AE53- AERODYNAMICS – II TWO MARK QUESTION AND ANSWERS

1. What is the basic difference between compressible and incompressible fluid flow?

Compressible	Incompressible
1. Fluid velocities are appreciable compared with the velocity of sound 2. Density is not constant 3. Compressibility factor is greater than one.	1. Fluid velocities are small compared with the velocity of sound 2. Density is constant 3. Compressibility factor is one.

2. Write the steady flow energy equation for an adiabatic flow of air.

In an adiabatic flow q = 0. Therefore energy equation becomes.

$$h1 + \frac{c_1^2}{2} + gZ_1 = h_2 + \frac{c_2^2}{2} + gZ_2 + W_s$$

Adiabatic energy equation is $h_0 = h + ½ c^2$

3. Define the mach number in terms of bulk modulus of elasticity.

Mach number is a non-dimensional number and is used for the analysis of compressible fluid flows.

$$M = \sqrt{\frac{\text{int ertiaforce}}{\text{elasticforce}}}$$

$$= \sqrt{\frac{\rho A c^2}{KA}}$$

where K = Bulk modulus of elasticity $K = \rho a^2$

$$\therefore M = \sqrt{\frac{\rho A c^2}{\rho A a^2}} = \frac{c}{a}$$

4. Explain the meaning of stagnation state with example.

The state of a fluid attained by isentropically decelerating it to zero velocity at zero elevation is referred as stagnation state.

(e.g.) Fluid in a reservoir (or) in a settling chamber.

5. Distinguish between static and stagnation pressures.

In stagnation pressure state, the velocity of the flowing fluid is zero whereas in the static pressure state, the fluid velocity is not equal to zero.

6. What is the use of mach number?

Mach number is defined as the ratio between the local fluid velocity to the velocity of sound.

i.e. Mach number $M = \frac{Local fluid velocity}{Velocity of sound} = \frac{c}{a}$

It is used for the analysis of compressible fluid flow problems. Critical mach number is a dimensionless number at which the fluid velocity is equal to its sound velocity. Therefore,

$M_{critical} = \frac{c^*}{a^*} = 1$ [$\therefore c^* = a^*$]

Crocco number is a non – dimensional fluid velocity which is defined as the ratio of fluid velocity to its maximum fluid velocity.

i.e. $C_r = \frac{c}{c_{max}} = \frac{Fluid velocity}{Maximum fluid velocity}$

7. Write down the relationship between stagnation and static temperature interms of the flow, mach number for the case of isentropic flow.

$\frac{T_0}{T} = 1 + \frac{\gamma - 1}{2} M^2$ where,

T_0 = stagnation temperature

T = Static temperature

M = Mach number.

8. Give the expression of $\frac{P}{P_0}$ for an isentropic flow through a duct.

The expression of

$\frac{T_0}{T} = 1 + \frac{\gamma - 1}{2} M^2$, but we know that,

$$\frac{T_0}{T} = \left(\frac{P_0}{P}\right)^{\frac{\gamma-1}{\gamma}} (or) \frac{P_0}{P} = \left(\frac{T_0}{T}\right)^{\frac{\gamma}{\gamma-1}}$$

Therefore $\frac{P_0}{P} = \left[1 + \frac{(\gamma - 1)}{2} M^2\right]^{\frac{\gamma}{\gamma-1}} (or)$

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$$\frac{P_0}{P} = \frac{1}{\left[1+\frac{(\gamma-1)}{2}M^2\right]}\frac{\gamma}{\gamma-1}$$

9. Name the four reference velocities that are used in expressing the fluid velocities in non-dimensional form?

i. Local velocity of sound a = $\sqrt{\gamma RT}$

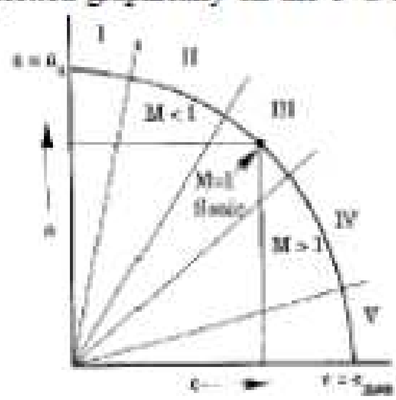
ii. Stagnation velocity of sound $a_0 = \sqrt{\gamma RT_0}$

iii. Maximum velocity of sound $C_{max} = a_0\sqrt{\frac{2}{\gamma-1}}$

iv. Critical velocity of sound / fluid a* = c* = $\sqrt{\gamma RT^*}$

10. What are the different regions of compressible flow.

The adiabatic energy equation for a perfect gas is derived in terms of fluid velocity © and sound velocity (a). This is then plotted graphically on the c- a co-ordinates, a steady flow ellipse is obtained.



The various regions of flow are:

(i) Incompressible region (M ≈ 0)

(ii) Subsonic region (M < 1)

(iii) Transonic region (0.8 – 1.2)

(iv) Supersonic region (M > 1 and M < 5)

(v) Hypersonic region (M ≥ 5)

11. Define M* and give the relation between M and M*.

It is a non-dimensional mach number and is defined by the ratio between the local fluid velocity to its critical velocity of sound / fluid.

$$M^* = \frac{c}{c^*} = \frac{c}{a^*}$$

It is also called a characteristic Mach number.

$$M^* = \sqrt{\frac{M^2(\gamma-1)}{2+M^2(\gamma-1)}}$$

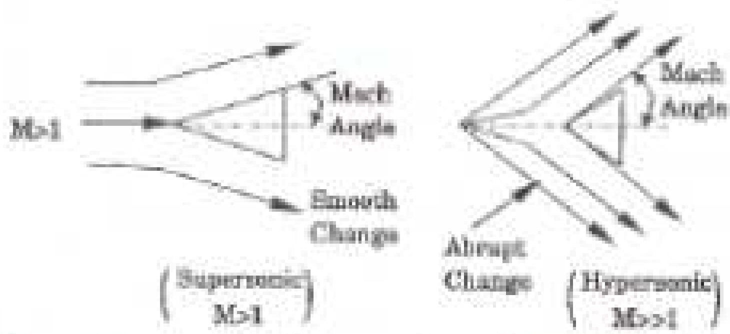
12. If an aeroplane goes to higher altitudes maintaining the same speed, the Mach number will remain constant. Say true or false.

False.

W.R.T. $M = \frac{c}{a}$

At higher altitude, the sound velocity 'a' will decrease and hence M will increase. Therefore, M is not constant.

13. Define mach angle and mach wedge.



Mach angle is formed, when an object is moving with supersonic speed. The wave propagation and changes are smooth. When an object is moving with hypersonic speed the changes are abrupt is shown in Fig. Hence for a supersonic flow over two – dimensional object "**mach wedge**" is used instead of "**mach cone**".

14. What is meant by isentropic flow with variable area?

A steady one dimensional isentropic flow in a variable area passages is called "variable area flow". The heat transfer is negligible and there are no other irreversibilities due to fluid friction, etc.

15. Find the sonic velocity in oxygen when it is at 110° C, γ=1.4 and molecular weight 32.

$a = \sqrt{\gamma RT} = \sqrt{1.4 x 259.8125 x 383} = 373.244 m/s$

16. A plane travels at a velocity of 1600 kmph at an altitude. Where the pressure and temperature of 40kpa and 35°C. Find the Mach angle and Mach number.

Given:

$C = 1600$ Km/hr, $P = 40$ Kpa, $T = -35^0C$

$$M = \frac{C}{a}$$

$$M = 1.43$$

$$\alpha = Sin^{-1}\left[\frac{1}{M}\right]$$

$$\alpha = 44.09^0$$

17. The static temperature of air is 300^0 C and velocity is 200 m/sec. find the maximum possible velocity obtainable by air.

Given:

T = 300^0C, C = 200 m/sec

$$h_0 = \frac{C^2_{max}}{2}$$

C_{max} = 1091.67 m/sec

18. An aircraft flying at a speed of 1000 km/hr, the variables of speed of sound and Mach number with altitude are as follows.
    a. At sea level altitude
    b. At 11000m altitude

Given:

C= 100 kmph, T = 15^0C at sea level T = -56.5^0C at 11000 m height

$$M = \frac{C}{G}$$

$$M = 0.816$$

$$M = 0.94$$

19. An aero plane travels at an altitude where the temperature -37^0C with a Mach number 1.2. Determine the velocity of the aero plane in Km/hr.

Given:

T = 37^0 K, M = 1.2

$$a = \sqrt{\upsilon RT}$$

a = 307.936 m/s

$$M = \frac{C}{a}$$

C = 1330.27 kmph

20. Calculate the velocity of sound and stagnation temperature of jet at 300k. Assume Mach number 1.2.

Given:

T = 300K M = 1.2

$$a = \sqrt{\upsilon RT} = 347.18 \text{ m/s}$$

$$\frac{T_0}{T} = \left[1 + \frac{\upsilon - 1}{2} M^2\right]$$

T_0 = 386.4 K

21. Differentiate between the static and stagnation temperatures.

The actual temperature of the fluid in a particular state is known as **"static temperature"** whereas the temperature of the fluid when the fluid velocity is zero at zero elevation is known as **"stagnation temperature"**.

$$T_0 = T + \frac{c^2}{2C_p} \text{ where}$$

T = static temperature

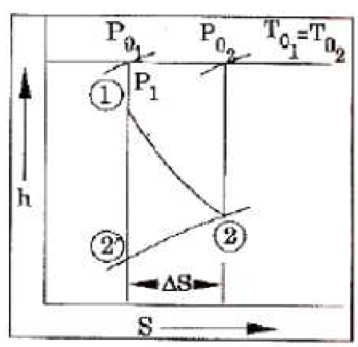
T_0 = stagnation temperature

$\frac{c^2}{2C_p}$ = velocity temperature

22. Show h – S diagram for the flow through a nozzle. Show how the stagnation properties get affected.

1 – 2' = Isentropic expansion

1 – 2 = Adiabatic expansion



It is assumed that, the exit pressure is same for both cases. But stagnation pressure at the exit of the adiabatic process (P_{0_2}) will be less than isentropic pressure (P_{0_2}'). This is due to friction and irresversibilities. But stagnation temperature remains constant.

23. A plane travels at a speed of 2400 KM/h in an atmosphere of 5°C, find the mach angle.

$$c = \frac{2400}{3.6} = 666.66667; T = 278K$$

$$M = \frac{c}{\gamma RT} = \frac{666.6667}{\sqrt{1.4x287x278}} = 1.9947$$

$$\alpha = \sin^{-1}\left(\frac{1}{M}\right) = 30.0876°$$

24. How will you illustrate the role of mach number as a measure of compressibility?

If the flow is assumed to be incompressible, the value of pressure co-efficient (or) compressibility factor obtained by Bernoulli equation is unity.

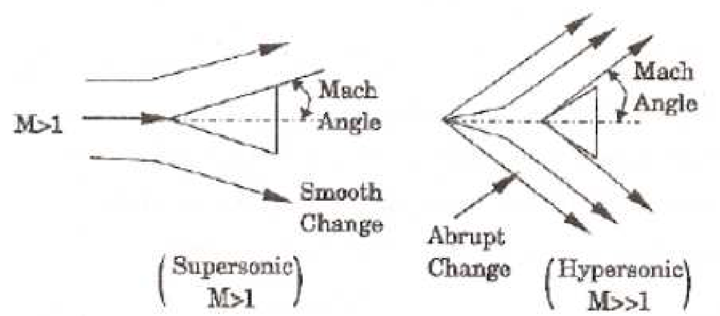
$$i.e., \quad \frac{P_0 - P}{\frac{\rho C^2}{2}} = 1 + \frac{M^2}{4} + \frac{M^4}{40} + ... \quad [\text{for} \quad \gamma = 1.4]$$

By substituting different values of M, we can get different values of compressibility factor and is given in the table.

M	Compressibility factor (%)	M	Compressibility factor(%)	M	Compressibility factor(%)
0.1	0.3	0.5	6.4	0.9	22
0.2	1.0	0.6	9.3	10	27.5
0.3	2.3	0.7	12.9		
0.4	4.1	0.8	17.0		

In the above table, when M increases, the compressibility factor also increases from the initial value 1. Thus the role of mach number is a measure of compressibility.

25. Define mach angle and mach wedge.



Mach angle is formed, when an object is moving with supersonic speed. The wave propagation and changes are smooth. When an object is moving with hypersonic speed the changes are abrupt is shown in Fig. Hence for a supersonic flow over two – dimensional object **"mach wedge"** is used instead of **"mach cone"**.

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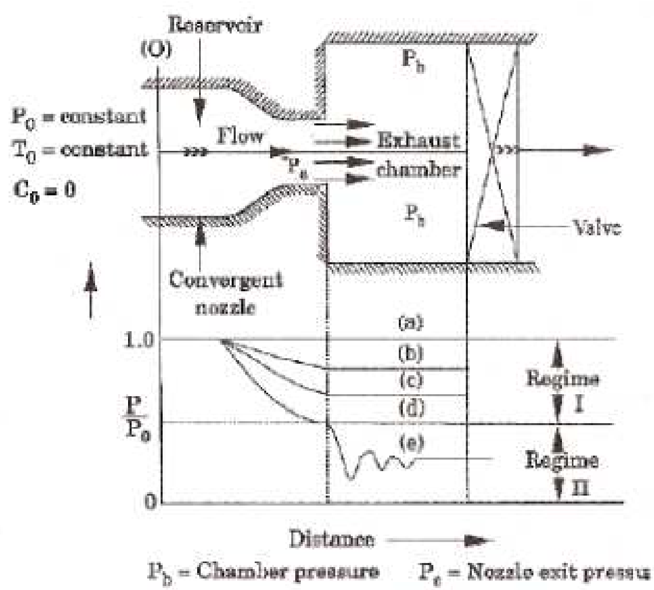
1. Give the expression for $\frac{T_0}{T}$ and $\frac{T}{T^*}$ for isentropic flow through variable area in terms of Mach number.

$$\frac{T}{T_0} = \frac{1}{1+\frac{\gamma-1}{2}M^2}$$

$$\frac{T}{T^*} = \frac{(\gamma-1)}{1+\frac{\gamma-1}{2}M^2}$$

2. Sketch the isentropic and adiabatic expansion process in P-V and T-S diagram.



Isentropic flow through a convergent nozzle

3. What will happen if the air flowing through a nozzle is heated?

When the flowing air is heated in a nozzle, the following changes will occur.

Velocity of air will increase.

Increase in temperature and enthalpy

Pressure increases

Increase in entropy

4. Write the Fliegner's formula.

$$\frac{m_{max}}{A^*} X \frac{\sqrt{T_0}}{P_0} = \sqrt{\frac{\gamma}{R}}\left(\frac{2}{\gamma+}\right)^{\frac{\gamma+1}{2(\gamma-1)}}$$

For air $\gamma = 1.4$ and R – 287 J / Kg° K [SI units]

$$\therefore \frac{m_{max}}{A^*}\frac{\sqrt{T_0}}{P_0} = 0.0404 \Rightarrow \text{Fliegner's formula}$$

5. Write the equation for efficiency of the diffuser.

$$\text{Diffuser efficiency} = \frac{\text{static pressure rise in the actual process}}{\text{static pressure rise in the ideal process}}$$

$$\frac{P_2 - P_1}{P_2' - P_1}$$

6. What is impulse function and give its uses?

Impulse function is defined as the sum of pressure force and intertia force. Impulse function F = Pressure force ρA + intertia force $\rho A c^2$

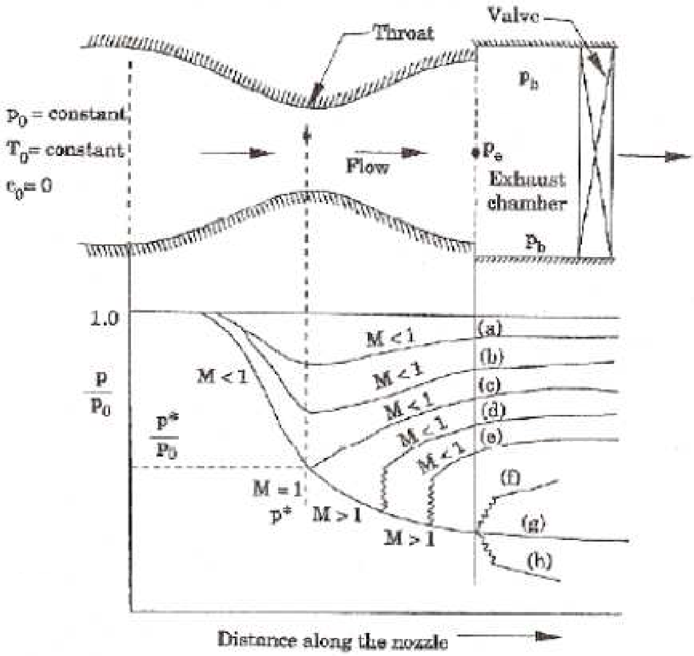
Since the unit of both the quantities are same as unit of force, it is very convenient for solving jet propulsion problems. The thrust exerted by the flowing fluid between two sections can be obtained by using change in impulse function.

7. What is chocked flow? State the necessary conditions for this flow to occur in a nozzle.

When the back pressure is reduced in a nozzle, the mass flow rate will increase. The maximum mass flow conditions are reached when the back pressure is equal to the critical pressure. When the back pressure is reduced further, the mass flow rate will not change and is constant. The condition of flow is called "chocked flow". The necessary conditions for this flow to occur in a nozzle is

* The nozzle exit pressure ratio must be equal to the critical pressure ratio where the mach number M = 1.

8. Draw the variation of $\frac{P}{P_0}$ along the length of a convergent divergent device when it functions as (a) diffuser, (b) nozzle and (c) venturi.



Curves

a, b, c ⇒ venture

d, e ⇒ diffuser

g ⇒ nozzle

9. Give the expression for nozzle efficiency and diffuser efficiency with h – s diagram.

$$\text{Nozzle efficiency } \eta N = \frac{\text{actual enthalpy drop}}{\text{ideal enthalpy drop}} = \frac{T_1 - T_2}{T_1 - T_2'}$$

$$\text{Diffuser efficiency} \eta D = \frac{\text{ideal enthalpy rise}}{\text{actual enthalpy rise}} = \frac{T_2' - T_1}{T_2 - T_1}$$

10. Give the important difference between nozzle and venturi.

NOZZLE	VENTURI
1. The flow is accelerated continuously i.e., Mach number and velocity increases	1. The flow is accelerated upto M = 1 and then Mach number is decreased.

continuously.	
2. Used to increase velocity and Mach number.	2. Used for flow measurement (discharge)
3. Generally convergent portion is short.	3. Convergent and divergent portions are equal.

11. A diffuser has exit throat, area ratio of 1.5 to 1. The inlet Mach number is 0.8. The initial pressure and temperature are 1 bar and 15^{0}C. Assume the flow to be isentropic. Calculate the following for air (i) exit pressure (ii) exit temperature (iii) exit Mach number.

Given:

$\frac{A_2}{A^*} = \frac{1.5}{1}$, $M_1 = 0.8$, $P_1 = 1$ bar, $T_1 = 15^0$C.

$$P_2 = 1.343 \text{ x } 10_5 \text{ N) m}^2$$

$$T_2 = 313.3 \text{ K}$$

$$M_2 = 0.43$$

12. Derive an expression for $\frac{A}{A^*} = \frac{1}{M}\left[\frac{2}{\upsilon+1} + \frac{\upsilon-1}{\upsilon+1}M^2\right]^{\frac{\upsilon+1}{2/\upsilon-1}}$

$$m = PAC$$

$$\frac{A}{A^*} = \frac{P^*}{P} x \frac{C^*}{C}$$

$$\frac{C^*}{C} = \frac{1}{M}\left[\frac{2}{(\upsilon+1)} + \left(\frac{\upsilon-1}{\upsilon+1}\right)M^2\right]^{\frac{1}{2}}$$

$$\frac{A}{A^*} = \frac{1}{M}\left[\frac{2}{\upsilon+1} + \frac{\upsilon-1}{\upsilon+1}M^2\right]^{\frac{\upsilon+1}{2/\upsilon-1}}$$

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13. Derive an expression for $\frac{F}{F*} = \frac{PA\left[1+\upsilon M^2\right]}{P^* A^* \left[1+\upsilon\right]}$

$$F = PA + RAC^2$$

$$PC^2 = \upsilon PM^2$$

$$\frac{F}{F*} = \frac{PA\left[1+\upsilon M^2\right]}{P^* A^* \left[1+\upsilon\right]}$$

14. Derive the expression for the area variation in terms of Mach number and velocity variation in terms of Mach number?

$$m = PAC$$

$$\frac{A}{A^*} = \frac{P^*}{P} x \frac{C^*}{C}$$

$$\frac{C^*}{C} = \frac{1}{M}\left[\frac{2}{(\upsilon+1)} + \left(\frac{\upsilon-1}{\upsilon+1}\right)M^2\right]^{\frac{1}{2}}$$

$$\frac{A}{A^*} = \frac{1}{M}\left[\frac{2}{\upsilon+1} + \frac{\upsilon-1}{\upsilon+1}M^2\right]^{\frac{\upsilon+1}{2/\upsilon-1}}$$

15. Derive the expansion for Mach flow rate in terms of Area ratio.

$$m = PAC$$

$$T^* = \left(\frac{2}{\upsilon+1}\right)T_0$$

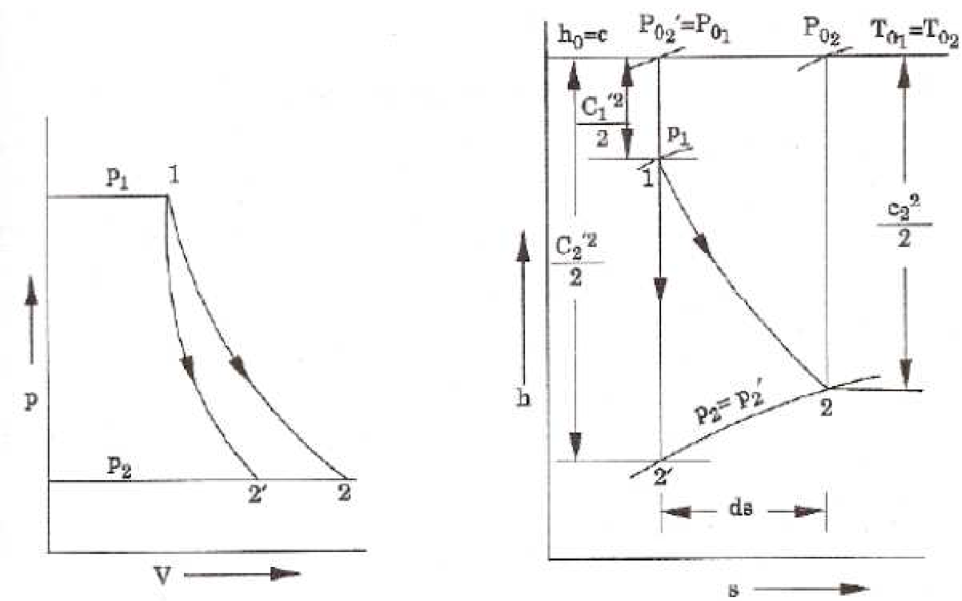
$$P^* = \left(\frac{2}{\upsilon+1}\right)^{\frac{\upsilon}{\upsilon-1}} xP_0$$

$$\frac{m\sqrt{T_0}}{AP_0} = \sqrt{\frac{R}{\upsilon} = \left(\frac{2}{\upsilon+1}\right)^{\frac{\upsilon+1}{2(\upsilon-1)}} x\ \frac{A^*}{A}}$$

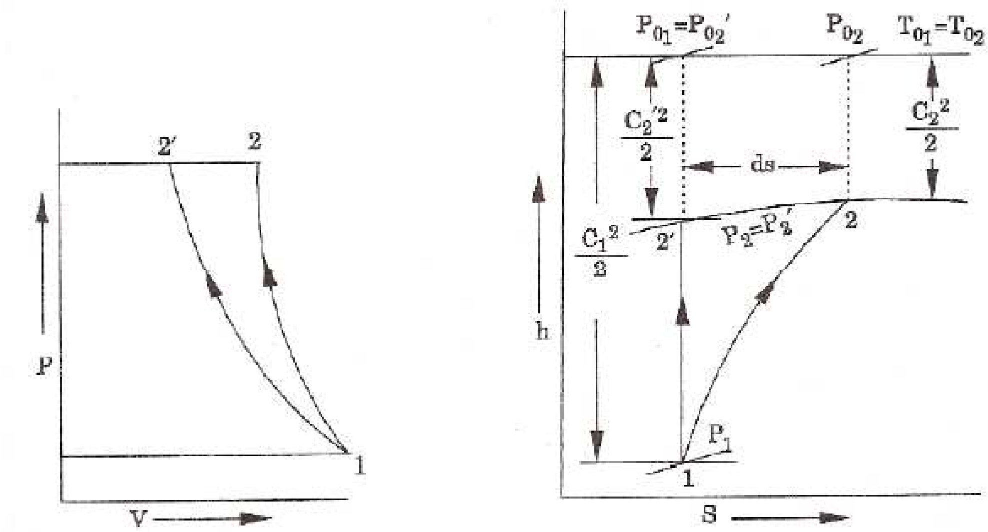
16. Derive the expression fro Mach flow rate in terms of Mach number.

$$\frac{M\sqrt{T_0}}{AP_0}\sqrt{\frac{R}{\upsilon}} = \frac{M}{\left[1+\frac{\upsilon-1}{2}M^2\right]^{\frac{\upsilon+1}{2(\upsilon-1)}}}$$

17. Represent the adiabatic flow through a diffuser on T-S diagram. Label the different states, the initial and final points.



18. Air from a reservoir is discharged through a nozzle. Show the variation of pressure along the axis of the nozzle.



AE53- AERODYNAMICS – II TWO MARK QUESTION AND ANSWERS

1. Define the term "Fanno flow".

A steady one-dimensional flow in a constant area duct with friction in the absence of work and heat transfer is known as **"fanno flow".**

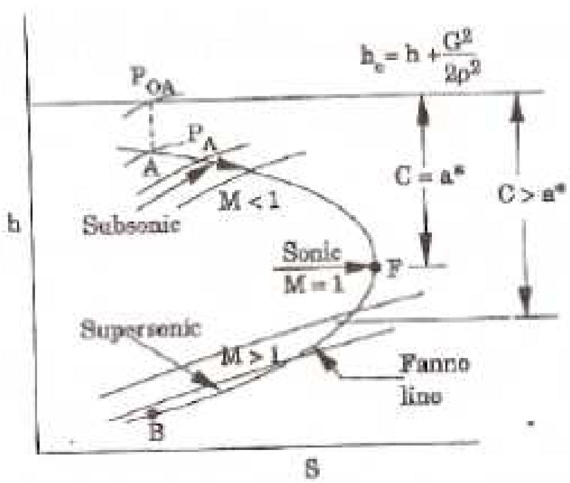
2. Define Fanno line.

The locus of the state which satisfy the continuity and energy equation for a frictional flow is known as **"fanno line".**

3. Give fanno line in h – s diagram with isentropic stagnation line and show various mach number regions.

A to F -	heating process	M < 1
F to A -	cooling process	
B to F -	heating process	M > 1
F to B -	cooling process	

Point F is critical point where mach number M = 1.



The equation which yields the fanno line for the given values of h_0 and G is called "fanno flow equation".

i.e., $h = h_0 - \frac{G^2}{2[f(h,s)^2]} \Rightarrow$ Fanno equation

4. Explain briefly the chocking in fanno flow.

In a fanno line, any heating process (both subsonic and supersonic) will increase the enthalpy, entropy and mass flow rate. This will go upto the limiting state where mach number $M^* = 1$. Further heating is not possible, because the entropy change will be negative which violates the second law of thermodynamics. Hence the mass flow rate is maximum at the critical state and is constant afterwards, then the flow is said to be **"chocked flow".**

5. Give two practical examples where the fanno flow occurs.

Flow occurs in gas ducts of aircraft propulsion engines, flow in air-conditioning ducts and flow of oil in long pipes. etc.

6. Give the effect of increasing the flow length after reaching critical condition in a fanno flow.

The mass flow rate will increase only upto the critical condition and is constant afterwards. Therefore, if the length of pipe is increased afterwards will not give any effect.

7. Write down the expression for the length of duct in terms of the two mach numbers M1 and M2 for a flow through a constant area duct with the influence of friction.

$$\frac{4fL}{D} = \left(\frac{4fL_{max}}{D}\right)_{M_1} - \left(\frac{4fL_{max}}{D}\right)_{M_2}$$

8. Define isothermal flow with friction. Give the applications.

A steady one dimensional flow with friction and heat transfer in a constant area duct is called isothermal flow with friction. Such a flow occurs in long ducts where sufficient time is available for the heat transfer to occur and therefore the temperature may remains constant. Hence the friction factor may be assumed constant along the duct. The applications of isothermal flow are oil or water flow in buried pipe.

9. State assumptions made to derive the equations for isothermal flow.

i. One dimensional flow with friction and heat transfer.
ii. Constant area duct
iii. Perfect gas with constant specific heats and molecular weights
iv. Isothermal flow i.e., the temperature is constant
v. On account of constant temperature the friction factor may be assumed constant along the duct.

10. Differentiate between isothermal flow and fanno flow.

ISOTHERMAL	FANNO FLOW
a) Static temperature is constant	a) Static temperature is not constant
b) With heat transfer.	b) Without heat transfer.
c) Flow occurs in a long ducts where sufficient time is required for heat transfer.	c) Long ducts are not required.
d) On account of constant temperature, the friction factor is assumed as constant.	d) Friction factor is constant.

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11. Define the term "Rayleigh flow".

The one-dimensional flow in a constant area duct with heat transfer and without friction is called **"Rayleigh flow".**

12. Define Rayleigh line.

The locus of the points of properties during a constant area frictionless flow with heat exchange is called **"Rayleigh line".**

13. What is diabatic flow?

It is the flow which deals with the exchange of heat from the system in the absence of friction (Rayleigh flow).

14. Give the assumptions made in Rayleigh flow.

i. Perfect gas with constant specific heats and molecular weight.
ii. Constant area duct,
iii. One dimensional, steady frictionless flow with heat transfer.
iv. Absence of body forces.

15. What do you understand by chocking in Rayleigh flow.

When the fluid is heated in a subsonic region, the entropy increases and the mach number and fluid properties move to the right unitil the maximum entropy is reached where $M^* = 1$. When the fluid is heated in a supersonic region, the entropy increases and the mach number and the fluid properties move to the right until the maximum entropy is reached where $M^* = 1$. Further heating is not possible because, if it is heated the change in entropy is negative which violates the second law of thermodynamics. Therefore, the type of flow when the limiting condition $M^* = 1$ is called **"chocked flow".**

16. Differentiate between Fanno flow and Rayleigh flow.

FANNO FLOW	RAYLEIGH FLOW
a) One dimensional steady frictional flow. b) Stagnation temperature is constant. c) Because of considering the wall friction forces it is accurate. d) Without heat transfer.	a) One dimensional steady frictionless flow. b) Stagnation temperature is not constant c) Less accurate. d) With heat transfer.

17. Air is decelerated from Mach Number 3 to sonic speed in a 4cm inner ia pipe having a friction factor of 0.002. Find the length of the pipe to achieve this deceleration.

Given:

$M_1 = 3$, $M_2 = 1$, $D_1 = 4\text{cm}$, $\overline{f} = 0.002$

Length of the pipe, L = 2.61 m

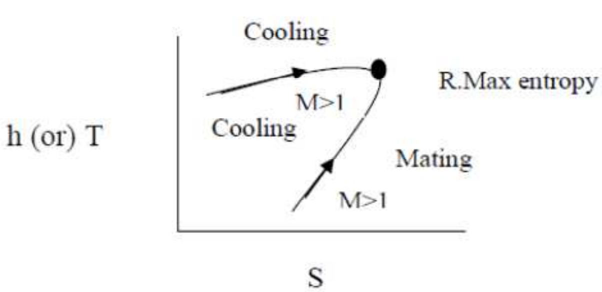
18. Air at stagnation pressure, temperature is 10 bars, 400K is supplied to a 50 mm diameter pipe, the friction factor for the pipe surface is 0.002. If the Mach number changes from 3 at entry to 1 at the exit. Determine the length of the pipe.

Given:

$P_0 = 10$ bar, $T_0 = 400\text{K}$, D = 50 mm, $\overline{f} = 0.002$

L = 3.26 m

19. Sketch the Rayleigh line on h-s or T-S plane and explain the significance of it.



20. Show that $\frac{P_2}{P_2} = \frac{M_1}{M_2}\left[\frac{1+\frac{\upsilon-1}{2}M_1^2}{1+\frac{\upsilon-1}{2}M_2^2}\right]^{\frac{1}{2}}$

$$P = PRT$$

$$\frac{P_2}{P_1} = \frac{P_2 RT_2}{P_1 RT_1}$$

$$\frac{P_2}{P_2} = \frac{M_1}{M_2}\left[\frac{1+\frac{\upsilon-1}{2}M_1^2}{1+\frac{\upsilon-1}{2}M_2^2}\right]^{\frac{1}{2}}$$

21. With a suitable T-S diagram explain Isothermal flow with friction.

One dimensional frictional flow
Flow takes place in constant sectional area
The gas is perfect with constant specific heat
Temperature remains constant

22. Prove that $\frac{P_2}{P_1} = \frac{1+\upsilon M_1^2}{1+\upsilon M_2^2}$

Momentum equation between state 1 & 2

$$P_1A + mC_1 = P_2A + mC_2$$

$$\frac{P_2}{P_1} = \frac{1+\upsilon M_1^2}{1+\upsilon M_2^2}$$

23. Show that static temperature ratio $\frac{T_2}{T_1} = \frac{M_2^2}{M_1^2} x \left[\frac{1+\upsilon M_1^2}{1+\upsilon M_2^2}\right]^2$

$$\frac{M_2}{M_1} = \frac{C_2}{C_1}\left(\frac{T_1}{T_2}\right)^{\frac{1}{2}}$$

$$\frac{T_2}{T_1} = \frac{M_2^2}{M_1^2} x \left[\frac{1+\upsilon M_1^2}{1+\upsilon M_2^2}\right]^2$$

1. What is the normal shock?

When the shock waves are right angles to the direction of flow and the rise in pressure is abrupt are called normal shock waves.

2. What is meant by normal shock as applied to compressible flow?

Compression wave front being normal to the direction of compressible fluid flow. It occurs when the flow is decelerating from supersonic flow. The fluid properties jump across the normal shock.

3. Shock waves cannot develop in subsonic flow? State the reason.

Shocks are introduced to increase the pressure and hence it is a deceleration process. Therefore, shocks are possible only when the fluid velocity is maximum. In a subsonic flow, the velocity of fluid is less then the critical velocity and hence deceleration is not possible. Thus, shock waves cannot develop in subsonic flow.

4. Define strength of a shock wave.

Strength of a shock wave is defined as the ratio of increase in static pressure across the shock to the inlet static pressure.

Strength of shock = $\frac{p_y - p_x}{p_x}$

5. Calculate the strength of shock wave when normal shock appears at M = 2.

From normal shock table M = 2, $\gamma = 1.4$. $\frac{p_y}{p_x} = 4.5$

$\therefore$ Strength of shock = $\frac{p_y}{p_x} - 1 = 4.5 - 1 = 3.5$

6. Define oblique shock where it occurs.

The shock wave which is inclined at an angle to the two dimensional flow direction is called as oblique shock. When the flow is supersonic, the oblique shock occurs at the corner due to the turning of supersonic flow.

7. What are application for moving shock wave?

It is used in

a. Jet engines
b. Shock tubes
c. Supersonic wind tunnel
d. Practical admission turbines.

8. What are properties changes across a normal shock?

1. Stagnation pressure decreases
2. Stagnation temperature remains constant

9. The stagnation pressure -------- and static pressure -------- across a normal shock.

Decreases, Increases

10. Shock wave cannot developing subsonic flow? Why?

In subsonic flow, the velocity of fluid is less than the velocity of sound. Due to this reason, deceleration is not possible in subsonic flow. So shockwaves cannot develop in subsonic flow.

11. The state of a gas ($(\upsilon = 1.3, R = 0.469 kJ/kgK)$ upstream of a normal shock wave is given by the following data M_x=2.5, P_x = 2 bar, T_x = 275K. Calculate the Mach number, pressure, Temperature of the downstream of the shock.

Ans:

$$M_x = 2.5, P_x = 2 \text{ bars}, T_x = 275 \text{ K}$$

Refer Normal shocks takes

$$P_y = 13.87 \times 10^5 \text{ N/m}^2$$

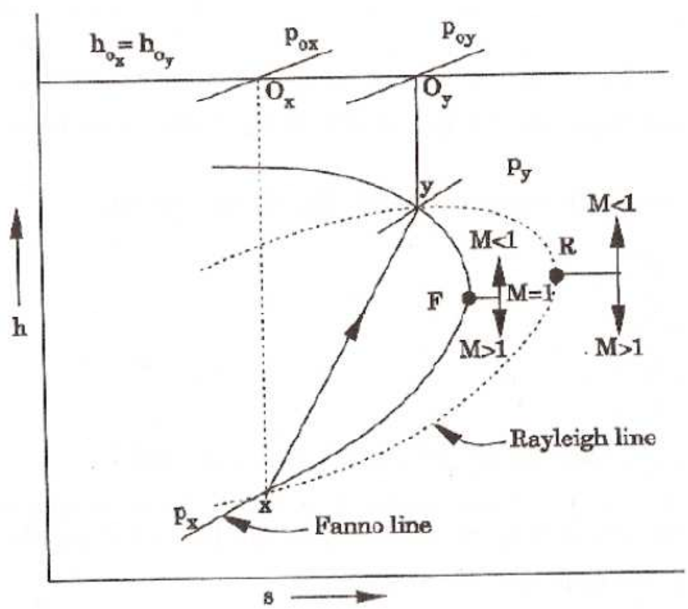
$$T_y = 513.97 \text{ K}$$

$$M_y = 0.493$$

12. Calculate the strength of shock wave when normal shock appears of M = 2.

$$\text{Strength of shock} = P_x$$
$$= 3.5$$

13. **Shown a normal shock in h-s diagram with the help of Rayleigh line and Fanno line.**



14. **What is Prandtl-Meyer relation? What its significance?**

The fundamental relation between gas velocities before and after the normal shock and the critical velocity of sound is known as Prandtl-Meyer relation.

i.e., (i) $c_x \times c_y = a^{*2}$ and (ii) $M^*_x \times M^*_y = 1$

it signifies the velocities (before and after the shock) with the critical velocity of sound and the product of mach numbers before and after the shock is unity.

15. **Give the difference between normal and oblique shock.**

NORMAL SHOCK	OBLIQUE SHOCK
(b) The shock waves are right angles to the direction of flow. (c) May be treated as one dimensional analysis.	(a) The shock waves are inclined at an angle to the direction of flow. (b) Oblique shock is two dimensional analysis.