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| **Name:**<br><br>**Enrolment No:** | UPES<br>UNIVERSITY WITH A PURPOSE |

**UNIVERSITY OF PETROLEUM AND ENERGY STUDIES**
**End Semester Examination, July 2020**

**Course: Supersonic Aerodynamics** **Semester: VI**
**Program: B. Tech. ASE** **Time: 24 hrs.**
**Course Code: ASEG 3008** **Max. Marks: 100**

**Instructions:**
1. **Read the Instruction carefully before attempting**
2. **For Theory based : Type the Answers in word file**
3. **For Figures if any : Draw a free hand sketch and insert the same word file**
4. **For Numerical : Solve it in a paper and insert in the same word file**
5. **Upload as a single word/pdf file for all the questions in Blackboard.**

**Note: Please upload the word document/pdf format only. The answer scripts will be considered for evaluation only through Blackboard. No other mode of submission is acceptable.**

**Instructions: Assume any missing data appropriately. The use of tables for Isentropic flow properties, Normal shock tables, Oblique shock tables, Prandtl's Meyer function and *θ-β-M* chart is permitted.**

**SECTION A**

| S. No. | | Marks | CO |
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| Q 1 | A pitot-static tube is placed in a supersonic flow in which the static temperature is 0°C. Measurements indicate that the static pressure is 80 kPa and that the ratio of pitot to the static pressure is 4.1. Find the Mach number and the velocity in the flow. | **04** | **CO1** |
| Q 2 | Discuss any four characteristics of hypersonic flows. | **04** | **CO1** |
| Q 3 | Prove that the tangential component of velocity is unchanged across an oblique shock wave. | **04** | **CO2** |
| Q 4 | Consider an isentropic expansion corner wherein the flow is turned through a total angle of 20º. The Mach number and pressure upstream of the wave are $M_1$ = 5.0 and $p_1$=1 atm., respectively. Calculate the Mach number and pressure in region 2 behind the compression waves. | **04** | **CO2** |
| Q 5 | Using linearized theory, calculate the lift and wave drag coefficients for an infinitesimally thin flat plate in a Mach 2.6 freestream at angles of attack of α =5°. | **04** | **CO4** |

**SECTION B**

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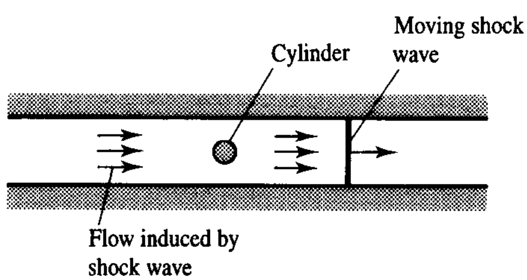
| Q 6 | Consider a supersonic flow with an upstream Mach number of 4 and a pressure of 1 atm. This flow is first expanded around an expansion corner with $\theta=15^o$ and then compressed through a compression corner with equal angle $\theta=15^o$ so that it is returned to its original upstream direction. Calculate the Mach number and pressure downstream of the compression corner.<br>***OR***<br>Consider a supersonic flow with Mach number, pressure and temperature of 3.0, 1 atm and 300 K respectively. The flow is deflected through an angle $\theta_1=14^\circ$ by a compression corner at a point A on the lower wall, creating an oblique shock wave emanating from point A. This shock impinges on the upper wall at point B. Also precisely at point B the upper wall is bent through an angle $\theta_2=10^\circ$.The incident shock is reflected at point B, creating a reflected shock wave which propagates downward and to the right. Calculate the Mach number, pressure and temperature in the region behind the reflected shock wave. | **10** | **CO2** |
| Q 7 | A normal shock wave, across which the pressure ratio is 1.17, moves down a duct, into still air at a pressure of 105 kPa and a temperature of 30°C. Find the pressure, temperature and velocity of the air behind the shock wave. This shock wave passes over a small circular cylinder as shown in Figure below. Assuming that the shock is unaffected by the small cylinder, find the pressure acting at the stagnation point on the cylinder after the shock has passed over it.<br><br> | **10** | **CO1** |
| Q 8 | A simple wing may be modeled as a 0.25 m wide flat plate set at an angle of $3^o$ to an air flow at a Mach number of 2.5, the pressure in this flow being 60 *kPa*. Assuming that the flow over the wing is two-dimensional, estimate the lift and drag force per meter span due to the wave formation. Use shock expansion theory. | **10** | **CO3** |

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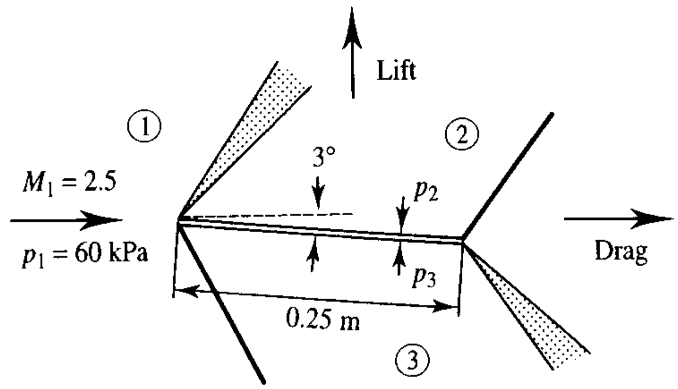
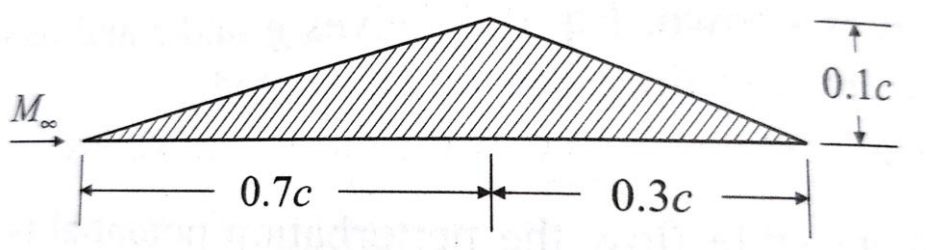
| |  | | |
| Q 9 | A two-dimensional wing profile as shown in figure below is placed in a stream of Mach number 2.5 at an incidence of 2°. Using linear theory, calculate the lift and drag coefficients.<br> | **10** | **CO4** |
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| **SECTION-C** | | | |
| Q 10 | Under low-speed incompressible flow conditions, the pressure coefficient at a given point on an airfoil is -0.54. Calculate the *Cp* at this point when the freestream Mach number is 0.58. If the given point is the point of minimum pressure, find the critical Mach number for this airfoil.<br>***OR***<br>A shallow irregularity of length l, in a plane wall, as shown in figure below, is given by expression $y = kx(1 - x/l)$, where $0 < x < l$ and $k << 1$. A uniform supersonic stream with freestream Mach number $M_\infty$ is flowing over it. Using linearized theory; derive an expression for the perturbation velocity potential and local pressure coefficient on the surface. | **20** | **CO4** |

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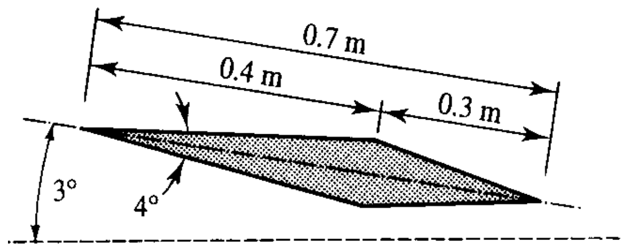
| Q 11 | Using the shock expansion theory, find the lift per meter span for the wedge shaped airfoil shown in figure below. The Mach number and the pressure ahead of the airfoil are 3.0 and 50 *kPa* respectively.<br><br> | **20** | **CO3** |
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NOTE : The submission time of the Question Paper Answer Sheet is 24 Hhrs from the scheduled time (*exceptional provision due to extraordinary circumstance due to COVID-19 and due to internet connectivity issues in the far-flung areas*).
No Submission will be entertained after 24 Hrs