MAE 112 Fall 2024
Homework Assignment #1

Due: midnight, Sunday, October 13, 2024
Follow submission instructions from TAs Andrew Nichols and Wes Hellwig

You will see some problems posed with metric units and others with British units. In engineering practice, you will have to contend with both.

1. A rocket uses propellants with a specific impulse equal to 270 seconds. It has an empty weight (i.e., without propellants) of 200 lbf (measured at sea level). The desire is to accelerate it from rest to a velocity of 8,000 feet per second. Assume that drag forces are negligible and gravity acts normal to the flight direction.

(a) What is the required mass of propellant?

(b) If the rocket had an initial horizontal velocity of 1000 feet per second (e.g., with a launch from an aircraft), how much propellant mass is required to achieve the 8,000 feet per second speed?

(c) If the specific impulse were increased to the new value of 290 seconds, what would be the answer to part (a)?

2. Consider a rocket at takeoff from Earth at sea level. The mass flux of propellants is 160 kg/s. The hot gas exits the nozzle at a velocity of 1200 m/sec and a pressure of 0.85 atmospheres through an exit area of one square meter. Ambient pressure is one atmosphere. What is the thrust magnitude? If the initial acceleration is 50 m/sec^2, what is the initial mass of the vehicle including propellants?

3. Consider an air-breathing jet engine which is flying at a velocity of 600 feet per second. For every lbm/second of air mass flow, a 0.030 lbm/sec mass flow of fuel is injected into the engine. The thrust force is 5000 lbf.; the entrance pressure equals the ambient pressure at the altitude (0.5 atm) and the exhaust pressure is 0.75 times the ambient atmospheric pressure. The incoming air temperature is 525^{o}R. The entrance area and exhaust areas are both ten square feet. Determine:

(a) exhaust velocity

(b) specific fuel consumption

4. For an air-breathing jet engine, specific thrust is defined as thrust divided by air mass flow rate. Consider an engine that is flying at a velocity equal to 250 meters per second. For every kilogram/second of air mass flow, 0.040 kgm/sec mass flow of fuel is injected into the engine. The exit pressure and the entrance pressure both equal the ambient pressure, which is 0.7 atm. What must be the value of the exhaust velocity u_e if a specific thrust equal to 400m/s is desired?

END

MAE 112 Fall 2024
Homework Assignment #2

Due: midnight, Sunday, October 20, 2024
Follow submission instructions from TAs Andrew Nichols and Wes Hellwig

1. Calculate theoretical (ideal) flame temperature for methane in stoichiometric ratio with enriched air (50% O_2, 50% N_2 by volume). Pressure is constant at 15 atm and the initial temperature is 298 K.

2. H_2O exists as the major product of combustion for hydrogen and oxygen originally in stoichiometric proportions with no other gas present. Find the fraction of products dissociated to H_2 and O_2 in each of the following conditions for the products:
   (a) 1850 º R and one atmosphere;
   (b) 1850 º R and 10 atmospheres;
   (c) 3100 º R and 10 atmospheres; and
   (d) 1850 º R and one atmosphere but now with hydrogen and air originally.

3. (a) Calculate AF at stoichiometric condition (AF_{st}) for ethyl alcohol C_2H_5OH (aka ethanol) initially at 550 º R burning in air at 20 atmospheres of pressure. AF is the ratio of mass flow of air to mass flow of fuel. Also, calculate FA = 1/ AF for the same condition.
   (b) Calculate AF and $\Phi = AF_{st} / AF = FA / FA_{st}$ for ethyl alcohol and 50% excess air at the same conditions.

4. (a) Establish the equations which can be employed for the calculation of the equilibrium composition and the flame temperature when one mole of propane C_3H_8 burns adiabatically at a constant pressure of ten atmospheres. The mixture is lean with 75% excess air. Both air and fuel enter at a temperature of 800°R. Consider the products to be CO_2, CO, H_2O, H_2, O_2, and N_2 only. Write all the required equations with known quantities and parameters substituted into the equation. Identify the unknowns. Propane is gaseous at room temperature. Explain what would be different in the analysis if propane entered at a lower temperature in liquid form.
   (b) Use the computer software to calculate the final flame temperature and concentrations of the products with the gaseous propane fuel.
   (c) For the adiabatic situation with gaseous fuel described in Part a, establish the equations to solve for the theoretical (ideal) temperature and composition. What are the products in this case? Again, write the necessary equations, identify the known quantities, and identify the unknowns. Solve the equations for the final temperature and composition. Which of the two temperatures from 2b and 2c is larger? Why?

5. Do a preliminary design on a rocket combustor. A peak temperature of 4500 ºR is desired and the average molecular weight of the hot products should be no greater than 27. Choose an appropriate fuel that will burn with oxygen (O_2). Candidates are ethanol (C_2H_5OH), hydrogen (H_2), and methane (CH_4). Design to have the theoretical (ideal) flame temperature (without dissociation) at 4500 ºR. Determine the proper mixture ratio of fuel to oxygen for that case. Then, calculate the equilibrium temperature which will be a little lower. Estimate the mass

fractions of the products at equilibrium. Determine the average molecular weight of the equilibrium products.

END.

MAE 112 Fall 2042
Homework Assignment #3
Due: 11:59 pm, Sunday, October 27, 2024
Follow submission instructions from TAs

1. A rocket nozzle has initial pressure and temperature of fifty atmospheres and 5000°R with γ = 1.25; c_p = 0.30 Btu/1bm °R; and A* = 1.5 ft^2. The flow is slightly over-expanded to a Mach number M_e = 3.5 at the nozzle exit with the ambient pressure at 0.50 atmosphere. Assume 95% for nozzle polytropic efficiency. Calculate: (a) the characteristic velocity c*; (b) the mass flow; (c) nozzle exit pressure and cross-sectional area (*beware of tables and graphs constructed for air flow*); (d) nozzle exit velocity U; and (e) effective exhaust velocity c.

2. Consider a nozzle with initial upstream entry pressure and temperature of thirty atmospheres and 4000°R. The value of γ = 1.2 and the value of c_p = .30 Btu/1bm °R. The throat area is 0.75 ft^2. The flow is perfectly expanded to the ambient pressure of 0.70 atmospheres. Calculate: (a) the mass flow, (b) the exhaust velocity, (c) the exit area, and (d) the thrust coefficient.

3. Consider a rocket engine that uses liquid oxygen and liquid ethanol (C_2H_5OH) fuel aka ethyl alcohol. The oxygen mass-flow rate is 2.0 times greater than the fuel mass-flow rate. Ethanol is stored at 298K while the oxygen is stored at 80K just slightly below its boiling point. Oxygen has a heat of vaporization of 6.81kJ/mole while the value for ethanol is 38.6 kJ/mole. The heat of formation of liquid ethanol is -277.0 kJ/mole. The specific heat at constant pressure for gaseous oxygen is 30.77 joules/mole °K. The liquids are sprayed into the combustion chamber.

(a) How much energy per mole is required to vaporize and heat a mole of oxygen to the temperature of 298K.

(b) What is the expected ideal flame temperature? The ideal flame temperature aka theoretical flame temperature is the value with no dissociation. Assume that the products are H_2O, CO_2, and CO. You should make this calculation without using the online computer code. A key step is to determine what fraction of the carbon will appear in CO_2 and what fraction will appear in CO.

4. Suppose we have a rocket combustor that has hot products produced at the following conditions:

T = temperature = 4600 º R
P = pressure = 75 atmospheres
γ = ratio of specific heats = 1.25
MW = average molecular weight = 27

(a) Design a nozzle that will produce 75,000 pounds of thrust with an ambient pressure of one atmosphere. In particular, determine the following quantities: mass flow rate, exit pressure, exit or exhaust velocity, effective exhaust velocity, thrust coefficient, throat cross-sectional area, and exit cross-sectional area.

(b) Design a nozzle that produces 100,000 pounds of thrust with an ambient pressure at vacuum conditions. Limit the nozzle exit cross-sectional area to no more than thirty times the throat cross-sectional area. Determine the same quantities as described in Part (a).

5. Consider a jet engine flying at a Mach number of 1.4. A normal shock sits at the entrance of the divergent diffuser. The diffuser entrance cross-sectional area is 2.5 ft^2. The ambient conditions are 500°R for temperature and 0.8 atmosphere for pressure.

(a) What is the stagnation pressure immediately in front (upstream) of the shock? What is the stagnation pressure immediately behind (downstream) the shock? What is the Mach number immediately behind the shock? What is the mass flow through the diffuser?

(b) What is the minimum cross-sectional area required at the downstream end of the diffuser in order to assure that the Mach number of the flow there does not exceed 0.10?

MAE 112 Fall 2024
Homework Assignment #4
Due: 11:59 pm, Sunday, November 3, 2024
Follow submission instructions from TAs

1. . Compare a normal shock with an oblique shock. Suppose the inflowing velocity of the air had a Mach number of 2.0 at a temperature of 250 K and an ambient pressure of 0.70 atm.
(a) With the normal shock, determine the pressure, stagnation pressure, temperature, velocity, and Mach number behind (downstream of) the shock.
(b) Suppose we aim for a downstream stagnation pressure that is 15% higher than the value found in part (a). What is the angle of oblique shock here to the incoming velocity vector? Use the charts from Chapter 3, making the best interpolations you can.
(c) Determine the downstream values for the temperature, Mach number, velocity component normal to the oblique shock, and velocity component parallel to the oblique shock.

2. Consider a Kantrowitz-Donaldson diffuser designed for a flight Mach number of 1.75. The entrance area equals 1.5 ft^2 and the ambient air temperature and pressure are 500°R and 0.7 atmosphere. The flow is isentropic everywhere except across the normal shockwave. Determine:
(a) the minimum cross-sectional area of the throat such that a normal shock may be stabilized at the entrance,
(b) the maximum mass flow, and
(c) the maximum stagnation pressure possible at the end of the diffuser (with subsonic flow only in the divergent portion).
In each of these optimizations, consider the flight Mach number fixed at the design value while the final pressure (at the end of the diffuser) is allowed to adjust.

3. Consider a ramjet in flight at a Mach number of 2.75 with ambient conditions at 298 K and 0.9 atmosphere of pressure. The air capture area is 0.70 square meters. The inlet design involves first a wedge that deflects the stream by an angle of 15 degrees followed by a Kantrowitz-Donaldson (K-D) diffuser. Operation is at design conditions except for part (h).
(a) What is the mass flow through the ramjet?
(b) What is the stagnation temperature for that flow through the inlet / diffuser?
(c) What are the stagnation-pressure values ahead of and immediately behind the first shock?
(d) What is the flow Mach number immediately behind the first shock? What is the flow Mach number at the entrance to the K-D diffuser?
(e) What is the Mach number at the diffuser throat?
(f) What is the final stagnation pressure?
(g) Determine the value of the polytropic efficiency for this inlet design.
(h) Determine the polytropic efficiency value for a shock at the entrance of the K- D diffuser.

4. Suppose a particular compressor has a compression ratio $P_3/P_2 = 25$; the incoming air temperature is 300 K and its pressure is 1.2 atm. 20 kgm per sec. of air flows through the compressor.
(a) If the adiabatic efficiency is 90%, what is the final temperature?

(b) What is the power required?

(c) What is the minimum number of stages (pairs of rotor and stator sections) required to protect against separation due to adverse pressure gradients?

5. Do a preliminary design on a ramjet engine which produces 5000 1bf of thrust. Size constraints limit the intake cross-sectional capture area to 0.80 square feet. The engine is intended to cruise at a Mach number of 2.5. Assume ambient air conditions are one atmosphere of pressure and 500°R. Indicate your choices of inlet type, fuel, temperature at entrance to nozzle, and extent of expansion in nozzle. All choices must be rational and defensible, of course. Indicate mass flows of air and fuel, mixture ratio(s), thrust specific fuel consumption, exhaust velocity, stagnation pressure ratios across each component, throat area, nozzle exit area, and nozzle exit pressure.

END

MAE 112 Fall 2024
Homework Assignment #5
Due: 5 pm, Sunday, November 10, 2024
Follow submission instructions from TAs

1. Consider one-stage of a compressor with an 8% static pressure rise across the rotor followed by another 9% pressure rise across the stator (compounded to be 17.7%). The incoming flow has a velocity of 75 ft/sec in the axial direction, a temperature of 560°R and a pressure of 2.0 atmospheres. $\gamma = 1.4$; $c_p = 0.24$ Btu/1bm°R; polytropic efficiency = 0.95 for the compressor stage. (a) What is the power per unit mass flow of the compressor? (b) If the rotor blade speed averages 1000 ft/sec, what is the tangential component of velocity exiting the rotor?

2. Suppose a particular compressor has a compression ratio $P_2/P_1 = 15$ and the incoming air temperature is 300K. If the adiabatic efficiency is .95, what is (a) the final temperature, (b) the average polytropic efficiency, and (c) the entropy change? (d) What is the power required, if 25 kgm per sec. flow through the compressor?

3. Consider a turbine stage that has a polytropic efficiency of 0.95 for the stator (nozzle) and for the rotor flow. 30% of the total static enthalpy drop through the stage occurs in the rotor portion. The initial and final velocities for the stage are axial and have no swirl (tangential component). Assume that only the tangential component of velocity changes through the stator (nozzle) portion. The flow has $\gamma = 1.3$ and $c_p = 0.30$ Btu/1bm°R; the incoming flow has a static temperature of 2400°R, a static pressure of 25 atmospheres, and a velocity of 200 ft/sec. The average rotational velocity of the rotor blade is 1100 ft/sec. The flow exiting the stage has a temperature of 1900°R.

(a) What are the enthalpy drop and pressure drop across the stator?
(b) What is the tangential velocity at the position between the stator and the rotor measured in a frame of reference fixed to the stator?
(c) What is the Mach number of the flow at the position between the stator and rotor measured in a frame of reference fixed to the rotor?
(d) What are the enthalpy drop and the pressure drop across the rotor?
(e) What is the power output per unit mass flux?

4. Suppose we had a gas turbine engine driving a propeller. Consider takeoff only where flight velocity is 120 ft/sec. Consider the product of gearbox efficiency and propeller efficiency to be 0.8. The pressure ratio across the compressor is ten and the pressure ratio across the turbine is ten. The pressure drops across the combustor and the nozzle are negligible. The fuel heating value is 10,000 Btu/1bm and the mixture ratio is 28. For air or products, consider $\gamma = 1.4$ and $c_p = 0.24$ Btu/1bm°R. Ambient temperature is 550°R. (a) What is the propeller power per unit mass flow of air? (b) What is the propeller thrust per unit mass flow of air? Assume isentropic compression and expansion.

5. Do a preliminary design on a turbojet engine which produces 250,000 newtons of thrust. Size constraints limit the intake cross-sectional capture area to 0.3 square meters.

The engine is intended to cruise at a Mach number of 2.5. Assume ambient air conditions are one atmosphere of pressure and 270 K. Indicate clearly your choice of fuel, temperature at entrance to turbine, diffuser type, and extent of expansion in nozzle. Also indicate whether you elect to have an afterburner. All choices must be rational and defensible, of course. Indicate mass flows of air and fuel, mixture ratio(s), thrust specific fuel consumption, exhaust velocity, work done by compressor, work done on turbine, stagnation pressure ratios across each component, throat area, nozzle exit area, and nozzle exit pressure.

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