Code No: 09A52102 | R09 | SET-1

B.Tech III Year I Semester Examinations,
AERODYNAMICS – II
(AERONAUTICAL ENGINEERING)

Time: 3 hours **Max. Marks: 75**

Answer any five questions
All questions carry equal marks

1.a) Mathematically define compressible flow. What do you mean by isothermal compressibility, isentropic compressibility?
b) Explain in brief various flow regimes of compressible flow using neat sketches. [7+8]

2. Obtain the energy equation for one dimensional flow given by,

$$h_1 + \frac{u_1^2}{2} + q = h_2 + \frac{u_2^2}{2}$$

Mention its alternative forms [15]

3. Explain the design of high lift to drag hypersonic configurations(Wave riders). [15]

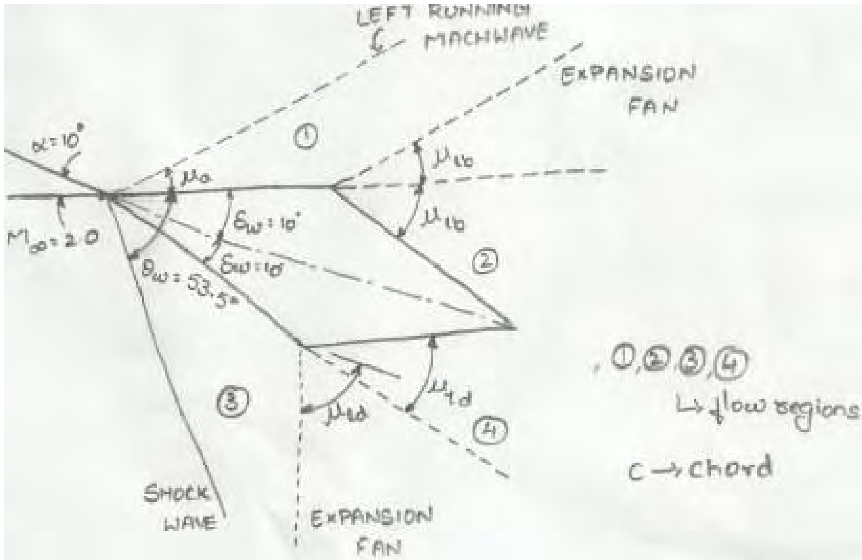
4. Define oblique shock wave. Obtain the relation between flow properties viz., Mach number, pressure, density and temperature ahead and behind the oblique shock wave. [15]

5.a) Obtain the area Mach number relation for a variable area duct when there is an isentropic flow of a calorically perfect gas through it.
b) Consider the subsonic-supersonic flow through a convergent-divergent nozzle. The reservoir pressure and temperature are 10 atm and 300 K respectively. There are two locations in the nozzle where A/A^* =6; one in the convergent section and other in the divergent section, calculate Mach number, pressure, Temperature and velocity. [8+7]

6. Write a short notes on
a) Critical Mach number
b) Drag divergence Mach number
c) Super critical airfoil [5+5+5]

7.a) Using neat sketches explain briefly how the supersonic flow past different wing plan forms is evaluated.
b) Write short notes on aerodynamic interaction. [9+6]

8. The geometry of a double wedge airfoil section is as shown in figure



Using linear theory, calculate the lift coefficient, wave drag coefficient, pitching moment coefficient and pressure coefficient on each panel of the airfoil. [15]

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Code No: 09A52102 | R09 | SET-2

B.Tech III Year I Semester Examinations,
AERODYNAMICS – II
(AERONAUTICAL ENGINEERING)

Time: 3 hours **Max. Marks: 75**

Answer any five questions
All questions carry equal marks

1.a) Define shock polar using neat sketches.
b) Define Mach reflection using neat sketches. [8+7]

2.a) Define first law of Thermodynamics. Obtain the relations for internal energy and enthalpy for a perfect gas.
b) Briefly explain the fluid models which are used to extract the mathematical equations using neat sketches. [7+8]

3.a) Explain the working of a supersonic wind tunnel using neat sketches.
b) A supersonic wind tunnel is designed to produce flow in the test section at Mach 2.4 at standard atmospheric conditions. Calculate:
i. The exit-to- throat area ratio of the nozzle
ii. Reservoir pressure and temperature [7+8]

4. A rectangular wing having an aspect ratio of 3.5 is flying at M_∞ = 0.85 at 12 Km. A NACA 0006 airfoil section is used at all span wise stations. What is the airfoil section and aspect ratio for equivalent wing in an incompressible flow? [15]

5. Explain the method of characteristics for a two dimensional irrotational supersonic flow and determine the characteristic lines for the same. [15]

6. Solve the equation $(M_\infty^2 - 1)\emptyset_{xx} - \emptyset_{yy} - \emptyset_{zz} = 0$ using singularity distribution method. Obtain the pressure distribution given an arbitrary configuration. [15]

7. Write a short notes on
a) Aerodynamic heating
b) Mach Number independence
c) Law of hypersonic similarity. [5+5+5]

8.a) Define normal shocks, quasi one dimensional flow using neat sketches.
b) At a given point in the high speed flow over an airplane wing, the local Mach number, pressure and temperature are 0.7, 0.9 atm and 250 K, respectively. Calculate the values of stagnation pressure, stagnation temperature, characteristic temperature, characteristic Mach number, ***a**** at this point. [4+11]

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Code No: 09A52102	R09	SET-3

B.Tech III Year I Semester Examinations,
AERODYNAMICS – II
(AERONAUTICAL ENGINEERING)

Time: 3 hours **Max. Marks: 75**

Answer any five questions
All questions carry equal marks

1.a) State the fundamental physical principles from the laws of nature.
b) Derive momentum equation considering an appropriate fluid model. [4+11]

2.a) Derive the governing differential equation for Prandtl-Meyer flow given by $d\theta = \sqrt{M^2 - 1}\,\frac{dV}{V}$
b) Consider a wedge with a half angle of 10^o flying at Mach 2. Calculate the ratio of total pressures across the shock wave emanating from the leading edge of the wedge. [7+8]

3. In detail, explain the design considerations for supersonic aircraft. What are the major considerations to build a supersonic transport aircraft (SST)? Discuss. [15]

4. Write a short notes on
a) Diffusers
b) Wave reflection from a free boundary
c) Three dimensional shock waves [5+5+5]

5. Derive the relation $C_p = \frac{C_p'}{\sqrt{1-M_\infty^2}}$ in accordance with linearized theory of compressible subsonic flow about a thin wing with relatively small angles of attack. [15]

6.a) What are the general features for linearized supersonic flow past a thin airfoil? Discuss.
b) Under proper assumptions, derive the incremental lift force and incremental drag force expressions using linear theory which are given by
$\frac{dC_l}{d\alpha} = \frac{4}{\sqrt{M_\infty^2-1}}$; $C_{d,thickness} = \frac{2}{\sqrt{M_\infty^2-1}}\left[\overline{\sigma_u^2} - \overline{\sigma_l^2}\right]$ [5+10]

7.a) Obtain the expression for speed of sound, given by, $a = \sqrt{\gamma RT}$. Define Mach number.
b) Relate the thermodynamic properties across a shock wave and name the relation. [6+9]

8. Derive the shock wave relations for a hypersonic flow. [15]

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Code No: 09A52102	R09	SET-4

B.Tech III Year I Semester Examinations,
AERODYNAMICS – II
(AERONAUTICAL ENGINEERING)

Time: 3 hours **Max. Marks: 75**

Answer any five questions
All questions carry equal marks
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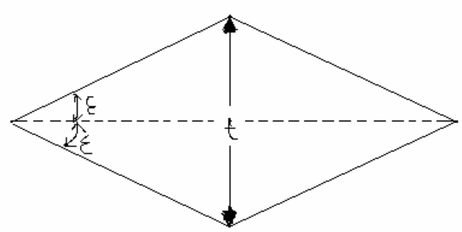
1. Obtain the following relations for the flow properties ahead and behind the normal shock wave

$$M_2^2 = \frac{1 + [\frac{\gamma - 1}{2}]M_1^2}{\gamma M_1^2 - \frac{\gamma - 1}{2}}$$

$$\frac{p_2}{p_1} = 1 + \frac{2\gamma}{\gamma + 1}[M_1^2 - 1]$$

And using above deduce $\frac{\rho_2}{\rho_1}, \frac{T_2}{T_1}$ [15]

2. Using shock expansion theory, calculate the lift and drag on a symmetrical diamond airfoil of semi-angle ε=15^o as shown in the figure, at an angle of attack to the free stream of 5^o when the upstream mach number and pressure are 2.0 and 10331.2 kg/m^2, respectively. The maximum thickness of the airfoil is t=0.5 ft. assume, a unit length of 1 foot (0.348m) in the span direction (perpendicular to the page). [15]



3. Using neat sketches explain how lift and drag varies in a transonic flow past un-swept airfoils. Discuss about pressure distribution and local Mach number distribution for the same. [15]

4. Explain
a) Shock expansion technique
b) Second order theory (Busemann's theory)
c) General features for linearized supersonic flow past a thin airfoil. [5+5+5]

5.a) Derive the energy equation assuming an appropriate flow model.
b) Focus on the special forms of energy equation. [8+7]

6. Using neat sketch explain Newtonian flow model and how is it modified. [15]

7.a) Define conical flow.

b) How is a supersonic flow over rectangular wing, treated using conical-flow theory? Explain using neat sketches. [3+12]

8.a) Briefly explain over expanded and under expanded nozzles using neat sketches.

b) Obtain the area-velocity relation for a quasi-one dimensional flow through a variable area control volume. Use general symbolic notations. [8+7]

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