Name: Enrolment No:	UPES UNIVERSITY WITH A PURPOSE

UNIVERSITY OF PETROLEUM AND ENERGY STUDIES
End Semester Examination, May 2019

Course: Supersonic and Hypersonic Flows **Semester: II**
Program: M. Tech. CFD **Time 03 hrs.**
Course Code: ASEG 7034 **Max. Marks: 100**

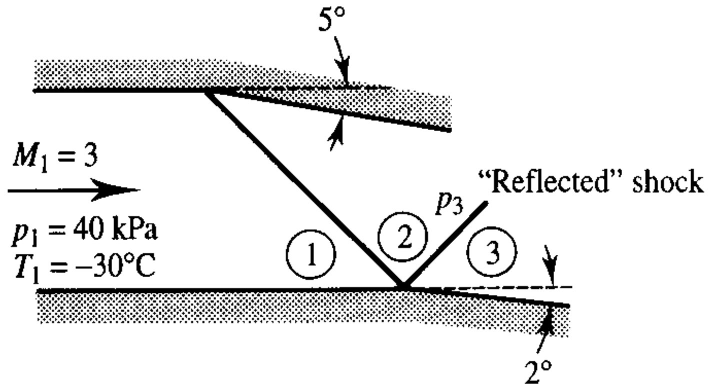
Instructions: Assume any missing data appropriately. The use of tables for isentropic flow properties, normal shock properties, Prandtl's Meyer function and *θ-β-M* chart is permitted.

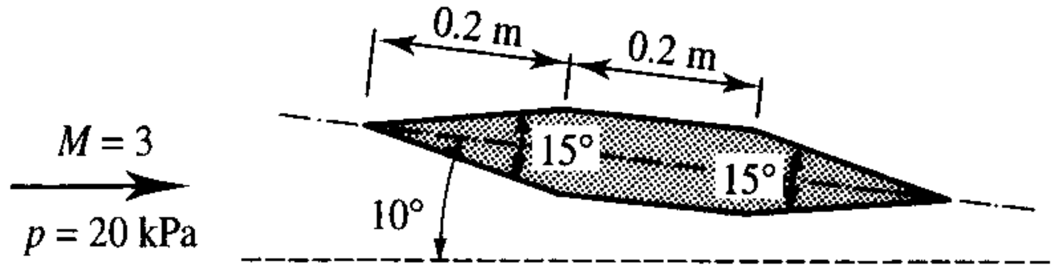
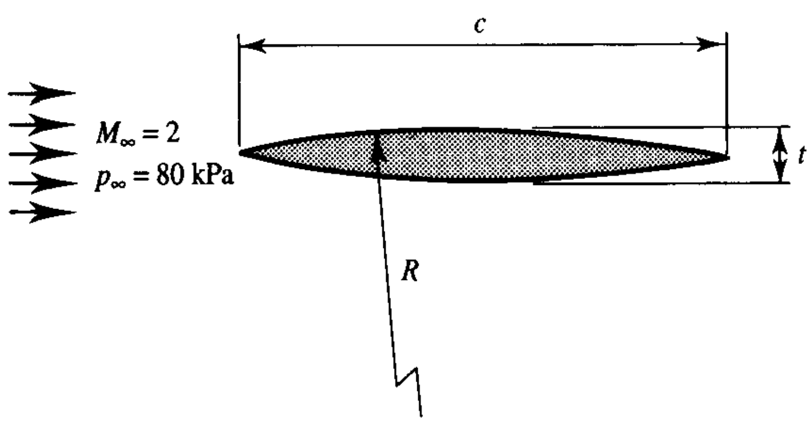
SECTION A

S. No.		Marks	CO
Q 1	A pitot tube is inserted into an airflow where the static pressure is 1 atm. Calculate the flow Mach number when the Pitot tube measures 2.714 atm.	**04**	**CO1**
Q 2	Discuss the following phenomena associated with hypersonic flows. (a) Shock Layer (b) Real Gas Effects	**04**	**CO1**
Q 3	Discuss the phenomena of Mach reflection with the help of suitable sketches.	**04**	**CO2**
Q 4	List the effects of following in context to oblique shock waves a. Change in Mach number for given deflection b. Change in deflection for a given Mach number	**04**	**CO2**
Q 5	Define Critical Mach number. Discuss the effects of airfoil thickness and wing sweep on Critical Mach number.	**04**	**CO4**

SECTION B

Q 6	Prove that the characteristic speed of sound, in an adiabatic flow across a normal shock, is equal to the geometric mean of the velocities ahead and behind the normal shock.	**10**	**CO1**
Q 7	Find the pressure, temperature and velocity in region 3 shown in figure below. The pressure, temperature and Mach number in region 1 are 40 kPa, -30 ^{0}C and 3.0 respectively.	**10**	**CO2**

			
Q 8	A simple wing may be modeled as a 0.3 m wide flat plate set at an angle of 3^0 to an airflow at a Mach number of 3, the pressure in this flow being 40 kPa. Assuming that the flow over the wing is two-dimensional, estimate the lift and drag force per meter span due to wave formation on the wing.	**10**	**CO3**
Q 9	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_∞ is flowing over it. Using linearized theory, derive an expression for the perturbation velocity potential and local pressure coefficient on the surface. ***OR*** Prove that the following transformations maps an irrotational isentropic flow in compressible space (x, y) to an irrotational, isentropic flow in an incompressible space (ξ, η). $\xi = x$ $\eta = \beta y$ $\acute{\phi}(\xi, \eta) = \beta \hat{\phi}(x, y)$ Also, establish that the shape of the airfoil is unaltered in the above transformation.	**10**	**CO4**
SECTION-C			
Q 10	Consider two-dimensional flow over the double-wedge airfoil as shown in figure below. Find the lift and drag coefficients of the airfoil as well as the pressure variations on its surface.	**20**	**CO3**

			
Q 11	Consider a diamond-wedge airfoil with a half-angle ε_0. The airfoil is at an angle of attack α to a freestream with Mach number M_∞. Using linear theory, derive expressions for lift and wave-drag coefficients of the airfoil. Estimate these coefficients if $\varepsilon_0 = 15^0$, $\alpha = 10^0$ and $M_\infty = 3.0$ ***OR*** An airfoil of the shape shown in figure below is placed in a flow at a Mach number of 2 and a pressure of 80 kPa. Derive expressions for the pressure variations along the upper and lower surfaces. 	**20**	**CO4**

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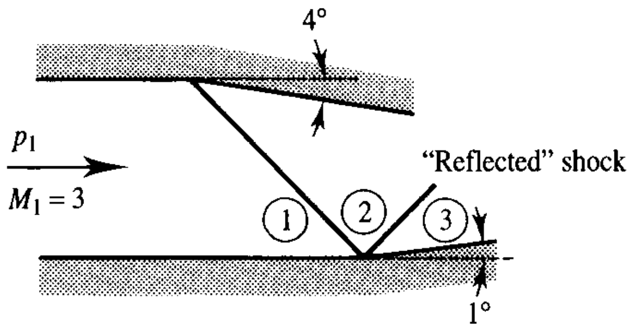
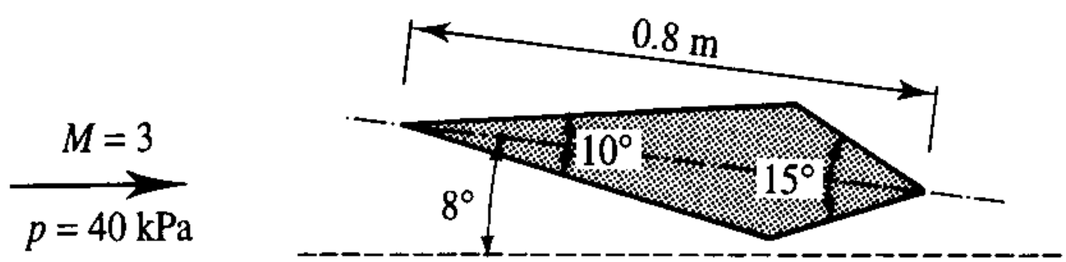
SECTION A

S. No.		**Marks**	**CO**
Q 1	What are the major inferences that can be extracted from the *θ-β-M* curves for oblique shock waves, which is a plot of wave angle versus flow deflection with Mach number as parameter?	**04**	**CO2**
Q 2	Discuss the qualitative aspects of hypersonic high temperature flows.	**04**	**CO1**
Q 3	For an inviscid two-dimensional compressible flow, prove that the component of velocity tangential to an oblique shock wave remains unchanged.	**04**	**CO1**
Q 4	Consider an isentropic compression corner wherein the flow is turned through a total angle of 15^0. The Mach number and pressure upstream of the wave are M_1 = 10 and p_1=1 atm., respectively. Calculate the Mach number and pressure in region 2 behind the compression waves.	**04**	**CO2**
Q 5	Consider a 15 ^{0} half-angle wedge at zero angle of attack in a Mach 3 flow of air. Calculate the drag coefficient. Assume that the pressure exerted over the base of the wedge, the base pressure, is equal to the freestream pressure.	**04**	**CO3**

SECTION B

Q 6	Consider motionless air with p_1=0.1 atm and T_1=300 K in a constant area tube. We wish to accelerate this gas to Mach 1.5 by sending a normal shock wave through the tube. Calculate the necessary value of the wave relative to the tube.	**10**	**CO1**
Q 7	Find the pressure, temperature and velocity in region 3 shown in figure below. The pressure, temperature and Mach number in region 1 are 20 kPa, 20 ^{0}C and 3.0 respectively.	**10**	**CO2**

			
Q 8	Establish that the Prandtl - Glauert's transformation maps an irrotational isentropic flow in compressible space (x, y) to an irrotational, isentropic flow in an incompressible space (ξ, η). Hence, find the P-G compressibility correction for pressure coefficient. ***OR*** Under low-speed incompressible flow conditions, the pressure coefficient at a given point on an airfoil is -0.36. Calculate the *Cp* at this point when the freestream Mach number is 0.62. If this point is the point of maximum suction on the airfoil, estimate the critical Mach number for the airfoil.	**10**	**CO4**
Q 9	Consider a supersonic flow with an upstream Mach number of M_∞. This flow moves over a wavy wall with a contour given by $y_w = h\cos(2\pi x/l)$, where y_w is the ordinate of the wall, h is the amplitude, and l is the wavelength. Assume that h is small. Using the small perturbation theory, derive an equation for the perturbed velocity potential and the surface pressure coefficient.	**10**	**CO4**
SECTION-C			
Q 10	For the double wedge airfoil shown in figure below, find the lift per meter span if the Mach number and pressure in the uniform flow ahead of the airfoil are 3 and 40 kPa respectively. Use shock-expansion theory. 	**20**	**CO3**

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<td>Q 11</td>
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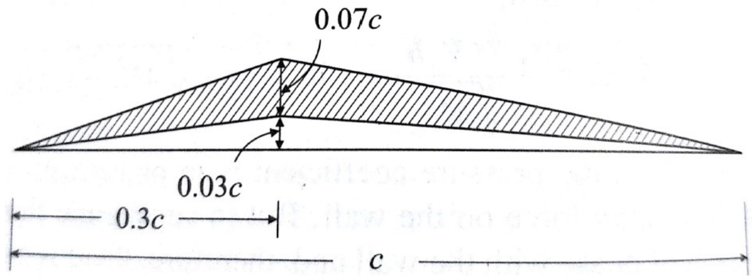
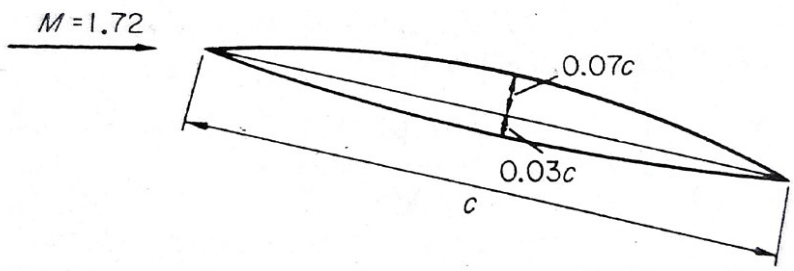
A two-dimensional thin airfoil shown in figure below is placed in a stream with Mach number 3.0 and is at an angle of attack of 2^0. Using linearized theory, estimate the pressure coefficients on the top and bottom surfaces. Also, find the lift and drag coefficients.


OR
Consider a biconvex airfoil section made up of two unequal circular arcs dimension as shown in figure below. The airfoil is placed in a freestream of Mach number 1.7 and angle of attack α. Using linearized theory; estimate the values of lift, and drag coefficients.


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