Code No: RR322105

Set No.1

III B.Tech. II Semester Regular Examinations, April/May -2005
HIGH SPEED AERODYNAMICS
(Aeronautical Engineering)

Time: 3 hours **Max Marks: 80**

Answer any FIVE Questions
All Questions carry equal marks

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1. If U is the internal energy of a system under equilibrium ,then show that $dU = \left(\frac{\partial U}{\partial S}\right)_V dS + \left(\frac{\partial U}{\partial V}\right)_S dV$ .Describe that entropy is extensive property and that it is contained in the 2^{nd} law of thermodynamics.

2. Prove the relation $M_2^2 = \frac{1+\frac{\gamma-1}{2}M_1^2}{\gamma M_1^2 - \frac{\gamma-1}{2}}$ for a Normal shock and comment on the situation when $M_1 = 1$.What happens when M_1 just becomes greater than 1.

3. Consider a supersonic flow with M =2, p = 1 atm and T =288 K.The flow is defected at a compression corner through 20^0 . Calculate M ,p, T ,p_0 and T_0 behind the resulting oblique shock.

4. The θ - β - M relation for an oblique shock wave is given by $\tan\theta = 2\cot\beta \frac{M_1^2 \sin^2\beta - 1}{M_1^2(\gamma + \cos 2\beta) + 2}$ .Consider the θ - β - M diagram and explain the following situations;

   (a) If in a given physical problem θ is fixed and M_1 is decreased

   (b) if $\theta > \theta_{\ max}$. Make use of sketches and plots along with a drawn θ - β - M diagram on your answer sheet.

5. Define the term Impulse function in regard to the thrust exerted by isentropic flow of a fluid. Show that the non-dimensional impulse function is given in terms of the Mach number of the fluid stream as $\frac{F}{F^*} = \frac{1+\gamma M^2}{M\sqrt{2(1+\gamma)\left(1+\frac{\gamma-1}{2}M^2\right)}}$

6. The Mach number and pressure at the entry of a subsonic diffuser are 0.9 and 4.163 bar. Determine the area ratio reqd. and the pressure rise if the mach number at the exit of the diffuser is 0.20. Assume isentropic diffusion of air.

7. The equation of 2-D motion of fluid in small perturbation velocity components is given by $(1 - M_\infty^2)\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = M_\infty^2(\gamma + 1)\frac{u}{U}\frac{\partial u}{\partial x}$, where u and v are the perturbation velocity components given by u_1 = U + u and u_2 = v and M∞ is the free stream Mach number of the flow. Develop the expression for the pressure coefficient C_p [$C_p = \frac{p-p_\infty}{\frac{1}{2}\rho_\infty V_\infty^2}$ in incompressible flow] in terms of the flow over an elongated body.

8. Define the term critical Mach number. Hence present the variation of lift and drag coefficients over a selected aerodynamic shape with Mach number. Is the critical Mach number unique in the consideration of such flows? Make use of illustrated sketches and plots.

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Code No: RR322105 **Set No.2**

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Time: 3 hours **Max Marks: 80**

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1. Define the terms enthalpy, stagnation temperature and stagnation pressure. Air at a temperature of 293 K and a pressure of 1 atm (101,325 kPa) flows isentropically at a velocity of 310 m/s. Assuming air to behave as a perfect gas of constant specific heats, calculate the enthalpy, stagnation temperature and stagnation pressure .Explain the significance of stagnation properties

2. The state of a gas ($\gamma = 1.3$, R = 0.469 kJ/kg K) upstream of a normal shock wave is given by the following data: M_∞ =2.6, p_∞ = 2 bar, T_∞ =285 K. Calculate the Mach number, pressure, temperature and velocity of the gas down stream of the shock. Verify your calculations from the gas tables.

3. A thin wedge of semi vertex angle θ is placed in a supersonic flow of free stream Mach number M_1 and the shock angle referenced from the axis of the wedge is β . Show that the θ - β - M relation is given by $\tan\theta = 2\cot\beta \frac{M_1^2 \sin^2\beta - 1}{M_1^2(\gamma + \cos 2\beta) + 2}$

4. Air at $M_1 = 2.3$ and at a pressure of 70 kPa flows along a wall which bends away at an angle of 12^0 from the direction of flow. Determine the Mach number and pressure after the bend. If in another case the flow experiences a compression over the concave wall which actually bends through the same angle, determine the Mach number and pressure with the same free stream conditions. Sketch the flow fields in both the cases.

5. Consider the equation of continuity in isentropic flow given as $\rho AV = \rho^* A^* V^*$,where A* represents reference area at sonic conditions. Show that $\frac{A}{A^*}\frac{p}{p_0} = \frac{1}{M}\left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/2(\gamma-1)} / \left(1 + \frac{\gamma-1}{2}M^2\right)^{1/2}$ Plot the variation of area ratio for subsonic and supersonic isentropic acceleration and deceleration.

6. Air at a temperature of 284 K and atmospheric pressure flows isentropically through a C-D nozzle.The velocity at the inlet is 150 m/s and the inlet area is $10cm^2$ .If the flow at the exit of the nozzle is supersonic, find
   (a) M_{inlet},
   (b) p_0 and T_0
   (c) temperature and pressure at the throat ,
   (d) the velocity and Mach number at exit if T_2 = 220 K, and
   (e) area at the throat. Assume isentropic flow.

7. A thin airfoil when kept in a compressible ,irrotational ,isentropic flow introduces a perturbation potential $\hat{\phi}$. Making use of the velocity potential equation for compressible isentropic flows, develop $\left[a^2 - \left(V_\infty + \frac{\partial\hat{\phi}}{\partial x}\right)^2\right]\frac{\partial^2\hat{\phi}}{\partial x^2} + \left[a^2 - \left(\frac{\partial\hat{\phi}}{\partial y}\right)^2\right]\frac{\partial^2\hat{\phi}}{\partial y^2} - 2\left(V_\infty + \frac{\partial\hat{\phi}}{\partial x}\right)\frac{\partial\hat{\phi}}{\partial y}\left(\frac{\partial^2\hat{\phi}}{\partial x\partial y}\right) = 0$ .State the B.C.

8. Define critical Mach number and drag divergence Mach number while studying the variation of drag on a given aerodynamic shape with Mach number up to sonic speed. Explain this variation for an airfoil and a circular cylinder on a single plot. Hence locate crest critical Mach number also along with. Describe the whole phenomenon in details.

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1. Consider the change of density of a single component single phase fluid as a function of temperature and pressure. Hence show that $E = \frac{1}{K-\beta\frac{dT}{dp}}$ where E is the bulk modulus of elasticity , β = coefficient of volume expansion and K = isothermal coefficient of compression.

2. The state of a gas (γ = 1.3, R = 0.469 kJ/kg K) upstream of a normal shock wave is given by the following data: M_∞ =2.5, p_∞ = 2 bar, T_∞ =275 K. Calculate the Mach number, pressure, temperature and velocity of the gas down stream of the shock. Verify your calculations from the gas tables.

3. The θ - β - M relation for an oblique shock wave is given by $\tan\theta = 2\cot\beta\frac{M_1^2\sin^2\beta-1}{M_1^2(\gamma+\cos 2\beta)+2}$. Show that if $\theta < \theta_{max}$, then a plot of this equation give two solutions ,a weak shock and a strong shock. Which solution is to be used and under what condition? Demonstrate with sketches and plots.

4. Air at M_1 = 2.2 and at a pressure of 70 kPa flows along a wall which bends away at an angle of 12^0 from the direction of flow. Determine the Mach number and pressure after the bend. If in another case the flow experiences a compression over the concave wall which actually bends through the same angle, determine the Mach number and pressure with the same free stream conditions. Sketch the flow fields in both the cases.

5. Consider the equation of continuity under isentropic flow conditions and define the non-dimensional mass flow parameter .Obtain the relationship for the same as given below $\frac{m\sqrt{T_0}}{Ap_0}\sqrt{\frac{R}{\gamma}} = \left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/2(\gamma-1)}\sqrt{\frac{2}{\gamma-1}\left(\frac{\gamma+1}{2}-1\right)}$

6. Air at velocity of 210 m/s decelerates through a diffuser to a velocity of 60 m/s The temperature and pressure at the inlet are 278 K and 80 kPa with the exit pressure of 90 kPa.Assuming 1-D steady flow, calculate
   (a) the change in stagnation pressure ,
   (b) the change in entropy, and
   (c) the diffuser efficiency.

7. An airfoil placed in an irrotational , compressible, isentropic flow field introduces small perturbation in the flow field in the form of perturbation potential. Making use of the velocity potential equation already developed; obtain the small perturbation equation for flow over the airfoil. State all limitations.

8. Define critical Mach number and plot lift and drag coefficient v/s Mach number for a conventional airfoil .Now describe a supercritical airfoil due to Whitcomb and plot the aerodynamic characteristics for this airfoil section on the same plot. Illustrate further with Cp plot.

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1. Consider the change of density of a single component single phase fluid as a function of temperature and pressure. Hence show that $E = \frac{1}{K-\beta\frac{dT}{dp}}$
where E is the bulk modulus of elasticity , β = coefficient of volume expansion and K = isothermal coefficient of compression.

2. Describe the phenomenon of shock wave in a compressible flow . Properties of air upstream of a Normal shock are V =690 m/s, static pressure =90 kPa, static temperature = 363 K. Determine the velocity ,static properties and stagnation properties of the gas down stream of the shock. Determine the entropy increase as well.

3. Develop the Prandtl relation in supersonic flows for oblique shocks and show that the normal shock may be considered as the limiting case for a strong oblique shock in which the shock angle is 90^0 and that the deflection angle of the streamline is zero.

4. The θ - β - M relation for an oblique shock wave is given by $\tan\theta = 2\cot\beta\frac{M_1^2\sin^2\beta-1}{M_1^2(\gamma+\cos 2\beta)+2}$ .Consider the θ - β - M diagram and explain the following situations;
   (a) If in a given physical problem θ is fixed and M_1 is decreased
   (b) if $\theta > \theta_{max}$. Make use of sketches and plots along with a drawn θ - β - M diagram on your answer sheet.

5. Consider the equation of continuity under isentropic flow conditions and define the non-dimensional mass flow parameter .Obtain the relationship for the same as given below $\frac{m\sqrt{T_0}}{Ap_0}\sqrt{\frac{R}{\gamma}} = \left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/2(\gamma-1)}\sqrt{\frac{2}{\gamma-1}\left[\left(\frac{2}{\gamma+1}\right)^{(1-\gamma)/(\gamma-1)}-1\right]}$

6. Air flows isentropically through a nozzle of throat area $6cm^2$ and exit area $24cm^2$. If p_0 = 600 kPa and $T_0 = 200^0$C, compute the mass flow, exit pressure and exit mach number for
   (a) subsonic flow,
   (b) supersonic flow.

7. A thin airfoil when kept in a compressible ,irrotational ,isentropic flow introduces a perturbation potential $\hat{\phi}$. Making use of the velocity potential equation for compressible isentropic flows, develop

$\left[a^2 - \left(V_\infty + \frac{\partial \hat{\phi}}{\partial x}\right)^2\right] \frac{\partial^2 \hat{\phi}}{\partial x^2} + \left[a^2 - \left(\frac{\partial \hat{\phi}}{\partial y}\right)^2\right] \frac{\partial^2 \hat{\phi}}{\partial y^2} - 2\left(V_\infty + \frac{\partial \hat{\phi}}{\partial x}\right) \frac{\partial \hat{\phi}}{\partial y} \left(\frac{\partial^2 \hat{\phi}}{\partial x \partial y}\right) = 0$ .State the B.C.

8. Explain when a fluid is termed incompressible and what conditions make it compressible. Hence demonstrate the effect of Mach number on the flow past an airfoil with Mach number increasing from near zero to unity. Make use of sketches and plots to illustrate your point.

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