| Name:<br><br>Enrolment No: | UPES — UNIVERSITY OF TOMORROW |
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**UPES**

**End Semester Examination, December 2024**

**Course: Aerodynamics – II** **Semester: V**

**Program: B.Tech, ASE** **Time: 03 hrs.**

**Course Code: ASEG 3011** **Max. Marks: 100**

Note: Use of Gas Table is allowed. θ- β -M curve is given at the end of the question paper.

**SECTION A**

**(5Qx4M=20Marks)**

| S. No. | | Marks | CO |
|---|---|---|---|
| Q1. | How do the flow properties change across Prandtl Mayer expansion waves? Discuss with the help of a sketch. | **4** | **CO1** |
| Q2. | Write the basic governing equations for a steady, one-dimensional, and inviscid flow across an oblique shock wave. | **4** | **CO3** |
| Q3. | Define critical Mach number and drag divergence Mach number. | **4** | **CO4** |
| Q4. | Write the linearized form of the perturbation velocity potential equation. write the conditions under which it is valid. | **4** | **CO5** |
| Q5. | Using thin airfoil theory for the theoretical lift coefficient for a thin, symmetric airfoil in an incompressible flow, estimate the lift coefficient for the airfoil in a compressible flow at $M_\infty = 0.7$ and 7° angle of attack. | **4** | **CO5** |

**SECTION B**

**(4Qx10M= 40 Marks)**

| S. No. | | Marks | CO |
|---|---|---|---|
| Q6. | Using the integral form of the momentum equation, derive the basic governing equations for a steady, one-dimensional, and inviscid flow across a normal shock wave. | **10** | **CO1** |
| Q7 | Consider a supersonic flow with $M = 2$, $p = 1$ atm, and $T = 288$ K. This flow is deflected at a compression corner through 20°. Calculate $M$, $p$, $T$, $p_0$, and $T_0$ behind the resulting oblique shock wave. | **10** | **CO3** |
| Q8 | Consider a point in an airflow where the local Mach number, static pressure, and static temperature are 3.5, 0.3 atm, and 180 K, respectively. Calculate the local values of $p_0$, $T_0$, $T*$, $a*$, and $M*$ at this point.<br><br>OR<br><br>An airfoil is in a freestream where $p_\infty = 0.61$ atm, $\rho_\infty = 0.819$ kg/m3, and $V_\infty = 300$ m/s. At a point on the airfoil surface, the pressure is 0.5 atm. | **10** | **CO2** |

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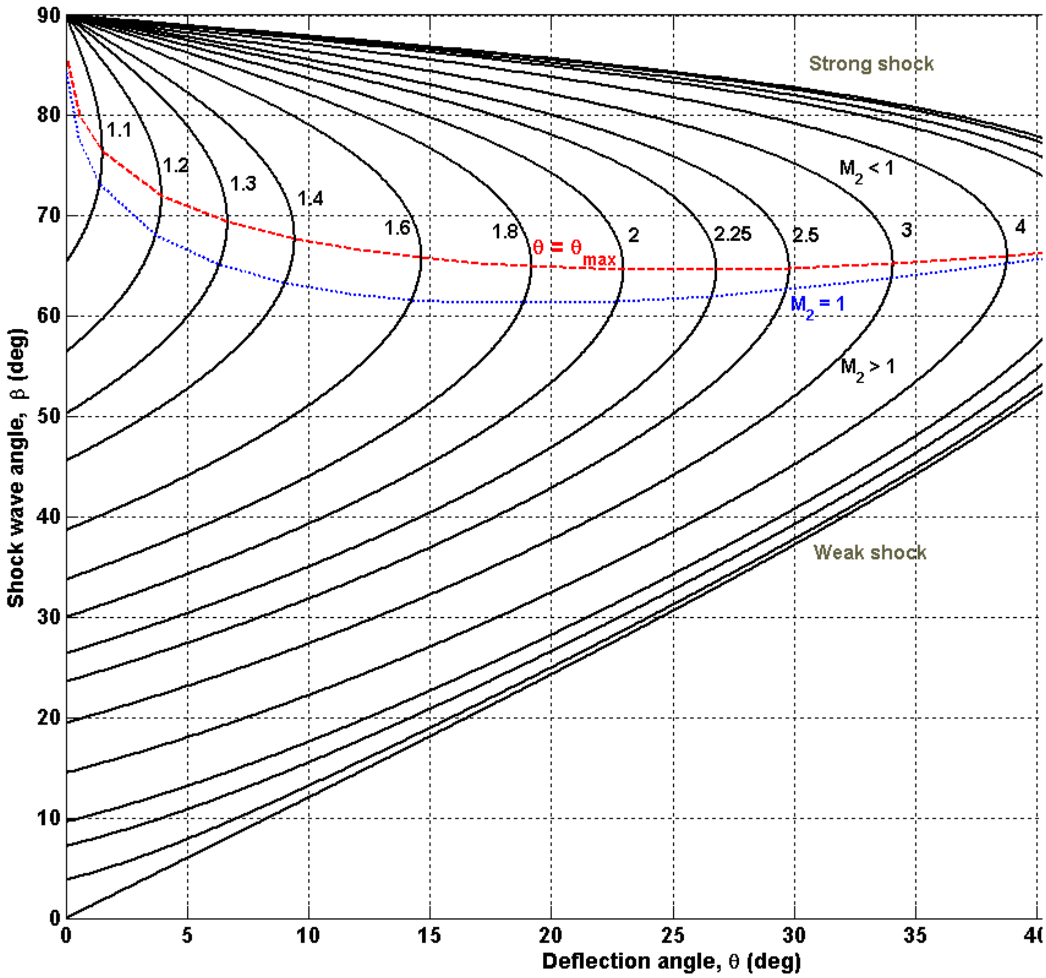
|---|---|---|---|
| | a) Assuming isentropic flow, calculate the velocity at that point.<br>b) Calculate the percentage error obtained if this problem is solved using (incorrectly) the incompressible Bernoulli equation. | | |
| Q9 | Using the velocity potential equation for an inviscid, compressible flow, derive the perturbation velocity potential equation. | **10** | **CO5** |
| **SECTION-C**<br>**(2Qx20M=40 Marks)** | | | |
| Q 10 | Consider an infinitely thin flat plate at an angle of attack α in a Mach 3 flow. Calculate the lift and wave-drag coefficients for:<br>(a) $\alpha = 5^\circ$<br>(b) $\alpha = 15^\circ$<br>OR<br>A supersonic flow at $M_1 = 3$, $T_1 = 285$ K, and $p_1 = 1$ atm is deflected upward through a compression corner with $\theta = 30.6^\circ$ and then is subsequently expanded around a corner of the same angle such that the flow direction is the same as its original direction. Calculate $M_3$, $p_3$, and $T_3$ downstream of the expansion corner. The resulting flow is in the same direction as the original flow. Comment on the results. | **20** | **CO4** |
| Q 11 | Consider an oblique shock wave generated by a compression corner with a $10^\circ$ deflection angle. The Mach number of the flow ahead of the corner is 3.6; the flow pressure and temperature are standard sea-level conditions. The oblique shock wave subsequently impinges on a straight wall opposite the compression corner. Calculate the angle of the reflected shock wave relative to the straight wall. Also, obtain the pressure, temperature, and Mach number behind the reflected wave. | **20** | **CO3** |




*Relation between Mach number, flow deflection angle and the wave angle*