Mattingly problem 4.3 (4.1)

The inlet for a high-bypass-ratio turbofan engine has an area $A_1 = 6.0m^2$ and is designed to have an inlet Mach number $M_1 = 0.6$. Determine the additive drag at the flight conditions of sea-level static test and Mach number of 0.8 at 12-km altitude.

$M_1 = 0.6 \qquad M_0 = 0 \qquad p_0 = 1.013 \cdot 10^5 \cdot Pa \qquad A_1 = 6 \cdot m^2$

$$D_{add} = \int_0^1 (p - p_0) dA_x = \dot{m}_1 V_1 + (p_1 - p_0) A_1 - \dot{m}_0 V_0$$

$$p_1 = \frac{p_1}{p_{t1}} \frac{p_{t1}}{p_0} p_0 = (\psi_1)^{-\frac{1}{k}} \cdot 1 \cdot 1.013 \cdot 10^5 = (1.072)^{-\frac{1}{0.286}} \cdot 1 \cdot 1.013 \cdot 10^5 = 79.4 \cdot kPa$$

$$\psi_1 = 1 + \frac{\gamma - 1}{2} M_1^2 = 1 + 0.2 \cdot 0.6^2 = 1.072$$

$$\dot{m}_1 V_1 = \rho_1 V_1^2 A_1 = \rho_1 V_1^2 A_1 \frac{p_1}{p_1} = V_1^2 A_1 \frac{\gamma p_1}{\gamma R T_1} = \gamma p_1 M_1^2 A_1 = 1.4 \cdot 79.4 \cdot 10^3 \cdot 0.6^2 \cdot 6$$
$$= 24.0 \cdot 10^4 \cdot N$$

$$(p_1 - p_0) A_1 = (79.4 - 101.3) 6 \cdot 10^3 = -131.4 \cdot 10^3 \cdot N$$

$$D_{add} = \dot{m}_1 V_1 + (p_1 - p_0) A_1 - \dot{m}_0 V_0 = 24.0 \cdot 10^4 - 131.4 \cdot 10^4 = 108.6 \cdot kN$$

$$\frac{D_{add}}{p_o A_1} = \frac{108.6}{1.013 \cdot 10^5 \cdot 6} = 0.1787$$

$$\frac{D_{add}}{p_o A_1} = \gamma M_1 \left(\frac{\psi_0}{\psi_1}\right)^K \left(M_1 \sqrt{\frac{\psi_0}{\psi_1}} - M_0\right) + \left(\frac{\psi_0}{\psi_1}\right)^{\frac{\gamma}{\gamma - 1}} - 1 = A + B$$

$$A = \gamma M_1 \left(\frac{\psi_0}{\psi_1}\right)^K \left(M_1 \sqrt{\frac{\psi_0}{\psi_1}} - M_0\right) \qquad B = \left(\frac{\psi_0}{\psi_1}\right)^{\frac{\gamma}{\gamma - 1}} - 1 \qquad K = \frac{\gamma + 1}{2(\gamma - 1)} = 3$$

$$\psi_0 = 1 + \frac{\gamma - 1}{2} M_0^2 = 1 + 0.2 \cdot 0^2 = 1.000 \qquad \psi_1 = 1 + \frac{\gamma - 1}{2} M_1^2 = 1 + 0.2 \cdot 0.6^2 = 1.072$$

$$A = 1 + \left(\frac{\psi_0}{\psi_1}\right)^K \left(M_1 \sqrt{\frac{\psi_0}{\psi_1}} - M_0\right) = 1 + \left(\frac{1}{1.072}\right)^3 \left(0.6 \sqrt{\frac{1}{1.072}} - 0\right) = 0.395$$

$$B = \left(\frac{\psi_0}{\psi_1}\right)^{\frac{\gamma}{\gamma - 1}} - 1 = \left(\frac{1}{1.072}\right)^{0.286} - 1 = -0.216 \qquad \frac{D_{add}}{p_o A_1} = A + B = 0.395 - 0.216 = 0.179$$

Mach number of 0.8 at 12-km altitude

$M_1 = 0.6 \qquad M_0 = 0.8 \qquad p_0 = 19.4 \cdot kPa \qquad A_1 = 6 \cdot m^2$

$$M_0 = 0.8 \xrightarrow{ISO} \frac{A_0}{A^*} = 1.038 \qquad \frac{p_0}{p_{t0}} = 0.656$$

$$M_1 = 0.6 \xrightarrow{ISO} \frac{A_1}{A^*} = 1.188 \qquad \frac{p_1}{p_{t1}} = 0.784$$

$$p_1 = \frac{p_1}{p_{t1}} \frac{p_{t1}}{p_{t0}} \frac{p_{t0}}{p_0} p_0 = \frac{0.784}{0.656} \cdot 19.4 = 23.2 \cdot kPa$$

$$A_0 = \frac{A_0}{A^*}\frac{A^*}{A_1}A_1 = \frac{1.038}{1.188}6 = 5.24 \cdot m^2$$

$$\dot{m}_1 V_1 = \rho_1 V_1^2 A_1 = \rho_1 V_1^2 A_1 \frac{p_1}{p_1} = V_1^2 A_1 \frac{\gamma p_1}{\gamma R T_1} = \gamma p_1 M_1^2 A_1 = 1.4 \cdot 23.2 \cdot 0.6^2 \cdot 6 = 70.2 \cdot kN$$

$$\dot{m}_0 V_0 = \gamma p_0 M_0^2 A_0 = 1.4 \cdot 19.4 \cdot 0.8^2 \cdot 5.24 = 91.1 \cdot kN$$

$$(p_1 - p_0)A_1 = (23.2 - 19.4)6 \cdot 10^3 = 22.8 \cdot kN$$

$$D_{add} = 70.2 + 22.8 - 91.1 = 1.9 \cdot kN$$

$$\frac{D_{add}}{p_o A_1} = \frac{1.9}{19.4 \cdot 10^3 \cdot 6} = 0.01632 \qquad \psi_0 = 1 + \frac{\gamma - 1}{2} M_0^2 = 1 + 0.2 \cdot 0.8^2 = 1.128$$

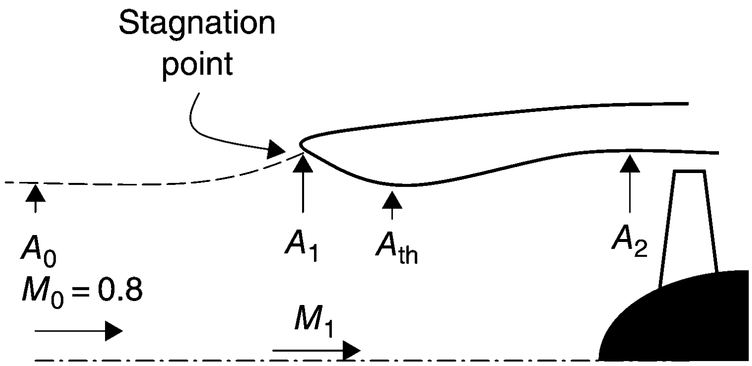
$$A = \gamma M_1 \left(\frac{\psi_0}{\psi_1}\right)^K \left(M_1 \sqrt{\frac{\psi_0}{\psi_1}} - M_0\right) = 1.4 \cdot 0.6 \cdot \left(\frac{1.128}{1.072}\right)^3 \left(0.6 \sqrt{\frac{1.128}{1.072}} - 0\right) = -0.1806$$

$$B = \left(\frac{\psi_0}{\psi_1}\right)^{\frac{\gamma}{\gamma - 1}} - 1 = \left(\frac{1.128}{1.072}\right)^{0.286} - 1 = 0.1949$$

$$\frac{D_{add}}{p_o A_1} = A + B = -0.1806 + 0.1949 = 0.0143$$

Farokhi problem 6.3

Consider a subsonic inlet at a flight cruise Mach number of 0.8. The captured streamtube undergoes a prediffusion external to the inlet lip, with an area ratio $A_0/A_1 = 0.92$ as shown. Calculate



(a) Cp (i.e., the pressure coefficient) at the stagnation point

$$c_p = \frac{p_{t0} - p_0}{\frac{1}{2}\rho_0 V_0^2} = \frac{\gamma p_0}{\frac{1}{2}\gamma\rho_0 V_0^2}\left(\frac{p_{t0}}{p_0} - 1\right) = \frac{a_0^2}{\frac{1}{2}\gamma V_0^2}\left(\frac{p_{t0}}{p_0} - 1\right) = \frac{2}{\gamma M_0^2}\left(\frac{p_{t0}}{p_0} - 1\right)$$

$$\frac{A}{A^*} = \frac{\Psi^*}{\Psi} = \frac{0.810}{\gamma M\left(1 + \frac{\gamma - 1}{2}M^2\right)^{-K}} \qquad K = \frac{\gamma + 1}{2(\gamma - 1)}$$

$$M_0 \xrightarrow{ISO} \frac{p_o}{p_{t0}} = 0.656 \quad \frac{T_o}{T_{t0}} = 0.887 \quad \frac{A_0}{A^*} = 1.038$$

$$c_p = \frac{2}{\gamma M_0^2}\left(\frac{p_{t0}}{p_0} - 1\right) = \frac{2}{1.4 \cdot 0.8^2}\left(\frac{1}{0.656} - 1\right) = 1.171$$

(b) inlet lip Mach number M1

$$\frac{A_1}{A^*} = \frac{A_1}{A_0}\frac{A_0}{A^*} = \frac{1.038}{0.92} = 1.128 \quad \xrightarrow{ISO} \quad M_1 = 0.658 \quad \frac{T_1}{T_{t1}} = \quad 0.920 \quad \frac{p_1}{p_{t1}} = 0.748$$

(c) lip contraction ratio A_1/A_{th} for a throat Mach number $M_{th} = 0.75$ (assume $p_{t,th}/p_{t1} = 1$)

$$M_{th} = 0.75 \quad \xrightarrow{ISO} \quad \frac{A_{th}}{A^*} = 1.062 \quad \frac{A_1}{A_{th}} = \frac{A_1}{A^*}\frac{A^*}{A_{th}} = \frac{1.128}{1.062} = 1.062$$

(d) the diffuser area ratio A_2/A_{th} if $M_2 = 0.5$ and $p_{t2}/p_{t,th} = 0.98$

$$M_2 = 0.5 \quad \xrightarrow{ISO} \quad \frac{A_2}{A^*} = 1.340 \qquad \left.\frac{A_2}{A_{th}}\right|_{ideal} = \frac{A_2}{A^*}\frac{A^*}{A_{th}} = \frac{1.340}{1.062} = 1.262$$

$$\frac{A_2}{A_{th}} = \left.\frac{A_2}{A_{th}}\right|_{ideal} \cdot \frac{p_{t,th}}{p_{t2}} = \frac{1.262}{0.98} = 1.288$$

(e) the nondimensional inlet additive drag $D_{add}/p_0 A_1$

$$D_{add} = \dot{m}_1 V_1 + (p_1 - p_0)A_1 - \dot{m}_0 V_0$$

$$\frac{a_1}{a_0} = \sqrt{\frac{T_1}{T_{t1}}\frac{T_{t0}}{T_0}} = \sqrt{\frac{0.920}{0.887}} = 1.018 \qquad \frac{p_1}{p_0} = \frac{p_1}{p_{t1}}\frac{p_{t1}}{p_{t0}}\frac{p_{t0}}{p_0} = 0.748 \cdot 1 \frac{1}{0.656} = 1.140$$

$$\frac{D_{add}}{p_0 A_1} = \frac{(p_1 - p_0)A_1}{p_0 A_1} + \frac{\dot{m}_0}{p_0 A_1}(V_1 - V_0) = \left(\frac{p_1}{p_0} - 1\right) + \frac{\gamma \rho_0 V_0^2 A_0}{\gamma p_0 A_1}\frac{(V_1 - V_0)}{V_0}$$

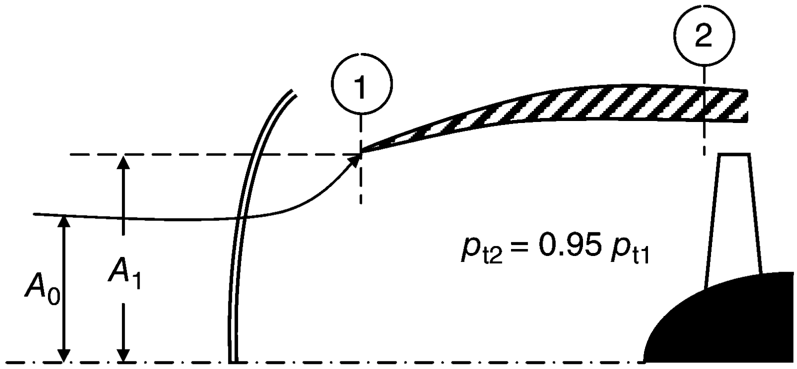
$$= \left(\frac{p_1}{p_0} - 1\right) + \frac{\gamma M_0^2 A_0}{A_1}\left(\frac{V_1}{V_0} - 1\right) = \left(\frac{p_1}{p_0} - 1\right) + \gamma \frac{A_0}{A_1} M_0 \left(M_1 \frac{a_1}{a_0} - M_0\right)$$

$$= 0.140 + 1.4 \cdot 0.92 \cdot 0.8(0.658 \cdot 1.018 - 0.8) = 5.890 \cdot 10^{-3}$$

Ao/A1	0.92			
γ	1.4			
k	3.5			
K	3			
pt2/pt0	0.98			
	0	*	th	2
M	0.8	1	0.75	0.5
ψ	1.128	1.2	1.1125	1.05
Ψ	0.78035	0.810185	0.762587	0.6046863
			Con le tabelle	
pt0/p0	1.52434	M0	$\pi 0/\pi\tau 0$	0.6560216
			T0/Tt0	0.8865248
Cpre	1.1704		A0/A*	1.03823
A0/A*	1.03823		Cpre	1.1704018
A1/A*	1.12851	Mth	Ath/A*	1.0624171
M1	0.65778		A1/A*	1.1285109
$\psi 1$	1.08654		M1	0.6577826
Ath/A*	1.06242		$\mathrm{A1/A}\tau\eta$	1.0622107
A1/Ath	1.06221	M2	$\mathrm{A2/A}*$	1.3398438
A2/A*	1.33984		A2/Ath Ide	1.2611278
A2/Ath Ide	1.26113		$\mathrm{A2/A}\tau\eta$	1.2868651
A2/Ath	1.28687	M1	p1/pt0	0.7479045
			T1/Tt0	0.9203564
$\psi 0/\psi 1$	1.03816		p1/p0	1.1400608
Dadd/poA1	0.00633		T1/T0	1.038162
			Dadd/poA1	0.0063316

Farokhi problem 6.17

A normal-shock inlet is operating in a supercritical mode, as shown. Flight Mach number is $M_0 = 1.6$. The inlet capture area ratio $A_0/A_1 = 0.90$ and the diffuser area ratio $A_2/A_1 = 1.2$ and $p_{t2}/p_{t1} = 0.95$ Calculate
(a) M_1, M_2
(b) inlet total pressure recovery π_d, i.e. p_{t2}/p_{t0}



■ **FIGURE P6.17**

$$M_0 = 1.6 \xrightarrow{NSW} M_{0v} = 0.668 \qquad \frac{p_{t0v}}{p_{t0}} = 0.895$$

$$M_{0v} = 0.668 \xrightarrow{Iso} \frac{A_{0v}}{A_{0v}^*} = 1.120$$

$$\frac{A_1}{A_1^*} = \frac{A_1}{A_0}\frac{A_0}{A_{ov}}\frac{A_{0v}}{A_{0v}^*} = \frac{1}{0.90} 1 \cdot 1.120 = 1.244 \xrightarrow{ISO} M_1 = 0.557$$

$$\frac{A_2}{A_2^*} = \frac{A_2}{A_1}\frac{A_1}{A_1^*}\frac{A_1^*}{A_2^*} = 1.2 \cdot 1.244 \cdot 0.95 = 1.418 \xrightarrow{ISO} M_2 = 0.463$$

$$\frac{p_{t2}}{p_{t0}} = \frac{p_{t2}}{p_{t1}}\frac{p_{t1}}{p_{t0v}}\frac{p_{t0v}}{p_{t0}} = 0.95 \cdot 1 \cdot 0.895 = 0.850$$

6.17		
Input	M0	1.6
Input	A0/A1	0.9
Input	A2/A1	1.2
Input	pt2/pt1	0.95
NSW	M0v	0.668437
NSW	ptv/ptm	0.8952
Iso	A/A*	1.119211
	A1/A*	1.243567
Iso	M1	0.557724
	A2/A*	1.417667
Iso	M2	0.462995
	pt2/pt0	0.85044

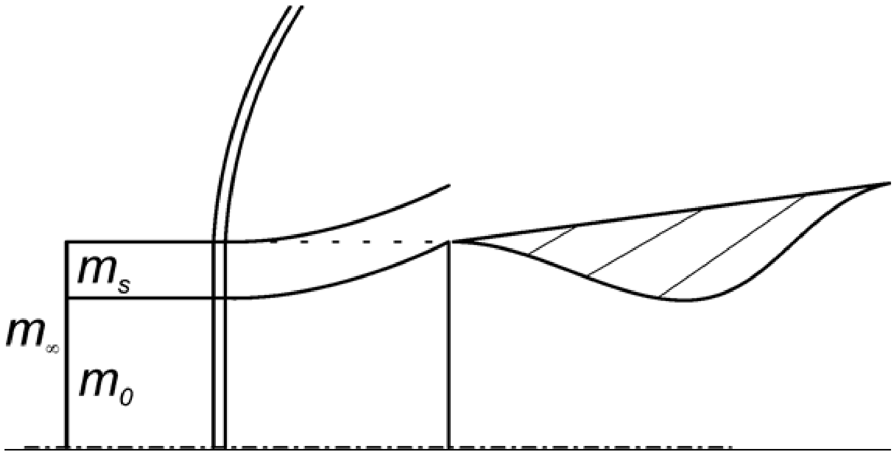
Farokhi problem 6.18

An isentropic convergent–divergent supersonic inlet is designed for $M_0 = 1.6$. Calculate the inlet's
(a) area contraction ratio A_1/A_{th}
(b) subsonic Mach number where the throat first chokes

$$M_0 = 1.6 \quad \xrightarrow{Iso} \quad \frac{A_0}{A_0^*} = \frac{A_1}{A_1^*} = \frac{A_1}{A_{\text{th}}} = 1.250 \quad \xrightarrow{Iso} \quad M_{sub} = 0.553$$

(c) percent spillage at $M_0 = 0.7$

(d) percent spillage at $M_0 = 1.6$ (in the unstarted mode)



$$\%SP = \frac{\dot{m}_s}{\dot{m}_{t0t}} = \frac{\dot{m}_\infty - \dot{m}_0}{\dot{m}_\infty} = 1 - \frac{\dot{m}_0}{\dot{m}_\infty}$$

$$\dot{m}_0 = \dot{m}_1 = \frac{p_{t1} A_1^* \Psi^*}{a_t} \qquad \dot{m}_\infty = \dot{m}_0 \frac{A_1}{A_0} \qquad \frac{\dot{m}_0}{\dot{m}_\infty} = \frac{A_0}{A_1} = \frac{A_0}{A_0^*}\frac{A_0^*}{A_1^*}\frac{A_1^*}{A_1} \quad dove\ in\ generale \quad \frac{A_0^*}{A_1^*} = \frac{p_{t1}}{p_{t0}}$$

$$M_0 = 0.7 \quad \xrightarrow{Iso} \quad \frac{A_0}{A_0^*} = 1.094 \qquad \frac{A_1}{A_1^*} = \frac{A_1}{A_{th}} = 1.250$$

$$\%SP = \frac{\dot{m}_s}{\dot{m}_{t0t}} = 1 - \frac{A_0}{A_0^*}\frac{p_{t1}}{p_{t0}}\frac{A_1^*}{A_1} = 1 - 1.094 \cdot 1 \cdot \frac{1}{1.250} = 12.48\%$$

$$M_0 = 1.6 \quad \xrightarrow{Iso} \quad \frac{A_0}{A_0^*} = 1.250 \qquad \frac{A_1}{A_1^*} = \frac{A_1}{A_{th}} = 1.250$$

$$M_0 = 1.6 \quad \xrightarrow{NSW} \quad \frac{p_{t1}}{p_{t0}} = 0.895$$

$$\%SP = \frac{\dot{m}_s}{\dot{m}_{t0t}} = 1 - \frac{A_0}{A_0^*}\frac{p_{t1}}{p_{t0}}\frac{A_1^*}{A_1} = 1 - 1.250 \cdot 0.895 \cdot \frac{1}{1.250} = 10.50\%$$

(e) overspeed Mach number to start this inlet, M overspeed

$$M_{sub} = 0.553 \quad \xrightarrow{NSW} \quad M_{over} = 2.16$$

(f) throat Mach number after the inlet was started, with still M overspeed as the flight Mach number

$$M_{over} = 2.16 \quad \xrightarrow{ISO} \quad \frac{A_1}{A^*} = 1.935 \qquad \frac{A_{\text{th}}}{A^*} = \frac{A_{\text{th}}}{A_1}\frac{A_1}{A^*} = \frac{1.935}{1.250} = 1.548$$

$$\frac{A_{th}}{A^*} = 1.548 \quad \xrightarrow{ISO} \quad M_{th} = 1.894$$

6.18		
Input	M0	1.6
NSW	pt2/pt0	0.8952
iso	A/A*	1.250235
iso	Msub crit	0.553164
Input	M0	0.7
iso	A/A*	1.094373
M=07	%spillage	12.46664
M=1.6	%spillage	10.47997
Nsw	M over	2.155573
iso	A/A*	1.927934
	Ath/A*	1.542057
iso	Mth	1.889287

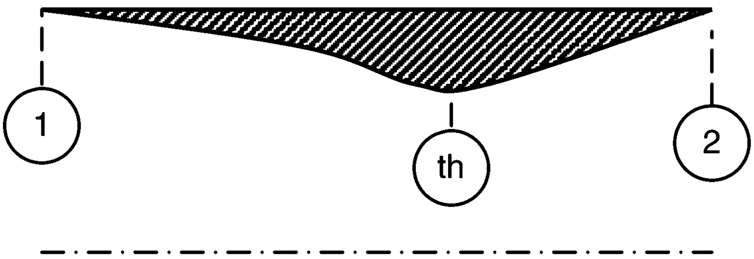
Farokhi problem 6.23

6.23 A Kantrowitz–Donaldson inlet is designed for $M_D = 1.7$. Calculate

(a) the inlet contraction ratio A_1/A_{th}

(b) the throat Mach number after the inlet self started

(c) the total pressure recovery with the best backpressure.



$$M_D = 1.7 \xrightarrow{NSW} M_v = 0.641 \xrightarrow{ISO} \frac{A_1}{A^*} = \frac{A_1}{A_{th}} = 1.144$$

$$M_D = 1.7 \xrightarrow{ISO} \frac{A_1}{A^*} = 1.338$$

$$\frac{A_{th}}{A^*} = \frac{A_{th}}{A^*}\frac{A_1}{A^*} = \frac{1.338}{1.144} = 1.170 \xrightarrow{ISO} M_{th} = 1.491$$

$$M_{th} = 1.491 \xrightarrow{NSW} \frac{p_{t2}}{p_{t1}} = 0.933$$

$$M_D = 1.7 \xrightarrow{NSW} \frac{p_{t2}}{p_{t1}} = 0.856$$

6.23		
Input	M0	1.7
iso	A/A*	1.337606
Nsw	Mv	0.640544
iso	A1/Ath=A/	1.144618
	Ath/A*	1.168605
iso	Mth	1.488692
NSW	ptv/pt0	0.933331

Farokhi problem 6.43

A convergent nozzle experiences π_n of 0.98, the gas ratio of specific heats γ = 1.30, and the gas constant is $R = 291\, J/kgK$. First, calculate the minimum nozzle pressure ratio that will choke the expanding nozzle, i.e., NPR_{crit}. This nozzle operates, however, at a higher NPR than the critical, namely, NPR=4. 2 and with an inlet stagnation temperature of $T_{t7} = 939K$. Assuming this nozzle operates in $p_0 = 100kPa$ ambient static pressure, calculate:

$$\gamma = 1.3 \qquad k = \frac{\gamma - 1}{\gamma} = \frac{0.3}{1.3} = 0.231 \qquad \psi(M = 1) = 1 + \frac{\gamma - 1}{2} = 1 + 0.15 \quad = 1.150$$

$$\frac{p_{t8}}{p_8} = \psi^{\frac{1}{k}} = 1.15^{\frac{1}{0.231}} = 1.831 \qquad NPR_{crit} = \frac{p_{t7}}{p_0} = \frac{p_{t7}}{p_8} = \frac{p_{t7}}{p_{t8}}\frac{p_{t8}}{p_8} = \frac{1}{\pi_n}\frac{p_{t8}}{p_8} = \frac{1.831}{0.98} = 1.868$$

(a) the exit static pressure and temperature p_9 and T_9, respectively

$$NPR = \frac{p_{t7}}{p_0} = 4.2 \qquad p_{t7} = NPR \cdot p_0 = 4.2 \cdot 100 = 420 \cdot kPa$$

$$p_{t8} = p_{t7}\pi_n = 420 \cdot 0.98 = 412 \cdot kPa \qquad p_9 = p_8 = \frac{p_8}{p_{t8}} p_{t8} = \frac{412}{1.831} = 225 \cdot kPa$$

$$T_9 = T_8 = \frac{T_{t7}}{\psi} = \frac{939}{1.150} = 817 \cdot K$$

(b) the actual exit velocity V_9 in m/s

$$V_9 = a_9 = \sqrt{\gamma R T_9} = \sqrt{1.3 \cdot 291 \cdot 817} = 556 \cdot \frac{m}{s}$$

(c) nozzle adiabatic efficiency η_n

$$\eta_n = \frac{h_{t7} - h_9}{h_{t7} - h_{9s}} = \frac{V_9^2/2}{V_{9s}^2/2} = \frac{\left(NPR\frac{p_0}{p_9}\right)^{k_9} - \pi_n^{-k_9}}{\left(NPR\frac{p_0}{p_9}\right)^{k_9} - 1} = \frac{1.868^{0.231} - 0.98^{-0.231}}{1.868^{0.231} - 1} = \frac{1.155 - 0.98^{-0.231}}{1.155 - 1}$$
$$= 0.970$$

(d) the ideal exit velocity V_{9s} in m/s

$$V_{9s} = \frac{V_9}{\sqrt{\eta_n}} = \frac{556}{\sqrt{0.970}} = 565 \cdot \frac{m}{s}$$

(e) percentage gross thrust gain, had we used a convergent–divergent nozzle with perfect expansion

$$\frac{F_{u_{conv}}}{\dot{m}_8 a_8} = \frac{V_8}{a_8} + A_8\frac{p_8 - p_0}{\dot{m}_8 a_8} = \frac{V_8}{a_8} + \frac{p_8 - p_0}{p_8}\frac{p_8 A_8}{\dot{m}_8 a_8} = \frac{V_8}{a_8} + \frac{p_8 - p_0}{p_8}\gamma p_8 \frac{A_8}{\gamma \rho_8 V_8 A_8 a_8}$$
$$= \frac{V_8}{a_8} + \frac{p_8 - p_0}{p_8}\frac{a_8^2}{\gamma V_8 a_8} = 1 + \left(1 - \frac{100}{225}\right)\frac{1}{1.3} = 1.427$$

$$\frac{F_{u_{convdiv}}}{\dot{m}_8 a_8} = \frac{V_9}{a_8} \qquad \frac{T_9}{T_{t9}} = \left(\frac{p_9}{p_{t9}}\right)^k = \left(\frac{p_9}{p_{t7}}\frac{p_{t7}}{p_{t9}}\right)^k = \left(\frac{p_0}{p_{t7}}\frac{p_{t7}}{p_{t9}}\right)^k = \left(\frac{1}{NPR}\frac{1}{\pi_n}\right)^k = \left(\frac{1}{4.2}\frac{1}{0.98}\right)^{0.231}$$
$$= 0.721$$

$$T_9 = \frac{T_9}{T_{t9}} T_{t7} = 0.721 \cdot 939 = 677 \cdot K \qquad a_9 = \sqrt{\gamma R T_9} = \sqrt{1.3 \cdot 291 \cdot 677} = 506 \cdot \frac{m}{s}$$

$$\frac{T_{t9}}{T_9} = \psi_9 = 1 + \frac{\gamma - 1}{2} M_9^2 \qquad M_9 = \sqrt{\frac{2}{\gamma - 1}(\psi_9 - 1)} = \sqrt{\frac{2}{0.3}\left(\frac{1}{0.721} - 1\right)} = 1.606$$

$$V_9 = M_9 a_9 = 1.606 \cdot\ 506 = 813 \cdot \frac{m}{s}$$

$$\frac{F_{u_{conv_{div}}}}{\dot{m}_8 a_8} = \frac{V_9}{a_8} = \frac{813}{556} = 1.462 \qquad \%F = \frac{F_{u_{conv_{div}}} - F_{u_{conv}}}{F_{u_{conv}}} = \frac{1.462 - 1.427}{1.427} = 2.45\%$$

(f) nozzle discharge coefficient C_{D8}

$$C_{D8} = \frac{\dot{m}_8}{\dot{m}_{8ideal}} = \pi_n = 0.98$$

(g) draw a qualitative wave pattern in the exhaust plume

6.43			
Input	γ	1.3	
Input	R	291	J/K
	k	0.230769	
Input	M	1	
	ψ	1.15	
	pt/p	1.832416	
Input	π_{cn}	0.98	
	NPRcr	1.869812	
Input	NPR	4.2	
Input	Tt7	939	K
Input	p0	100	kPa
	pt7	420	
Con perdit	pt9	411.6	
Output	p9	224.6215	
Output	T9	816.5217	
Output	a9	555.7789	
Output	η_n	0.969924	
	V9s	564.3301	
	F/m8a8	1.426774	
ideale	p9s	229.2056	
	T9	816.5217	
	a9	555.7789	
ConDiv Ide	T9/Tt7	0.721436	
	T9	677.4287	
	a9	506.2324	
	M9	1.604418	
	V9	812.2086	
	F/m8a8	1.461388	
	%	2.426007	

Farokhi problem 6.44

6.44 A convergent–divergent nozzle has a conical exhaust shape with the half-cone angle of $\alpha = 25°$. Calculate the divergence loss C_A for this nozzle due to nonaxial exhaust flow. Assuming the same (half) divergence angle of 25°, but in a 2D rectangular nozzle, calculate the flow angularity loss and compare it to the conical case.

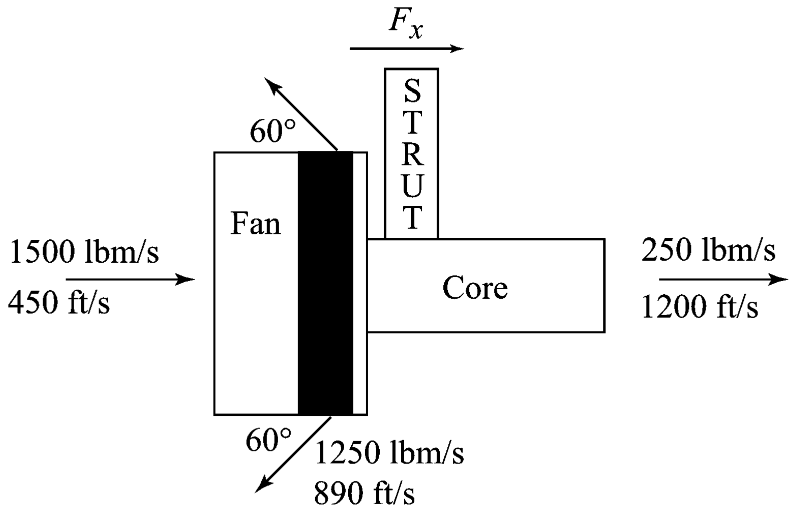
$$C_{A3D} = \frac{1+\cos(\alpha)}{2} = \frac{1+\cos(25)}{2} = 0.953$$

$$C_{A2D} = \frac{\sin(\alpha)}{\alpha} = \frac{\sin(25)}{25\frac{\pi}{180}} = 0.969$$

6.44		
Input	alpha	25
	CA	0.953154
	CA2D	0.96857
	%	1.617348

Mattingly 2.6

One method of reducing an aircarft's landing distance is through the use of thrust reversers. Consider the turbofan engine in Fig. P2.5 with thrust reverser of the bypass airstream. It is given that 1500 lbm/s of air at 60°F and 14.7 psia enters the engine at a velocity of 450 ft/s and that 1250 lbm/s of bypass air leaves the engine at 60 deg to the horizontal, velocity of 890 ft/s, and pressure of 14.7 psia. The remaining 250 lbm/s leaves the engine core at a velocity of 1200 ft/s and pressure of 14.7 psia. Determine the force on the strut F_x. Assume the outside of the engine sees a pressure of 14.7 psia.



$$D_{ram} = \dot{m}_{air} V_0 = 1500 \cdot 450 = 675 \cdot 10^3 \frac{lbm \cdot ft}{s^2} = 675 \cdot 10^3 \cdot \frac{0.454 \cdot kg \cdot 0.305 \cdot m}{s^2}$$

$$= 675 \cdot 10^3 \cdot 0.1383 \frac{kg \cdot m}{s^2} = 93.4 \cdot kN$$

$$F_{core} = \dot{m}_9 V_9 = 250 \cdot 1200 = 300 \cdot 10^3 \frac{lbm \cdot ft}{s^2} = 41.5 \cdot kN$$

$$F_{by} = -\dot{m}_{19} V_{19} \cos(60) = 1250 \cdot \frac{890}{2} = -556 \cdot 10^3 \frac{lbm \cdot ft}{s^2} = -76.9 \cdot kN$$

$$F = F_{by} + F_{core} - D_{ram} = -556 + 300 - 675 = -931 \cdot 10^3 \frac{lbm \cdot ft}{s^2} = -128.8 \cdot kN$$

m13(lbm/s	1,250	m13(lbm/s	1,250	m2(kg/s)	566.99
m3(lbm/s)	250	mf(lbm/s)	250	mf(kg/s)	113.398
V9(ft/s)	1,200			V9(m/s)	366
V19(ft/s)	890			V19(m/s)	271
V2(ft/s)	450			V2(m/s)	137
α	60				
m2	1,500			m2	680
Dram (lb ft	675,000	Dram((lbf)	20,979	Dram(N)	93,322
F_{Fan}(lb ft/s2	556,250	F_{Fan}(lbf)	17,288	F_{Fan}(N)	76,904
F_{main}(lb ft/s	300,000	F_{main}(lbf)	9,324	F_{main}(N)	41476.27
F(lb ft/s2)	931,250	F(lbf)	28,943	F(N)	90297.3