

Midterm 2 review

-All previous knowledge is assumed

Lift equation

$$L = \frac{1}{2} \rho V^2 C_L S$$

Ideal gas law

$$\rho = \frac{RT}{P}$$

Hydrostatic equation (manometer)

$$\Delta P = -\rho g \Delta h$$

Continuity

$$m_1 = m_2 \quad m = \rho A v \quad \rho_1 A_1 v_1 = \rho_2 A_2 v_2$$

If density is constant:

$$A_1 v_1 = A_2 v_2$$

Bernoulli. (Remember to check your incompressible assumption,  $M < 0.3$ ,  $v < 100$  m/s or  $300$  ft/s)

$$P_1 + \frac{1}{2} \rho v_1^2 = P_2 + \frac{1}{2} \rho v_2^2$$

(If incompressible flow is assumed always say why, don't just write density.)

Elementary thermodynamics

First law of thermodynamics:

States that the change of internal energy equals the heat added to and the work done to the system

$$\delta e = \delta q + \delta w$$

Assumes constant pressure

$$\delta w = -P dV$$

Assumes constant volume

$$\delta q = \Delta h - v dP$$

h - enthalpy  $h = e + PV$

for a constant volume process

$$\delta q = cv dT$$

for a constant pressure process

$$\delta q = c_p dT$$

since  $dV=0$  then

$$\delta e = \delta q$$

$$\delta e = cv dT$$

since  $dP=0$  then

$$\delta q = \Delta h$$

$$\Delta h = c_p dT$$

and

\*Even though these equations were derived based on constant P and V, they hold true for any process if the gas is a perfect gas. The arguments that prove this are beyond the scope of this course

(p. 158 book)

$$\begin{aligned} de &= C_v dT \\ dh &= C_p dT \\ e &= C_v T \\ h &= C_p T \end{aligned}$$

For any process

-Energy equation (relates temperature and velocity)

$$C_p T_1 + \frac{1}{2} v_1^2 = C_p T_2 + \frac{1}{2} v_2^2$$

-Isentropic flow

Assumptions:

Adiabatic: No heat transfer

Reversible: No friction

Isentropic relations

$$\frac{P_2}{P_1} = \left( \frac{\rho_2}{\rho_1} \right)^\gamma \quad \frac{P_2}{P_1} = \left( \frac{T_2}{T_1} \right)^{\gamma/\gamma-1} \quad \left( \frac{\rho_2}{\rho_1} \right)^\gamma = \left( \frac{T_2}{T_1} \right)^{\gamma/\gamma-1}$$

$$\frac{\rho_\infty}{\rho_\omega} = \left( \frac{\rho_\infty}{\rho_\omega} \right)^\gamma \quad \frac{P_0}{P_\omega} = \left( \frac{P_0}{P_\omega} \right)^\gamma$$

$$\frac{P^*}{P_\infty} = \left( \frac{P^*}{P_\omega} \right)^\gamma$$

We cannot assume incompressible; density must be allowed to change (Can't use Bernoulli)

Flight regimes:

M<1 subsonic

M=1 sonic

M>1 supersonic

Subsonic wind tunnels

Most of the times can assume incompressible because we are dealing with low speeds

- Velocity increases as the area decreases through the convergent nozzle, and the opposite occurs for the divergent part

### Different types of pressures

- Static pressure is the pressure we would feel if we were moving along with the flow (Standard atmosphere table)  $\rightarrow P_s, T_s, \rho_s$

Total pressure or stagnation pressure: The pressure obtained at a point where the flow velocity has been decreased to zero.  $V=0$   $M = \frac{V}{a}$   ~~$\sqrt{\rho R T_0}$~~

- It is a property of the flow.

- Constant throughout. (We can use it to find the pressure at other points)

- Since the velocity is zero, Mach number is zero.

- It is measured by a pitot tube

- If we assume incompressible the Bernoulli's equation can be used, which relates dynamic pressure, total pressure, and static pressure.

$$P_0 + \frac{1}{2} \rho v^2 = P_s + \frac{1}{2} \rho v^2$$

$$P_0 - P_s = \frac{1}{2} \rho v^2 \rightarrow \text{dynamic}$$

$$\frac{P_w}{P_a}$$

$$\frac{P_0}{P_{\infty}} = \left[ 1 + \frac{\gamma-1}{2} M_{\infty}^2 \right]^{\frac{1}{\gamma-1}}$$

$$\frac{P_0}{P_w} = \left[ 1 + \frac{\gamma-1}{2} M_{\infty}^2 \right]^{\frac{1}{\gamma-1}}$$

Solving for true airspeed which deals with the actual density.

$$\sqrt{\frac{(P_0 - P_s)^2 / \rho}{P_0}} = V_{true}$$

Equivalent airspeed is the airspeed measured by an airspeed indicator and deals with sea level density. If assuming incompressible flow, then

$$V_{eq} = \sqrt{(P_0 - P_s)^2 / \rho_{SL}}$$

An equation that relates true airspeed and equivalent airspeed:

$$V_{true} = V_{eq} \sqrt{\frac{\rho}{\rho_{SL}}}$$

If the incompressible assumption cannot be made, then one way to find the true velocity is solve for it using  $\text{Mach number} = V/a$  (Be careful not to use  $T_0$  when finding Mach number, remember velocity is zero at the stagnation point, therefore  $M=0$ !)

We can find Mach number using the isentropic Mach relations

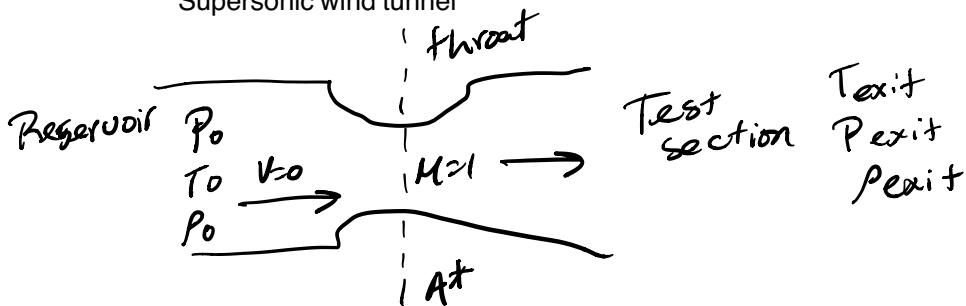
$$\frac{P_0}{P_1} = \left[ 1 + \frac{\gamma-1}{2} M_1^2 \right]^{\frac{1}{\gamma-1}} \quad \frac{P_0}{P_1} = \left[ 1 + \frac{\gamma-1}{2} M_1^2 \right]^{\frac{1}{\gamma-1}}$$

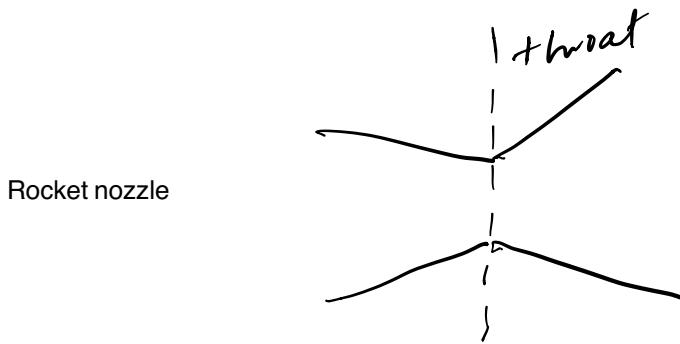
$$\frac{T_0}{T_1} = 1 + \frac{\gamma-1}{2} M_1^2$$

### Supersonic wind tunnels

- For the velocity to increase the area must increase

Supersonic wind tunnel





Reservoir:  $P_0, T_0, \rho_0$  (flow going into the wind tunnel)

Test section:  $P_{exit}, T_{exit}, \rho_{exit}$  (flow going out).

Area Mach relations

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

If we are given a Mach number or an area ratio, we can get either of those from the table

- Be careful when using the table, since your result will depend on if the flow is subsonic or supersonic
- It's important to know that a throat is the point where the smallest area of a wind tunnel or rocket nozzle can be found, but having a throat does not necessarily mean that you have a choke point where  $M=1$ .
- $A^*$  which is the area where  $M=1$  can be thought as a property of the flow like  $P_0, T_0$  and  $\rho_0$ . Even if we do not physically have it, we can still solve for it, and use this value to find other variables.
- You will see indications that wind tunnel has a physical choke point (e.g. If you are told that the flow goes from subsonic to supersonic)
- Even if there is throat, if the  $M$  is not 1 at this point, the flow can stay subsonic or supersonic.

~~Ain~~  
~~Aout~~

4.11 p. 280..Anderson book

The mass flow of air through a supersonic nozzle is 1.5 lbm/s. The exit velocity is 1500 ft/s, and the reservoir temperature and pressure are 1000°R and 7 atm, respectively. Calculate the area of the nozzle exit. For air,  $c_p = 6000 \text{ ft} \cdot \text{lb}/(\text{slug})(^\circ\text{R})$ .

4.14 .Calculate the Mach number at the exit of the nozzle in Prob. 4.11

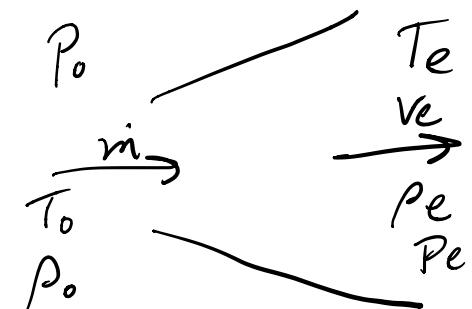
$$m = 1.5 \text{ lbm/s} = 0.0466214 \text{ kg/s}$$

$$V_e = 1500 \text{ ft/s}$$

$$T_0 = 1000^\circ\text{R}$$

$$P_0 = 7 \text{ atm} \cdot 2116.2 \frac{\text{lb}}{\text{ft}^2} = 14,813.4 \frac{\text{lb}}{\text{ft}^2}$$

$$c_p = 6000 \text{ ft} \cdot \text{lb}/(\text{slug})(^\circ\text{R})$$



$$A_e = ?$$

$$m = P_e A_e$$

$$V_e = A_e$$

$$\left(\frac{P_e}{P_0}\right)^{\frac{1}{\gamma}} = \left(\frac{T_e}{T_0}\right)^{\frac{1}{\gamma-1}}$$

$$P_0 = RT_0 \rho_0$$

$$\frac{P_0}{RT_0} = \rho_0$$

$$\frac{14,813.4}{1716 \cdot 1000} = 0.0086 \text{ slug/ft}^3$$

$$C_p T_0 + \cancel{\frac{1}{2} V_e^2} = C_p T_e + \frac{1}{2} V_e^2$$

$$T_e = \underline{C_p T_0 - \frac{1}{2} V_e^2}$$

$$T_e = \frac{C_p}{6000 \cdot 1800} - \frac{1}{2} \cdot 1500^2$$

$$T_e = 812.5^\circ R$$

$$\rho_e = \left[ \left( \frac{T_e}{T_0} \right)^{\gamma / (r-1)} \right]^{\gamma} \cdot \rho_0$$

$$\rho_e = 0.0051 \text{ slug/ft}^3$$

$$m = \rho_e A_e V_e$$

$$\frac{0.0466214}{0.0051 \cdot 1500} = A_e$$

$$A_e = 0.0061 \text{ ft}^2$$

$$M_e = \frac{V_e}{a_e}$$

$$M_e = \frac{1500}{\sqrt{1.4 \cdot 1716 \cdot 812.5}}$$

$$K_e = \frac{V_e}{\sqrt{r R T_e}}$$

$$M_e = 1.07$$

4.21. Given: low-speed  $h = 8,000 \text{ ft}$   $P_0 = 1650 \text{ lb/ft}^2$

From S.A.T.:  $P_{\infty} = 1.5721 \cdot 10^3 \frac{\text{lb}}{\text{ft}^2}$   $T_{\infty} = 500^\circ \text{R}$   $\gamma = 1.4$   
 $R = 1716 \frac{\text{ft} \cdot \text{lb}}{\text{slug} \cdot ^\circ \text{R}}$

Determine if incompressible:

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

$$M = \left\{ \frac{2}{1.4-1} \left[ \left( \frac{1572.1}{1650} \right)^{\frac{1}{1.4}} - 1 \right] \right\}^{\frac{1}{2}} = 0.2638$$

$M < 0.3$ ; incompressible assumption is OKAY ✓

Using Bernoulli:

$$P_0 + \frac{1}{2} \rho_0 V_0^2 = P_{\infty} + \frac{1}{2} \rho_{\infty} V_{\infty}^2 \Rightarrow V_{\infty} = \left[ \frac{2(P_0 - P_{\infty})}{\rho_{\infty}} \right]^{\frac{1}{2}}$$

$$P_{\infty} = \rho_{\infty} R T_{\infty} \Rightarrow P_{\infty} = \frac{P_0}{\gamma T_{\infty}} \Rightarrow P_{\infty} = \frac{1572.1}{(1716)(500)} = 0.001832 \frac{\text{slug}}{\text{ft}^3} \quad \left. \begin{array}{l} \text{Freestream} \\ \text{density} \end{array} \right\}$$

$$V_{\infty} = \left[ \frac{2(1650 - 1572.1)}{0.001832} \right]^{\frac{1}{2}} = \boxed{291.600 \frac{\text{ft}}{\text{s}} = V_{\infty}}$$

OR Using Mach Number

$$M = \frac{V_{\infty}}{a} \Rightarrow V_{\infty} = Ma ; a = \sqrt{\gamma R T} ; \Rightarrow V_{\infty} = M \sqrt{\gamma R T}$$

$$V_{\infty} = (0.2638) \sqrt{(1.4)(1716)(500)} = \boxed{289.079 \frac{\text{ft}}{\text{s}} = V_{\infty}}$$

REGARDLESS,  $V_{EAS} = V_{TAS} \left( \frac{P_0}{P_{SL}} \right)^{\frac{1}{2}}$  ;  $P_{SL} = 2.3769 \cdot 10^{-3} \frac{\text{slug}}{\text{ft}^3}$  incompressible flow

Bernoulli:

$$V_{EAS} = (291.600) \left( \frac{1.832 \cdot 10^{-3}}{2.3769 \cdot 10^{-3}} \right)^{\frac{1}{2}} = 256.02 \frac{\text{ft}}{\text{s}}$$

$$\boxed{V_{EAS} = 256.02 \frac{\text{ft}}{\text{s}}}$$

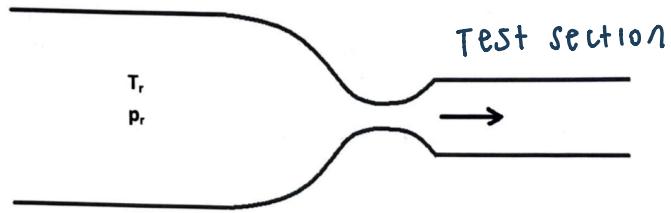
Mach:

$$V_{EAS} = (289.079) \left( \frac{1.832 \cdot 10^{-3}}{2.3769 \cdot 10^{-3}} \right)^{\frac{1}{2}} = 253.81 \frac{\text{ft}}{\text{s}}$$

$$\boxed{V_{EAS} = 253.81 \frac{\text{ft}}{\text{s}}}$$

**Question:** A nozzle needs to be designed for a supersonic wind tunnel with air as the primary fluid. The test section specifications are provided below:

- 1) Mach # = 4.92
- 2) Diameter = 13 cm
- 3) Static pressure = 54.05 kPa
- 4) Static temperature = 255.7 K



Determine the following wind tunnel characteristics to achieve the specified test section flow properties:

- a) Mass flow provided
- b) Nozzle throat area
- c) Reservoir temperature ( $T_r$ )
- d) Reservoir pressure ( $p_r$ )

Assume isentropic flow in the nozzle and neglect frictional effects. Assume that propane behaves as a perfect gas with  $\gamma=1.4$  and  $R=0.2870 \text{ kJ/kg}\cdot\text{K}$ .

Air

At test section:

$$M = 4.92$$

$$T = 255.7 \text{ K}$$

$$D = 0.13 \text{ m}$$

$$R = 287 \text{ J/kg}\cdot\text{K}$$

$$\rho = 54050 \text{ Pa} \quad \gamma = 1.4$$

a) Mass flow @ Test section

$$\dot{m} = \rho A V$$

$$\rho : P = \rho RT \rightarrow \rho = \frac{P}{RT}$$

$$A : A = \pi r^2, r = \gamma_2 D \rightarrow A = \gamma_4 \pi D^2$$

$$V : M = \frac{V}{a}, a = \sqrt{\gamma RT} \rightarrow V = M \sqrt{\gamma RT}$$

$$\dot{m} = \frac{P}{RT} \cdot \gamma_4 \pi D^2 \cdot M \sqrt{\gamma RT}$$

↓  
Evaluate

$$\dot{m} = 15.417 \text{ kg/s}$$

b) Nozzle throat area  $A^*$

$$\left(\frac{A}{A^*}\right)^2 = \frac{1}{m^2} \left[ \frac{2}{\gamma+1} \left( 1 + \frac{\gamma-1}{2} m^2 \right) \right]^{\frac{\gamma+1}{\gamma-1}}$$

$$A^* = A \cdot \left\{ \frac{1}{m^2} \left[ \frac{2}{\gamma+1} \left( 1 + \frac{\gamma-1}{2} m^2 \right) \right]^{\frac{\gamma+1}{\gamma-1}} \right\}^{-1/2}$$

↓ Evaluate

$$A^* = 5.662 \times 10^{-4} \text{ m}^2$$

c) Reservoir Temperature  $\rightarrow$  Stagnation Temperature

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

$$T_0 = T \left( 1 + \frac{\gamma - 1}{2} M^2 \right)$$

$\downarrow$  Evaluate

$$T_r = T_0 = 1493.615 \text{ K}$$

d) Reservoir Pressure  $\rightarrow$  Stagnation Pressure

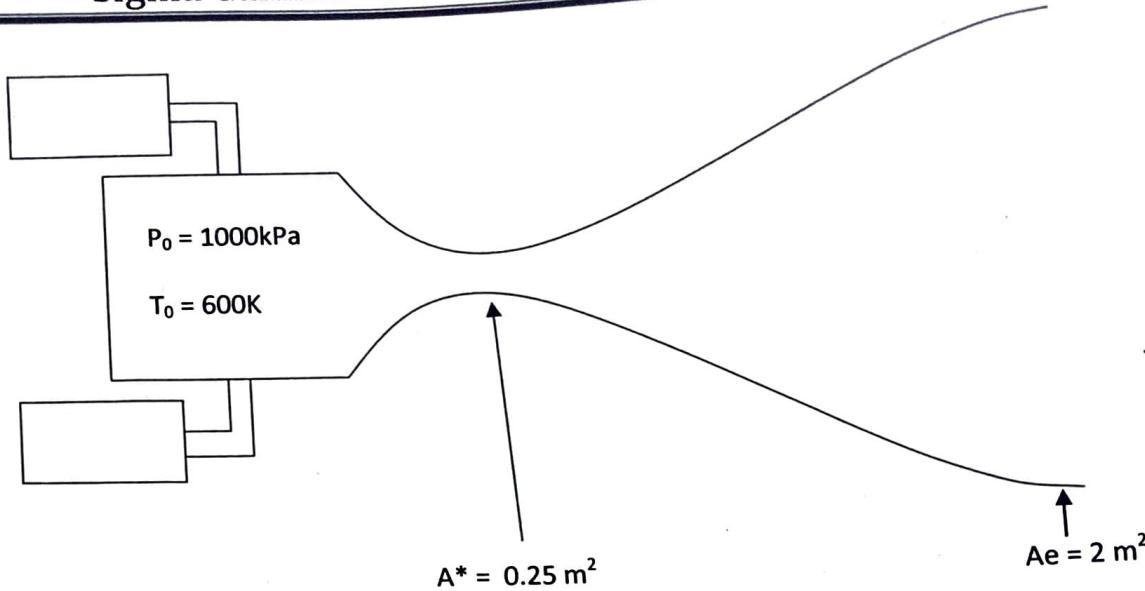
$$\frac{P_0}{P} = \left( 1 + \frac{\gamma - 1}{2} M^2 \right)^{\frac{\gamma}{\gamma - 1}}$$

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

$\downarrow$  Evaluate

$$P_r = P_0 = 26035971.3 \text{ Pa}$$

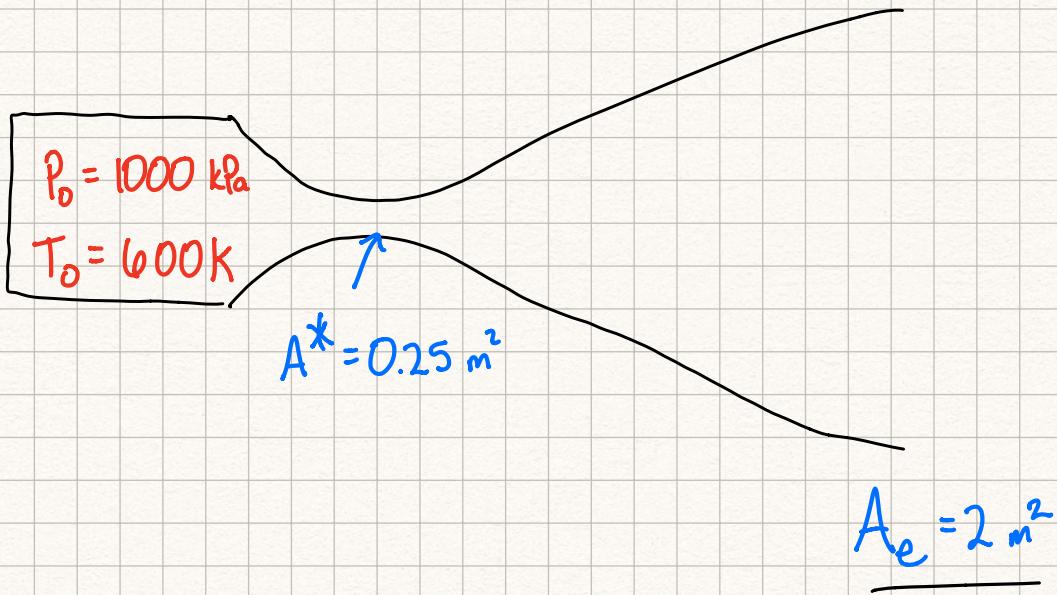
$$26035.9713 \text{ kPa}$$



The rocket motor shown above is designed to operate most efficiently at an altitude of 55,000 feet, producing approximately 87,000 pounds of thrust at this condition. From the conditions given, determine the following:

- The exit mach number
- The exit pressure
- The exit temperature
- The exit velocity

You may assume that gamma is constant, and the flow is isentropic.



a. Find Exit Mach Number

$$\left(\frac{A_e}{A^*}\right)^2 = \frac{1}{M_e^2} \left[ \frac{2}{\gamma + 1} \left( 1 + \frac{\gamma - 1}{2} M_e^2 \right) \right]^{\frac{\gamma+1}{\gamma-1}}$$

$$\frac{A_e}{A^*} = \frac{2 \text{ m}^2}{0.25 \text{ m}^2} = 8$$

M	A/A*	T/T <sub>o</sub>	p/p <sub>o</sub>	P/P <sub>o</sub>	T/T*	p/p*	P/P*	u/u*
1.9	1.55526	0.5807	0.1492	0.257	0.6969	0.2825	0.4054	1.5861
1.95	1.61931	0.568	0.1381	0.2432	0.6816	0.2615	0.3836	1.6099
2	1.6875	0.5556	0.1278	0.23	0.6667	0.2419	0.3629	1.633
2.05	1.75999	0.5433	0.1182	0.2176	0.652	0.2238	0.3433	1.6553
2.1	1.83694	0.5313	0.1094	0.2058	0.6376	0.207	0.3246	1.6769
2.15	1.91854	0.5196	0.1011	0.1946	0.6235	0.1914	0.307	1.6977
2.2	2.00497	0.5081	0.0935	0.1841	0.6098	0.177	0.2903	1.7179
2.25	2.09644	0.4969	0.0865	0.174	0.5963	0.1637	0.2745	1.7374
2.3	2.19313	0.4859	0.08	0.1646	0.5831	0.1514	0.2596	1.7563
2.35	2.29528	0.4752	0.074	0.1556	0.5702	0.14	0.2455	1.7745
2.4	2.4031	0.4647	0.0684	0.1472	0.5576	0.1295	0.2322	1.7922
2.45	2.51683	0.4544	0.0633	0.1392	0.5453	0.1198	0.2196	1.8092
2.5	2.63672	0.4444	0.0585	0.1317	0.5333	0.1108	0.2077	1.8257
2.55	2.76301	0.4347	0.0542	0.1246	0.5216	0.1025	0.1965	1.8417
2.6	2.89598	0.4252	0.0501	0.1179	0.5102	0.0949	0.1859	1.8571
2.65	3.03588	0.4159	0.0464	0.1115	0.4991	0.0878	0.176	1.8721
2.7	3.18301	0.4068	0.043	0.1056	0.4882	0.0813	0.1665	1.8865
2.75	3.33766	0.398	0.0398	0.0999	0.4776	0.0753	0.1576	1.9005
2.8	3.50012	0.3894	0.0368	0.0946	0.4673	0.0698	0.1493	1.914
2.85	3.67072	0.381	0.0341	0.0896	0.4572	0.0646	0.1414	1.9271
2.9	3.84977	0.3729	0.0317	0.0849	0.4474	0.0599	0.1339	1.9398
2.95	4.0376	0.3649	0.0293	0.0804	0.4379	0.0556	0.1269	1.9521
3	4.23457	0.3571	0.0272	0.0762	0.4286	0.0515	0.1202	1.964
3.5	6.78962	0.2899	0.0131	0.0452	0.3478	0.0248	0.0714	2.0642
4	10.7188	0.2381	0.0066	0.0277	0.2857	0.0125	0.0436	2.1381
4.5	16.5622	0.198	0.0035	0.0174	0.2376	0.0065	0.0275	2.1936
5	25	0.1667	0.0019	0.0113	0.2	0.0036	0.0179	2.2361
5.5	36.869	0.1418	0.0011	0.0076	0.1702	0.002	0.012	2.2691
6	53.1798	0.122	0.0006	0.0052	0.1463	0.0012	0.0082	2.2953
6.5	75.1343	0.1058	0.0004	0.0036	0.127	0.0007	0.0057	2.3163
7	104.143	0.0926	0.0002	0.0026	0.1111	0.0005	0.0041	2.3333
7.5	141.841	0.0816	0.0002	0.0019	0.098	0.0003	0.003	2.3474
8	190.109	0.0725	0.0001	0.0014	0.087	0.0002	0.0022	2.3591
8.5	251.086	0.0647	7E-05	0.0011	0.0777	0.0001	0.0017	2.3689
9	327.189	0.0581	5E-05	0.0008	0.0698	9E-05	0.0013	2.3772
9.5	421.131	0.0525	3E-05	0.0006	0.063	6E-05	0.001	2.3843
10	535.938	0.0476	2E-05	0.0005	0.0571	4E-05	0.0008	2.3905

$$\frac{M}{A} \frac{A}{A^k}$$

$$\frac{3.5}{M_e} \frac{6.78}{8} \frac{10.71}{10.71}$$

Figure 2: Table of ratios for isentropic duct flows.

$$\frac{M_e - 3.5}{4 - 3.5} = \frac{8 - 6.78}{10.71 - 6.78}$$

$$M_e = \left( \frac{8 - 6.78}{10.71 - 6.78} \right) (4 - 3.5) + 3.5 = \boxed{3.65 = M_e}$$

b. Find Exit Pressure.

$$\frac{P_e}{P_0} = \left( 1 + \frac{r-1}{2} M_e^2 \right)^{\frac{-r}{r-1}}$$

$$P_e = \underbrace{1000 \text{ kPa}}_{\text{at } P_0} \left( 1 + \frac{r}{2} (3.65)^2 \right)^{\frac{-1.4}{r}}$$

$$\boxed{P_e = 10.62 \text{ kPa}}$$

c. Find Temperature at exit

$$\frac{T_e}{T_0} = \left( 1 + \frac{r-1}{2} M_e^2 \right)^{-1}$$

$$T_e = \underbrace{600 \text{ K}}_{\text{at } T_0} \left( 1 + \frac{r}{2} (3.65)^2 \right)^{-1}$$

$$\boxed{T_e = 163.7 \text{ K}}$$

Could also use tables.

Could also relate  $\frac{T_e}{T_0}$  and  $\frac{P_e}{P_0}$

d. Find exit velocity.

Definition of exit Mach number

$$M_e = \frac{v_e}{a_e}$$

$$v_e = M_e a_e$$

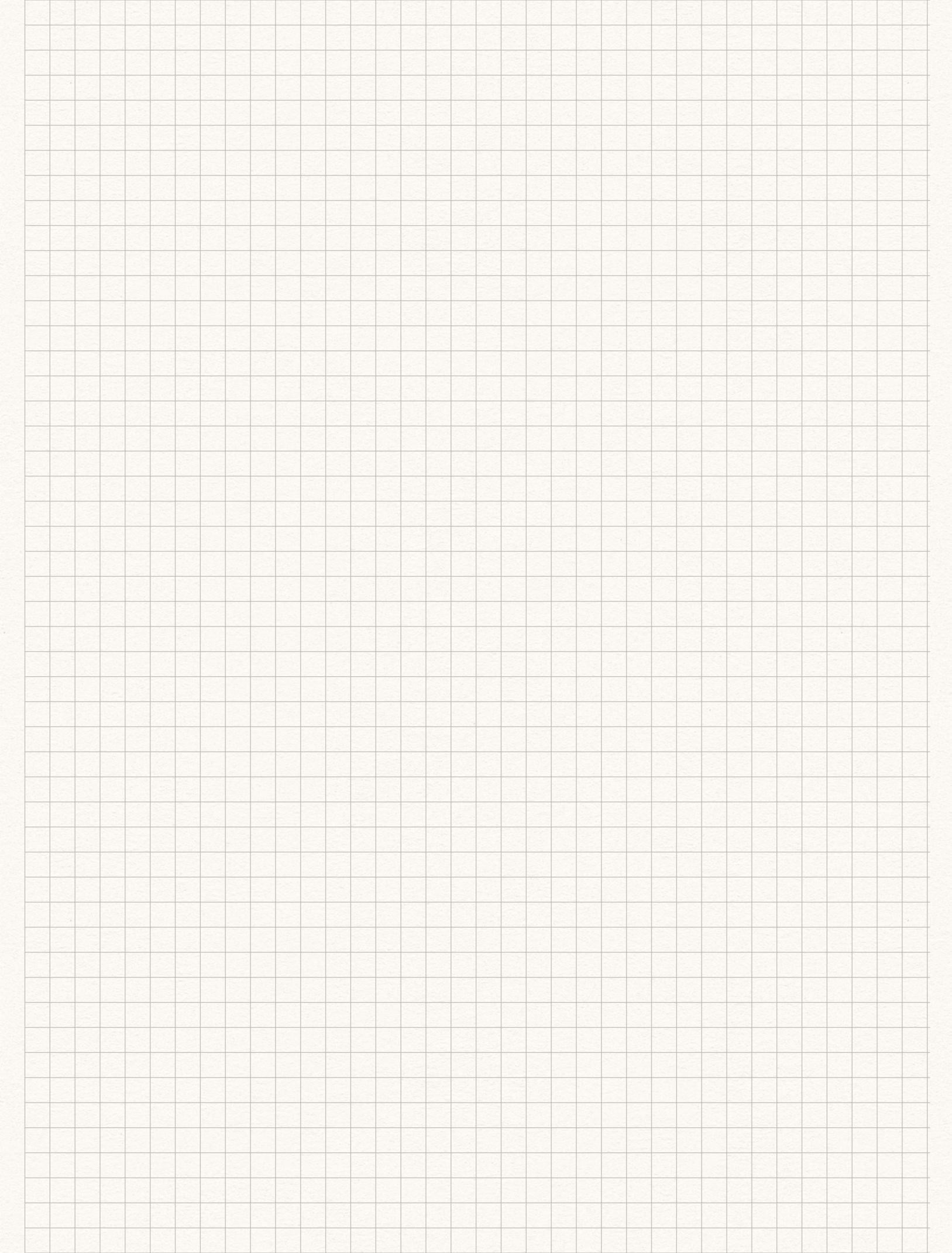
$$= M_e \sqrt{\gamma R T_e}$$

$$= 3.65 \sqrt{(1.4)(287 \frac{\text{N}\cdot\text{m}}{\text{kg}\cdot\text{K}})(163.7 \text{ K})}$$

$$= 3.65 \sqrt{65775 \frac{\text{N}\cdot\text{m}}{\text{kg}}}$$

$$\frac{\text{N}\cdot\text{m}}{\text{kg}} = \frac{\text{kg} \cdot \text{m} \cdot \text{m}}{\text{kg} \cdot \text{s}^2} = \frac{\text{m}^2}{\text{s}^2}$$

$$\boxed{v_e = 936 \frac{\text{m}}{\text{s}}}$$



# Sigma Gamma Tau AAE 200 Midterm 1 Review

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**Solution:**

\*Note that the thrust data is irrelevant to the problem at hand, but accurately matches the performance of the design for the data given.

(a)

We begin by solving the area relation formula for the exit mach number:

$$\left(\frac{A_e}{A_*}\right)^2 = \frac{1}{M_e^2} \left[ \frac{2}{\gamma + 1} \left( 1 + \frac{\gamma - 1}{2} M_e^2 \right) \right]^{\frac{\gamma+1}{\gamma-1}} \rightarrow M_e = 3.677$$

(b)

Use the isentropic relation to find the exit pressure:

$$\frac{P_e}{P_0} = \left( 1 + \frac{\gamma - 1}{2} M_e^2 \right)^{\frac{-\gamma}{\gamma-1}} \rightarrow P_e = 10.22 \text{ kPa}$$

(c)

Again, use an isentropic relation to determine the exit temperature:

$$\frac{T_e}{T_0} = \left( 1 + \frac{\gamma - 1}{2} M_e^2 \right)^{-1} \rightarrow T_e = 161.98 \text{ K}$$

(d)

Finally, use the speed of sound formula to determine the exit velocity:

$$V_e = M_e \sqrt{\gamma R T_e} \rightarrow V_e = 938.06 \frac{\text{m}}{\text{s}}$$