**Name:**

**Enrolment No:**

**UPES**

**End Semester Examination, May 2023**

**Program Name: B.TECH-Aerospace Engineering** | **Semester : IV**
**Course Name : Aerodynamics-I** | **Time : 03 hrs.**
**Course Code : ASEG2006** | **Max. Marks: 100**
**Nos. of page(s) : 03**

**Instructions: Attempt All Questions. Assume any missing data if required.**

## SECTION A
**(5Qx4M=20Marks)**

| S. No. | Statement of question | Marks | CO |
|---|---|---|---|
| Q1 | Many years ago, *Scientific American* published a letter about the aerodynamics of pollen spores. A photograph accompanied the letter showing a spore having a diameter of about 20 μm and looking remarkably like a golf ball. The gist of the letter was that nature had discovered the principle of golf-ball aerodynamics millions of years before man. Explain why the letter writer's logic is faulty. | **4** | **CO 1** |
| Q2 | Consider a hypothetical hypersonic re-entry vehicle returning to earth's surface. Let's focus our attention at a region very close to the vehicle surface. Assuming the vehicle surface temperature to be $T_s$ and the ambient temperature to be $T_a$ at a particular height, plot the approximate variation of temperatures in a region very close to the vehicle surface ($T$ vs. $y$). Provide justifications in few lines for the plot you have drawn. (1-3 lines maximum) | **4** | **CO 2** |
| Q3 | In the 1930s-40s, NACA made a systematic set of measurements of drag and lift characteristics for a variety of airfoils. It is interesting to note how these experiments were conducted. For instance, note the two points below:<br>a) Wingtips were butted against both the sidewalls of the wind tunnel.<br>b) No conventional force balance was used to measure lift or drag.<br>Explain the significance of the two above statements as it pertains to the NACA experiments (1-3 lines maximum for each). | | **CO 1** |
| Q4 | Panel methods are an effective tool to numerically simulate the flow around airfoils and other bodies. In particular, we have discussed the source panel method and the vortex panel methods for the same. Two questions arise:<br>a) Given the source panel method, what is the need for vortex panel method? (1-2 lines)<br>b) Are vortex panel methods prone to inaccuracy? Why or why not ? (1-2 lines) | **4** | **CO 2** |
| Q5 | Given NACA4421 and NACA 4412 airfoils – which according to you will have a greater maximum lift coefficient? Can you explain the reason behind the same? | **4** | **CO 1** |

## SECTION B
**(4Qx10M= 40 Marks)**

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|---|---|---|---|
| Q6 | For the NACA 2412 airfoil, the lift coefficient and moment coefficient about the quarter-chord at $-6^\circ$ angle of attack are $-0.39$ and $-0.045$, respectively. At $4^\circ$ angle of attack, these coefficients are 0.65 and $-0.037$, respectively. Calculate the location of the aerodynamic center. | **10** | **CO 2** |
| Q7 | Consider again the NACA 2412 airfoil discussed in Q6. The airfoil is flying at a velocity of 60 m/s at a standard conditions. The chord length of the airfoil is 2 m. Calculate the lift per unit span when the angle of attack is $4^\circ$. Calculate the value of the circulation around the airfoil. | **10** | **CO 2** |
| Q8 | Consider the twin-jet executive transport - for this airplane the zero-lift angle of attack is $-2^\circ$, the lift slope of the airfoil section is 0.1 per degree, the lift efficiency factor $\tau = 0.04$, and the wing aspect ratio is 7.96. Given that the lift coefficient is identical to the prediction made by thin airfoil theory for a symmetrical airfoil at an AoA of 5 degrees, calculate the angle of attack of the twin-jet executive airplane. | **10** | **CO 3** |
| Q9 | The measured lift slope for the NACA 23012 airfoil is 0.1080 degree$^{-1}$,and $\alpha L$=0 = $-1.3^\circ$. Consider a finite wing using this airfoil, with AR = 8 and taper ratio = 0.8. Assume that $\delta = \tau$ . Calculate the lift and induced drag coefficients for this wing at a geometric angle of attack = $7^\circ$. | **10** | **CO 4** |
| **SECTION-C**<br>**(2Qx20M=40 Marks)** | | | |
| Q10 | Consider a lifting flow over a circular cylinder of a given radius with uniform velocity $V_\infty$, obtain the equation for stream function, stagnation points and prove that $L^{`} = \rho_\infty V_\infty \Gamma$<br><br>**OR**<br><br>Consider the non-lifting flow over a circular cylinder. Derive an expression for the pressure coefficient at an arbitrary point *(r, θ)* in this flow. Next, show its generalization on the cylinder surface. | **20** | **CO 3** |
| Q11 | Consider a hypothetical airfoil, for which the mean camber line be defined and the derivative of the camber line, *z'(x)* when transformed into polar coordinates using *x/c* = 0.5(1-cos θ) gives:<br><br>*z/c* = *f* (*x/c*) ; *z'(x)* = (1+ cos θ ) for $0 \le x/c \le 0.25$<br>and, *z/c* = *g* (*x/c*) ; *z'(x)* = -0.02 for $0.25 \le x/c \le 1.0$<br><br>Calculate (a) the angle of attack at zero lift, (b) the lift coefficient when AoA is 4 degrees?(c)the moment coefficient about the quarter chord, and (d)the location of the center of pressure in terms of $x_{cp}/c$, when AoA is 4 degrees. | **20** | **CO 4** |