

Aircraft Propulsion | (2nd Edition)

Chapter 4, Problem 12P

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Step 1 of 20

(a)

Apply the equation for the velocity, V_0 .

$$V_0 = M_0 \sqrt{\gamma R T_0}$$

Substitute 0.8 for M_0 , 1.4 for γ , -25°C for T_0 , and 287 J/kgK for R .

$$V_0 = 0.8 \sqrt{1.4(287\text{ J/kgK})(-15^\circ\text{C})}$$
$$= 257.58\text{ m/s}$$

Comments (2)

Step 2 of 20

Apply the equation for Ram Drag, 3.15.

$$D_{ram} = \dot{m} V_0$$

Substitute 257.58 m/s for V_0 and 100 kg/s for $\dot{m}_0$.

$$D_{ram} = 257.58\text{ m/s}(100\text{ kg/s})$$
$$= 25.758\text{ kN}$$

Therefore, the Ram Drag is $[25.758\text{ kN}]$.

Comment

Step 3 of 20

(b)

Reference the Isentropic Table, for $M_0 = 0.8$. Find and approximate $\frac{p_{01}}{p_0}$.

$$\frac{p_{01}}{p_0} = 1.524$$

Substitute 10 kPa for p_0 .

$$\frac{p_{01}}{10\text{ kPa}} = 1.524$$
$$p_{01} = 15.24\text{ kPa}$$

Since, $p_{12} = p_{01}$, then $p_{12} = 15.24\text{ kPa}$. K.

Comment

Step 4 of 20

Apply the equation for the pressure ratio in the compressor.

$$\pi_c = \frac{p_{13}}{p_{12}}$$

Substitute 20 for π_c and 15.24 kPa for p_{12} .

$$20 = \frac{p_{13}}{15.24\text{ kPa}}$$
$$p_{13} = 304.9\text{ kPa}$$

Comment

Step 5 of 20

Apply the equation for the temperature ratio in the compressor.

$$\tau_c = \pi_c^{\frac{\gamma-1}{\gamma}}$$

Substitute 20 for π_c , 1 for γ_c , and 1.4 for γ .

$$\tau_c = 20^{\frac{1.4-1}{1.4}}$$
$$= 2.354$$

Comment

Step 6 of 20

Reference the Isentropic Table, for $M_0 = 0.8$. Find and approximate $\frac{T_{01}}{T_0}$.

$$\frac{T_{01}}{T_0} = 1.128$$

Substitute -15°C for T_0 .

$$\frac{T_{01}}{-15^\circ\text{C}} = 1.128$$
$$T_{01} = 291.17\text{ K}$$

Since, $T_{12} = T_{01}$, then $T_{12} = 291.17\text{ K}$. K.

Comments (2)

Step 7 of 20

Apply the equation for the temperature ratio in the compressor.

$$\tau_c = \frac{T_{13}}{T_{12}}$$

Substitute 2.354 for τ_c and 291.17 K for T_{12} .

$$2.354 = \frac{T_{13}}{291.17\text{ K}}$$
$$T_{13} = 685.41\text{ K}$$

Comment

Step 8 of 20

Apply the equation for shaft power.

$$\dot{Q}_s = \dot{m}_0 (T_{13} - T_{12})$$

Substitute 100 kg/s for $\dot{m}_0$, 291.17 K for T_{12} , and 685.41 K for T_{13} .

$$\dot{Q}_s = 100\text{ kg/s}(685.41\text{ K} - 291.17\text{ K})$$
$$= 39.57\text{ MW}$$

Therefore, the shaft power is $[39.57\text{ MW}]$.

Comments (2)

Step 9 of 20

(c)

Apply the equation for the fuel to ratio.

$$f = \frac{\dot{m}_f}{\dot{m}_0}$$

Substitute 100 kg/s for $\dot{m}_0$ and 2 kg/s for $\dot{m}_f$.

$$f = \frac{2\text{ kg/s}}{100\text{ kg/s}}$$
$$= 0.02$$

Therefore, the fuel to air ratio is $[0.02]$.

Comments (1)

Step 10 of 20

(d)

Apply the equation for fuel to air ratio to find T_{14} .

$$f = \frac{T_{14} - T_{13}}{\frac{\dot{Q}_f}{c_p} - T_{14}}$$

Substitute 0.02 for f , 42800 kJ/kg for $\dot{Q}_f$, 685.41 K for T_{13} , 1.004 kJ/kgK for c_p , and 1 for η_b , since it is ideal.

$$0.02 = \frac{T_{14} - 685.41\text{ K}}{\frac{42800\text{ kJ/kg}}{1.004\text{ kJ/kgK}} - T_{14}}$$
$$T_{14} = \frac{0.02 \left(\frac{42800\text{ kJ/kg}}{1.004\text{ kJ/kgK}} \right) + 685.41\text{ K}}{1.02}$$
$$= 1507.7\text{ K}$$

Comment

Step 11 of 20

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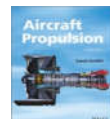
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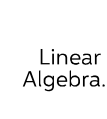
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$\tau_{i4} = \frac{-15}{T_6}$
Substitute 1507.7 K for T_{i4} and -15 C for T_6 .
 $\tau_{i4} = \frac{1507.7 \text{ K}}{-15^\circ \text{ C}}$
 $= 5.84$
Therefore, the enthalpy parameter is $[5.84]$.

Comments (1)

Step 12 of 20

(e)
Apply the equation for the turbine expansion parameter.
 $\tau_r = 1 - \frac{\tau_i(\tau_i - 1)}{\eta_a(1 + f)\tau_c}$
Substitute 0.02 for f , 5.84 for τ_{i4} , 2.354 for τ_{i5} , and 1 for η_a , since it is ideal components.
 $\tau_r = 1 - \frac{\tau_i(2.354 - 1)}{(1 + 0.02)5.84}$
 $= 0.7437$
Therefore, the equation for the turbine expansion parameter is $[0.7437]$.

Comments (2)

Step 13 of 20

(f)
Apply the equation for the turbine shaft power.
 $\dot{P}_t = \eta_m \dot{P}_r$
Substitute 39.57 MW for $\dot{P}_r$ and 1 for η_m , since it is ideal components.
 $\dot{P}_t = 39.57 \text{ MW} \left(1 \right)$
Therefore, the turbine shaft power is $[39.57 \text{ MW}]$.

Comment

Step 14 of 20

(g)
Apply the definition of the afterburner limit enthalpy parameter.
 $\tau_{i,ab} = \frac{T_{i,ab}}{T_6}$
Substitute -15 C for T_6 and 2200 K for $T_{i,ab}$.
 $\tau_{i,ab} = \frac{2200 \text{ K}}{-15^\circ \text{ C}}$
 $= 8.522$
Therefore the afterburner limit enthalpy parameter is $[8.522]$.

Comment

Step 15 of 20

(h)
Apply the equation to the afterburner fuel to air ratio.
 $f_{ab} = \frac{(1 + f)(\tau_{i,ab} - \tau_{i4}\tau_r)}{\frac{Q_{r,ab}}{c_p T_6} \eta_{ab} - \tau_{i,ab}}$
Substitute 8.522 for $\tau_{i,ab}$, 0.7437 for τ_{i5} , -15 C for T_6 , 0.02 for f , 5.84 for τ_{i4} , 42800 kJ/kg for $Q_{r,ab}$, 1.004 kJ/kgK for c_p , and 1 for η_{ab} , since it is ideal.
 $f_{ab} = \frac{(1 + 0.02)(8.522 - (5.84)(0.7437))}{\frac{(42800 \text{ kJ/kg})}{(1.004 \text{ kJ/kgK})(-15^\circ \text{ C})} - 8.522}$
 $= 0.0272$
Therefore, the fuel to air ratio is $[2.72\%]$.

Comment

Step 16 of 20

(i)
Apply the equation for nozzle gross thrust.
 $F_g = \dot{m}_0 V_9 + (p_9 - p_a) A_9$
Substitute the continuity equation for A_9
 $F_g = \dot{m}_0 V_9 + (p_9 - p_a) \frac{\dot{m}_0}{\rho_9 V_9}$
Substitute 102 kg/s for $\dot{m}_0$, 15 kPa for p_a , 10 kPa for p_9 .
 $F_g = (102 \text{ kg/s}) V_9 + (15 \text{ kPa} - 10 \text{ kPa}) \frac{(102 \text{ kg/s})}{\rho_9 V_9}$
Substitute $\frac{p_9}{RT_9}$ for ρ_9 and $M_9 \sqrt{\gamma RT_9}$ for V_9 .
 $F_g = (102 \text{ kg/s}) M_9 \sqrt{\gamma RT_9} + (15 \text{ kPa} - 10 \text{ kPa}) \frac{p_9}{RT_9} M_9 \sqrt{\gamma RT_9}$
Substitute 1.4 for γ , 1251 K for T_9 from $T_{r7} = T_{r9}$.
 $F_g = \left[\frac{(102 \text{ kg/s}) M_9 \sqrt{(1.4)287 \text{ J/kgK}(1251 \text{ K})}}{+ (15 \text{ kPa} - 10 \text{ kPa}) \frac{10 \text{ kPa}}{287 \text{ J/kgK}(1251 \text{ K})} M_9 \sqrt{(1.4)287 \text{ J/kgK}(1251 \text{ K})}} \right]$
Substitute 1.95 for M_9 from $\frac{108.18 \text{ kPa}}{15 \text{ kPa}} = 7.22$, referencing the isentropic Table.
 $F_g = \left[\frac{(102 \text{ kg/s})(1.95 \sqrt{(1.4)287 \text{ J/kgK}(1251 \text{ K})})}{+ (15 \text{ kPa} - 10 \text{ kPa}) \frac{10 \text{ kPa}}{287 \text{ J/kgK}(1251 \text{ K})} (1.95 \sqrt{(1.4)287 \text{ J/kgK}(1251 \text{ K})})} \right]$
 $= 153.64 \text{ kN}$
Therefore the nozzle gross thrust is $[153.64 \text{ kN}]$.

Comments (3)

Step 17 of 20

(j)
Apply the equation for the thrust specific fuel consumption.
 $TSFC = \frac{\dot{m}_f}{F_g}$
Substitute 153.64 kN for F_g and 2 kg/s for $\dot{m}_f$.
 $TSFC = \frac{2 \text{ kg/s}}{153.64 \text{ kN}}$
 $= 36.92 \frac{\text{mg/s}}{\text{N}}$
Therefore the thrust specific fuel consumption is $[36.92 \frac{\text{mg/s}}{\text{N}}]$.

Comments (3)

Step 18 of 20

(k)
Apply the equation for the un-installed thrust.
 $F_{net,un-installed} = F_g - D_{ram}$
Substitute 153.64 kN for F_g and $V_9/\dot{m}_0 = 100 \text{ kg/s}(257.58 \text{ m/s})$ for D_{ram} .
 $F_{net} = 153.64 \text{ kN} - 100 \text{ kg/s}(257.58 \text{ m/s})$
 $= 127.9 \text{ kN}$
Therefore, the net (un-installed) thrust is $[127.9 \text{ kN}]$.

Comment

Step 19 of 20

(l)
Apply the equation for the thermal efficiency.
 $\eta_{th} = \frac{(1 + f + f_{ab}) \frac{V_9^2}{2} - \frac{V_1^2}{2}}{f - Q_{r,ab} \eta_a + f_{ab} \frac{Q_{r,ab}}{2}}$
 $= 0.47731$
Therefore, the thermal efficiency is $[47.7\%]$.

Comment

Step 20 of 20

$$\eta_p = \frac{2F_a V_0}{\dot{m}_0 V_0^2 - \dot{m}_0 V_0^2}$$

Substitute $F_a = D_{\text{act}}$ for F_a and $\frac{F_a}{\dot{m}_0}$ for V_0^2 .

$$\eta_p = \frac{2(F_a - D_{\text{act}})V_0}{\dot{m}_0 \left(\frac{F_a}{\dot{m}_0} \right) - \dot{m}_0 V_0^2}$$

$\eta_p = 0.3414$

Therefore, the propulsive efficiency is 34.14%.

Comments (1)

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