Q1) Combustion

Solution:

a) With 15% excess air, total air is 115% or 1.15 times the stoichiometric air, then the equation will be,
$C_4H_{10} + 1.15*6.5\ O_2 + 1.15*6.5*3.76\ N_2 \rightarrow 4\ CO_2 + 5\ H_2O + 0.15*6.5\ O_2 + 1.15*6.5*3.76\ N_2$
$C_4H_{10} + 7.48\ O_2 + 28.1\ N_2$ -------$4CO_2 + 5H_2O + 0.98\ O_2 + 28.1\ N_2$ [2]
The A/F ratio with 15% excess air is,
A/F ratio vol = (7.48+28.1)/(1) = 35.6 [1]
By mass, multiply by molar masses:
A/F ratio vol = (7.48*32+28.1*28)/(12*4+10)) = 17.7 [1]

b) To find the mass proportions of products,

	ni (kmol)	ni/n (dry)	Mi	ni Mi/n [2]	mi/m (dry) [2]
CO_2	4	0.121	44	5.324	0.177
H_2O	~~5~~	-[1]		-	-
O_2	0.98	0.030	32	0.960	0.032
N_2	28.1	0.849	28	23.772	0.791
n(dry)=	33.1	1.000	M=	30.056 [1]	1.000

If mistake affecting both columns marked out of [2] then, penalise one of them by 1 mark, not the other – method is fine. M must be correct and highlighted in column base. Alternative method is to work it out without a table, by ni.Mi, which is the mass of each constituent, then add them up and then divide each one by the sum of them – which is the same thing but less formal.

c) the apparent molar mass is the sum in the 4th column, 30.06 kg·kmol^{-1} [1] – no points if no statement, simply printing the value in a calculation is insufficient to prove the point. In the case of the second method, it can be found by the sum of the masses divided by the total number of moles. The specific heat capacity can be easily determined now by multiplying the mass proportions by the c_p of each of the constituent gases [2]. Since the question doesn't specify reactant or product gas mixture, the reactant gas mixture is also acceptable for full marks – justified since method based, and reactants means more work done for the answer as well.

Q2) Convection heat transfer question

Answer:

a) The profile is zero velocity at the wall and maximum a short distance from the wall before returning to zero further from the wall. The temperature profile is the wall temperature at the wall surface and the far distant temperature far away with a monotonic decrease from the hot surface [2].
There are two forms of the Nusselt number depending on whether it is laminar regime in the lower GrPr range or turbulent in the higher [1].
b) Explanation 1 for full marks: The Prandtl number describes the ratio of thickness of velocity boundary layer to thickness of temperature boundary layer. If the fluid thermal layer is thick it means the conductivity of the fluid is low so it is hard to conduct heat away from the surface compared to the cold fluid sweeping over the surface carrying away the conducted heat, and it is viscous making poor mixing.
Explanation 2 for full marks: This can also be described as the ratio of momentum diffusivity to thermal diffusivity – therefore if momentum diffusivity is relatively high it means that the thermal diffusion is a smaller effect and that the momentum is a problem via viscosity. [1] For the high Prandtl number fluid the thermal boundary layer is thin and the velocity boundary layer is thick, therefore there is a difficulty in heat transfer which reduces the effectiveness of heat exchangers [1].

c) The expansion coefficient, β, is determined by the film temperature, $\beta = 1/T_f$**. [1]** In this case it is the average of the air and water temperature given: $T_f = (15+60)/2 = 37.5°C$, but it must be in K for use as a fraction, so 310.65 K, and $\beta = 0.00322\ K^{-1}$.
Kinematic viscosity of air at T_f is by interpolation strictly:

$$\frac{\nu-1.568}{1.807-1.568} = \frac{310.65-300}{325-300} \rightarrow \nu = 0.426 \times 0.239 + 1.568 = 1.67 \times 10^{-5}\ m^2{\cdot}s^{-1}$$

if finding μ and ρ separately, values will be 1.895×10^{-5} kg/ms and 1.138 kg/m^3 respectively.

$$Gr = \frac{g\beta l^3 \Delta T}{\nu^2} = \frac{9.81 \times 0.00322 \times 0.5^3 \times (60-15)}{1.67 \times 10^{-5^2}} = 6.371 \times 10^8$$

0.5 penalty for using width rather than height.

taking Pr from tables for 310 K, interpolating:

$$\frac{Pr - 0.707}{0.701 - 0.707} = \frac{310.65 - 300}{325 - 300} \rightarrow Pr = 0.704$$

deduct 0.5 if not interpolated off GrPr and Nu.

[2]

GrPr is 4.485×10^8, so the laminar form of the Nu is appropriate:

$$Nu = 0.59(GrPr)^{0.25}$$

And:

$Nu = 0.59(4.485\times10^8)^{0.25} = 85.9$ **[2]**

Therefore the heat transfer coefficient:

$$Nu_d = \frac{hd}{k}$$

Using k from the tables for air:

$\frac{k-2.624}{2.816-2.624} = \frac{310.65-300}{325-300} \rightarrow k = 2.659 \times 10^{-2}$ W·m^{-1}·K^{-1}. Important to get right units, not containing kW or take [-1]. And with height, d = 0.5 m:

$85.9 = \frac{h \times 0.5}{2.659 \times 10^{-2}} \rightarrow h = 4.57$ W·m^{-2}·K^{-1}. **[2]**

deduct 0.5 if not interpolated k.

Heat transfer is therefore:

$\dot{Q} = hA(T_f - T_w) = 4.57 \times 0.5 \times 0.6 \times (60 - 15) = 61.7$ W. **[1]**

3 Solution

(a) NACA1408 has max thickness equal to 8% of the chord. With c=0.4, this makes the thickness t=0.08*0.4=0.032 m **[2]**.

(b) The velocity is 180 km/h=50 m/s. The Re number is **[2]:**

$$Re = \frac{Uc}{\nu} = \frac{50\frac{m}{s} * 0.4\, m}{2 \cdot 10^{-5}\, m^2/s} = 10^6$$

For steady horizontal flight conditions, we need **[2]**:

$$W = mg = L = \frac{1}{2}\rho U^2 C_L A_p$$

Where the planform area is $A_p = 0.4\, m * 20\, m = 8\, m^2$ and the weight of the aircraft is $W = m * g = 850\, kg * 9.81\frac{m}{s^2} = 8338\, N$. This requires a lift coefficient of **[2]**:

$$C_L = \frac{2W}{\rho U^2 A_p} = \frac{2 * 8338\, N}{0.822\frac{kg}{m^3} * \left(50\frac{m}{s}\right)^2 * 8\, m^2} = 1$$

Looking at the graph in Fig. Q4 of the exam paper, for $Re = 10^6$ this requires an angle of attack of about $\alpha = 8\, deg$ **[1]**.

(c) The max mass that the airfoils can sustain is achieved when the lift coefficient is maximum, i.e. $C_{L,max} = 1.3$ at $\alpha = 12\, deg$ **[2]** from the plot in Fig. Q4. This gives a max mass of **[2]**:

$$m = \frac{1}{2g}\rho U^2 C_{L,max} A_p = \frac{1}{2 * 9.81\frac{m}{s^2}} * 0.822\frac{kg}{m^3} * \left(50\frac{m}{s}\right)^2 * 1.3 * 8\, m^2 = 1089\, kg$$

As written above, the angle of attack for $C_{L,max} = 1.3$ is $\alpha = 12\, deg$.

4. Solution

(a) Stagnation temperature is readily calculated **[3]:**

$$T_0 = T_1 \left(1 + Ma_1^2 \frac{\gamma - 1}{2}\right) = 600\ K * \left(1 + 0.8^2 \frac{0.4}{2}\right) = 676.8\ K$$

For the stagnation pressure, we first need the pressure in 1, using the ideal gas law **[1]:**

$$p_1 = \rho_1 R T_1 = 5 \frac{kg}{m^3} * 287.1\ \frac{J}{kg * K} * 600\ K = 861300\ Pa$$

And then **[3]:**

$$p_0 = p_1 \left[1 + Ma_1^2 \frac{\gamma - 1}{2}\right]^{\frac{\gamma}{\gamma - 1}} = 861300\ Pa * \left[1 + 0.8^2 \frac{0.4}{2}\right]^{\frac{1.4}{0.4}} = 1.31 \cdot 10^6\ Pa$$

(b) The area of the throat for the flow to be supersonic is calculated via **[2]**:

$$\frac{A_1}{A^*} = \frac{1}{Ma_1} \frac{(1 + 0.2 Ma_1^2)^3}{1.728} \Rightarrow A^* = A \frac{1.728 Ma_1}{(1 + 0.2 Ma_1^2)^3} = 0.05\ m^2 * \frac{1.728 * 0.8}{(1 + 0.2 * 0.8^2)^3} = 0.0482\ m^2$$

And therefore the throat diameter is **[2]:**

$$d^* = \sqrt{\frac{4A^*}{\pi}} = \sqrt{\frac{4 * 0.0482\ m^2}{\pi}} = 0.2477\ m$$

(c) The area of the outlet section to achieve $Ma_2 = 2$ is calculated via the same formula **[2]:**

$$A_2 = A^* \frac{(1 + 0.2 Ma_2^2)^3}{1.728 Ma_2} = 0.0482\ m^2 * \frac{(1 + 0.2 * 2^2)^3}{1.728 * 2} = 0.0813\ m^2$$