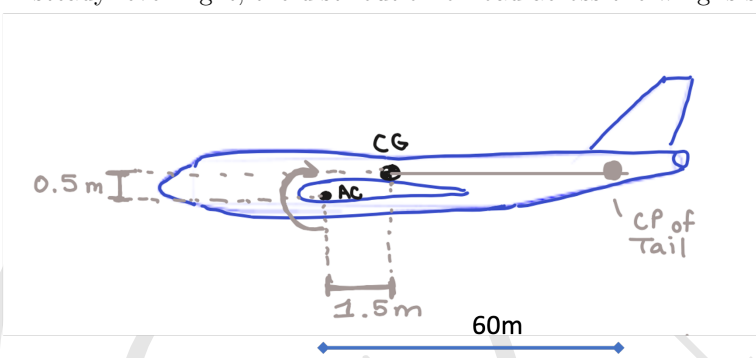
FINAL EXAM

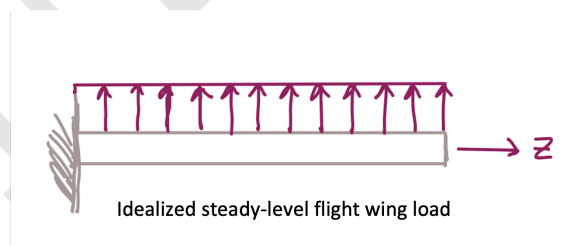
MEGN 498A

Due May 9, 2022 by 11:59pm Online

General Instructions - Read Carefully
The final must be done solely by the student submitting. Please write the following Honor Code at the top of your PDF online submission and write out your name (neatly), sign and date: "I have neither given nor received aid on this exam." Failing to perform this step will result in a zero on your final. You may only use what have for this course (and no other courses) on this exam, which includes anything provided on the course Canvas page, the course textbook and your own notes from this course. You may use the aid of mathematical software for this exam if you choose, please print out and include all your work if you do that. You must show all steps neatly and correctly to receive full points on these problems and please box in your final answers and use correct units where appropriate. GOOD LUCK!

Given a commercial aircraft with the following properties (see figure below) flying at 286 m/s equivalent airspeed, with wing area of 554 m^2 and mean chord length of 8 m, and total aircraft weight of 3,200,000 N. The drag coefficient of the entire aircraft is given by $C_D = 0.02 + 0.04C_L^2$ and the lift-curve slope of the wings is 4.2 and of the the tail (which has an area of 60 m^2) of 2.5. The pitching moment coefficient about the aerodynamic center (of the complete aircraft minus the tailplane) based on wing area is -0.02. Assume in steady level flight, the distribution of load across the wing is shown in the image below.





Question 1: *15 points*
Assume the fuselage during steady level flight is a thin-walled pressurized vessel with the stresses in the x and y direction as: $\sigma_x = 50$ MPa, $\sigma_y = -20$ MPa and $\tau_{xy} = -10$ MPa. Answer the following:

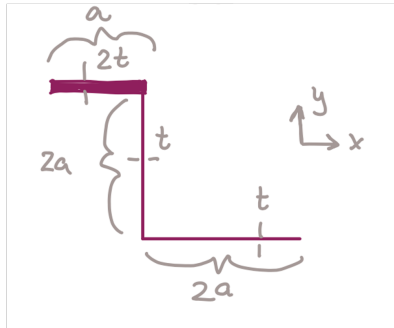
(a) Find the principal stresses and maximum shear stress in the fuselage (assume 2D here).

(b) Now for simplicity, assume the other stresses are zero and take those principal values and maximum shear stress you calculated above as a starting point overall stress state, assume a temperature change of –60°C, which is the average temperature change from runway to 30,000 feet (9 km), and calculate the corresponding strain tensor in the fuselage assuming it is made of aluminum, using the mean values from your table of material systems on page 39 of your course notes.

(c) Now assume the same starting stress state from the previous part is acting on a composite fuselage orthotropic single ply material piece made from an epoxy resin and carbon fibers with moduli equal to 5,000 N/mm^2 and 200,000 N/mm^2 and major Poisson's of 0.2 and 0.3 and a volume fraction of fibers as 43%, respectively. Taking a shear modulus value of 5,000 N/mm^2, and knowing that the fibers are at a 45 degree angle with respect to the global axis, calculate the stresses and strains that would be seen in ply direction (i.e. longitudinal and transverse).

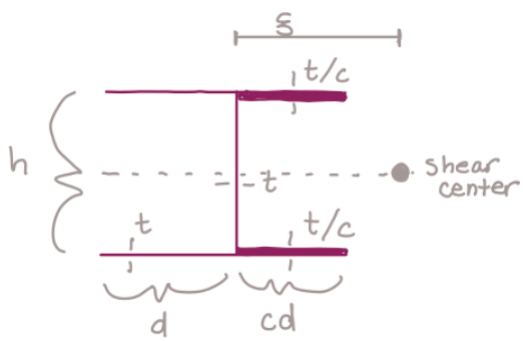
Question 2:. *12 points*
Calculate the changes in lift, and tail load, the resultant load factor and the forward inertial force should this aircraft go from a steady glide with zero thrust to an equivalent sharp-edge downgust of 10 m/s equivalent airspeed.

Question 3:. *10 points*
Assume the main spar on the wing takes all the load and is a Z-beam cross-section shown below. Find the following:



(a) State where along the length of the wing (i.e. beam z-direction) for the steady-level flight condition the bending moment will be maximum and why.
(b) Assuming just an M_x is loading the main spar, no M_y, find where in the bending stress will be maximum on the cross-section. Show all work.

Question 4:. *15 points*
Now assume the main spar on the wing has the following I-beam like geometry and find the following:



(a) State where along the length of the wing (i.e. beam z-direction) for the steady-level flight condition the shear force will be maximum and why.
(b) Assuming just an S_y, no S_x, where is shear flow maximum on the cross-section if in this case h = 100 mm, d= 50 mm, c = 0.5 and t = 2 mm?
(c) Assuming just an S_y is loading on the main spar, no S_x, show that the distance of the shear center from the vertical web is given by: $\frac{\xi}{d} = \frac{3P(1-c)}{1+12P}$ where $P = d/h$.

Question 5:. *13 points*
Now assume the main spar on the wing has the L-beam like geometry and find the following:

Compared to an L-beam of all the same aluminum as the fuselage with L = 500 mm and t = 5mm, and given an M_x of 10 kN-m, no M_y, create an isotropic, homogeneous, composite beam of your choosing with reasonable, but different materials for the web and flange on the L-beam and show that you have improved the design in bending from the all-aluminum original. You may not change geometry. What are some potential benefits of the L-beam structural composite design you created? What are some potential drawbacks of the L-beam structural composite design you created? List 2 each.