σ, τ	10^3 lb/in^2	*MPa*
E, G	10^6 lb/in^2	*GPa*

Stringer Definition

Variable	Description	(#1)	(#2)	(#3)	(#4)	Units
y	y-location	**0**	**16**	**48**	**16**	inch
z	z-location	**0**	**-3.6**	**0**	**3.6**	inch
A_s	Stringer Area	**0.2**	**0.2**	**0.2**	**0.2**	inch^2
I_{yy}	Inertia about y-axis	**0**	**0**	**0**	**0**	inch^4
I_{zz}	Inertia about z-axis	**0**	**0**	**0**	**0**	inch^4
I_{yz}	Inertia about z-axis	**0**	**0**	**0**	**0**	inch^4
E	Young's Modulus	**10**	**10**	**10**	**10**	Msi
σ_{yT}	Yield strength - tension	**37**	**37**	**37**	**37**	Ksi
σ_{uT}	Ult strength - tension	**43**	**43**	**43**	**43**	Ksi
σ_{yC}	Yield strength - compress	**-37**	**-37**	**-37**	**-37**	Ksi
σ_{uC}	Ult strength - compression	**-43**	**-43**	**-43**	**-43**	Ksi

Skin Definition

Variable	Description	1.2	2.3	3.4	4.1	Units
t	Skin Thickness	**0.05**	**0.05**	**0.05**	**0.05**	inch
G	Shear Modulus	**3.75**	**3.75**	**3.75**	**3.75**	Msi
τ_y	Yield strength - shear	**24**	**24**	**24**	**24**	Ksi
τ_u	Ultimate strength - shear	**28**	**28**	**28**	**28**	Ksi

Wing Geometry, Weight, Load Factor, and Safety Factor

Variable	Description	Value	Units
L	Wing length (b/2)	**250**	inch
c	Wing chord	**48**	inch
w_o	Wing weight	**3**	lb/inch
n	Load factor	**1**	1
SF_y	Safety Factor - yield	**1.1**	1
SF_u	Safety Factor - ultimate	**1.5**	1

Wing Aerodynamic Definition

Variable	Description	Value	Units
p_{zo}	Lift Distribution - constant	**15**	lb/in
p_{z2}	Lift Distribution - 2nd Order	**-4**	lb/in
p_{z4}	Lift Distribution - 4th Order	**-1**	lb/in
p_{yo}	Drag Distribution - constant	**1.8**	lb/in
p_{yn}	Drag Distribution - nth Order	**0.4**	lb/in
n	Drag - Polynomial Order	**10**	1
m_{xo}	Distributed Moment	**50**	lb-in/in

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MATLAB Project (#5) - Wing Bending and Torsion Analysis

SE-160A Aerospace Structural Analysis, University of California, San Diego (Copyright J.B. Kosmatka, 2020)

Version: Winter, 2020 (v1)

Project Title: Design Study Step 2: Single Cell Carbon/Epoxy Wing. Flight Condition: Cruise

Variable	Description	Value	Units
iInput	Input Units	1	1 = US, 2 = SI
iOutput	Output Units	1	1 = US, 2 = SI

	Units Reference	
	US	*SI*
σ, τ	10^3 lb/in^2	*MPa*
E, G	10^6 lb/in^2	*GPa*

Stringer Definition

Variable	Description	(#1)	(#2)	(#3)	(#4)	Units
y	y-location	**0**	**16**	**48**	**16**	inch
z	z-location	**0**	**-3.6**	**0**	**3.6**	inch
A_s	Stringer Area	**0.2**	**0.2**	**0.2**	**0.2**	inch^2
I_{yy}	Inertia about y-axis	**0**	**0**	**0**	**0**	inch^4
I_{zz}	Inertia about z-axis	**0**	**0**	**0**	**0**	inch^4
I_{yz}	Inertia about z-axis	**0**	**0**	**0**	**0**	inch^4
E	Young's Modulus	**23.5**	**23.5**	**23.5**	**23.5**	Msi
σ_{yT}	Yield strength - tension	**263**	**263**	**263**	**263**	Ksi
σ_{uT}	Ult strength - tension	**263**	**263**	**263**	**263**	Ksi
σ_{yC}	Yield strength - compress	**-263**	**-263**	**-263**	**-263**	Ksi
σ_{uC}	Ult strength - compression	**-263**	**-263**	**-263**	**-263**	Ksi

Skin Definition

Variable	Description	1.2	2.3	3.4	4.1	Units
t	Skin Thickness	**0.05**	**0.05**	**0.05**	**0.05**	inch
G	Shear Modulus	**6**	**6**	**6**	**6**	Msi
τ_y	Yield strength - shear	**88**	**88**	**88**	**88**	Ksi
τ_u	Ultimate strength - shear	**88**	**88**	**88**	**88**	Ksi

Wing Geometry, Weight, Load Factor, and Safety Factor

Variable	Description	Value	Units
L	Wing length (b/2)	**250**	inch
c	Wing chord	**48**	inch
w_o	Wing weight	**3**	lb/inch
n	Load factor	**1**	1
SF_y	Safety Factor - yield	**1.1**	1
SF_u	Safety Factor - ultimate	**1.5**	1

Wing Aerodynamic Definition

Variable	Description	Value	Units
p_{zo}	Lift Distribution - constant	**15**	lb/in
p_{z2}	Lift Distribution - 2nd Order	**-4**	lb/in
p_{z4}	Lift Distribution - 4th Order	**-1**	lb/in
p_{yo}	Drag Distribution - constant	**1.8**	lb/in
p_{yn}	Drag Distribution - nth Order	**0.4**	lb/in
n	Drag - Polynomial Order	**10**	1
m_{xo}	Distributed Moment	**50**	lb-in/in

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