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**The advanced thrust-vectoring exhaust system**

**F119-PW-100. Source: Reproduced by permission of United Technologies Corporation, Pratt & Whitney**

**F-22 Raptor air-superiority fighter. Courtesy of USAF**

# Problems

In solving the following problems, assume $\gamma = 1.4$ and $c_p = 1004$ J/kg K, unless otherwise stated.

**6.1** In a real inlet, the total pressure loss is 10% of the flight dynamic pressure, i.e., $p_{t0} - p_{t2} = 0.1\, q_0$ at a flight Mach number of $M_0 = 0.85$. Calculate

**(a)** inlet total pressure recovery $\pi_d$

**(b)** inlet adiabatic efficiency $\eta_d$.

**6.2** A subsonic inlet in cruise condition is shown. Calculate

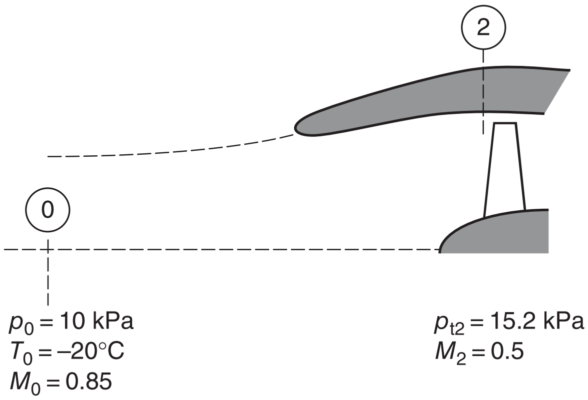
**(a)** total pressure recovery $\pi_d$

**(b)** area ratio $A_2/A_0$

**(c)** static pressure ratio $p_2/p_0$

**(d)** adiabatic efficiency $\eta_d$

**(e)** entropy rise $\Delta_s/R$.




■ **FIGURE P6.2**

**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

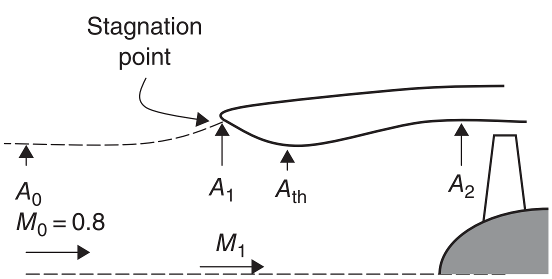
**(a)** $C_p$ (i.e., the pressure coefficient) at the stagnation point

**(b)** inlet lip Mach number $M_1$

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

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

**(e)** the nondimensional inlet additive drag $D_{add}/p_0A_1$.




■ **FIGURE P6.3**

**6.4** The captured streamtube for a subsonic inlet experiences external diffusion, where flight Mach number of $M_0 = 0.85$ decelerates to $M_1$ of 0.65 at the inlet lip. Calculate the inlet additive drag nondimensionalized by flight static pressure $p_0$ and inlet area $A_1$.

**6.5** An aircraft flies at an altitude where $p_0$ is 0.1915 atm and the flight total pressure $p_{t0}$ is 1.498 atm. The engine face total pressure is measured to be, $p_{t2} = 1.348$ atm. For this inlet calculate

**(a)** $\tau_r$, ram temperature ratio

**(b)** $\frac{\Delta s_d}{R}$, nondimensional entropy rise in the diffuser

**(c)** $\eta_d$, inlet adiabatic efficiency

**6.6** A subsonic diffuser has an area ratio of $A_2/A_1 = 1.3$. The inlet Mach number to the diffuser is $M_1 = 0.72$ and the total pressure loss in the diffuser is characterized by $\Delta p_t = 0.1\, q_1$. Assuming the flow in the diffuser is adiabatic, calculate

**(a)** the diffuser exit Mach number $M_2$ and

**(b)** diffuser static pressure recovery $C_{PR}$

**6.7** A subsonic inlet is cruising at $M_0 = 0.85$ and the capture area ratio $A_0/A_1$ is 0.90 (as shown). For an altitude pressure of $p_0 = 25$ kPa, the temperature of $T_0 = -25°C$, and inlet area $A_1 = 3\ m^2$, calculate

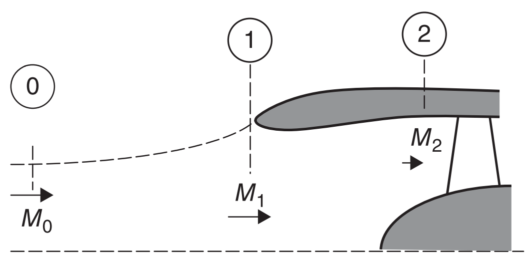
**(a)** the inlet Mach number $M_1$

**(b)** the inlet additive drag $D_{add}$ (N)

**(c)** inlet mass flow rate $\dot{m}$ (kg/s)

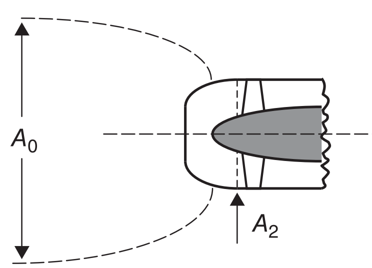
**(d)** the inlet ram drag $D_{ram}$ (kN)

**(e)** engine face area $A_2$, if $M_2 = 0.5$ (assuming $\pi_d = 0.99$)



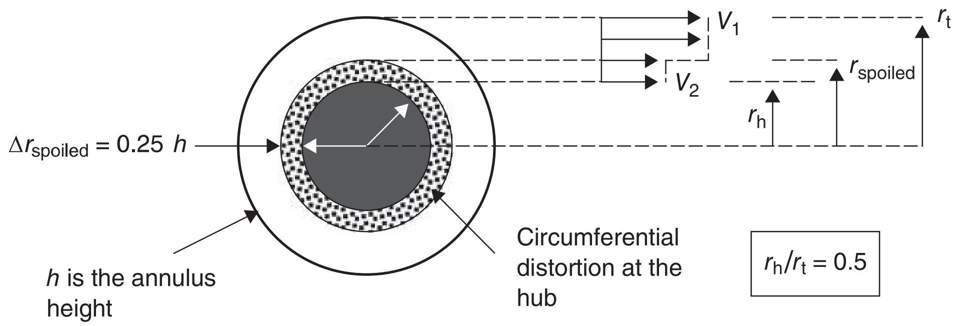

■ **FIGURE P6.7**

**6.8** The Mach number at the compressor face is $M_2 = 0.65$ at takeoff ($M_0 \cong 0.2$). Assuming the inlet suffers a 2% total pressure loss at takeoff, calculate the capture-to-engine face area ratio $A_0/A_2$.




■ **FIGURE P6.8**

**6.9** A subsonic aircraft cruises at $M_0 = 0.85$ and its inlet operates with a capture ratio of $A_0/A_1 = 0.7$. First, calculatethe lip Mach number $M_1$. Second, assuming an engine becomes inoperative and the inlet lip Mach number drops to 0.3 (the so-called engine wind-milling condition), calculate the additive drag $D_{add}$ for an inlet area of $A_1 = 4\ m^2$ and the ambient static pressure of $p_0 = 16.6$ kPa.




■ **FIGURE P6.12**

**6.10** A subsonic inlet is flying at Mach 0.8, with an inlet capture area ratio of $A_0/A_1 = 0.7$. The inlet lip contraction ratio $A_1/A_{th}$ is 1.15. Calculate the 1D throat Mach number and comment on the potential shock formation, near the convex surface, at the throat.

**6.11** A subsonic aircraft flies at $M_0 = 0.85$ with an inlet mass flow ratio (MFR) of 0.90. Calculate the critical pressure coefficient $C_{p,crit}$ on the nacelle. Also calculate the maximum cowl (frontal) area ratio $A_M/A_1$ if this inlet is to experience an average surface pressure coefficient corresponding to the critical value, i.e., $\bar{C}_p \approx C_{p,crit}$.

**6.12** An inlet creates a circumferential distortion at the engine face, as shown. The hub-to-tip radius ratio is $r_h/r_t = 0.5$. The spoiled sector has a 15% mass flow deficit (per unit area) as compared with a uniform flow. Assuming the static density, temperature, and pressure are uniform at the engine face and the mass flow deficit in the spoiled sector is caused by a velocity deficit (as shown), use Bernoulli equation to estimate the total pressure deficit in the spoiled sector, i.e., $[p_t - p]_{spoiled}/[p_t - p]_{uniform.}$ Also calculate

**(a)** $\overline{P}_{t,area-avg}$

**(b)** $\overline{P}_{t,mass-avg}$

**6.13** A subsonic inlet has a capture area ratio of $A_0/A_1 = 0.8$. Assuming the flight Mach number is 0.80 and the inlet area ratio is $A_2/A_1 = 1.25$, calculate

**(a)** Mach number at the inlet lip $M_1$

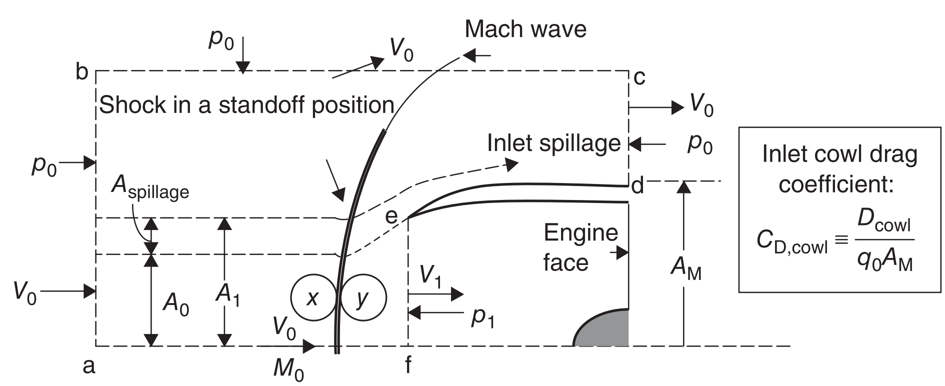
**(b)** diffuser exit Mach number if $p_{t2}/p_{t1}$ is 0.95

**(c)** inlet static pressure ratios $p_1/p_0$ and $p_2/p_1$ for $p_{t2}/p_{t1} = 0.95$

**6.14** A normal shock inlet operates in the subcritical mode, with the shock in standoff position, as shown. The bow shock is normal to the flow at the inlet centerline and weakens into an oblique shock and eventually a Mach wave away from the inlet.

Apply conservations of mass and momentum to the control volume a-b-c-d-e-f-a, to approximate inlet cowl external drag force coefficient, in terms of the flight and inlet parameters that are shown, for example $V_0$, $M_0$, $p_0$, $p_1$, $V_1$, $A_1$, $A_M$, and $A_{spillage}$.

**6.15** Consider a variable-geometry, convergent–divergent isentropic inlet that is designed for $M_D = 4.0$. To swallow the starting shock, i.e., to start the inlet, the throat needs to be opened. Calculate the percentage of throat area opening needed to start this inlet, $(\Delta A_{th}/A_{th}) \times 100$.

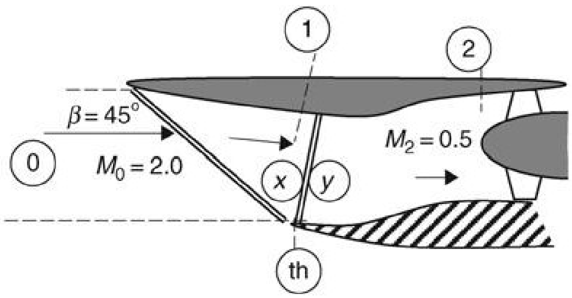



■ **FIGURE P6.14**

**6.16** An external compression inlet is in a Mach-2 flow. The shocks are positioned according to the figure shown. Calculate

- **(a)** shock total pressure ratio $\pi_s$
- **(b)** $A_2/A_{th}$

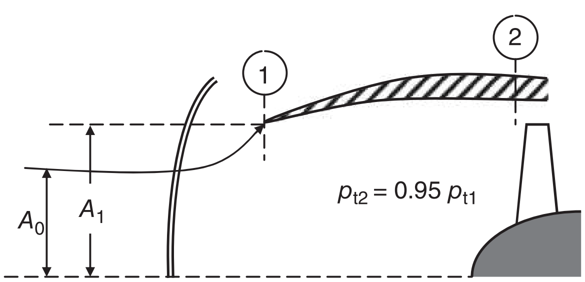
You may neglect the frictional losses in the subsonic diffuser.




■ **FIGURE P6.16**

**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 $A_2 = A_1 = 1{:}2$. Calculate

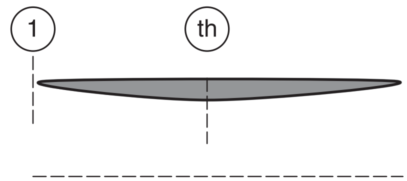
- **(a)** $M_1$
- **(b)** inlet total pressure recovery $\pi_d$, i.e., $p_{t2}/p_{t0}$




■ **FIGURE P6.17**

**6.18** An isentropic convergent–divergent supersonic inlet is designed for $M_D = 1.6$. Calculate the inlet's

- **(a)** area contraction ratio $A_1/A_{th}$
- **(b)** subsonic Mach number where the throat first chokes
- **(c)** percent spillage at $M_0 = 0.7$
- **(d)** percent spillage at $M_0 = 1.6$ (in the unstarted mode)
- **(e)** overspeed Mach number to start this inlet, $M_{overspeed}$
- **(f)** throat Mach number after the inlet was started, with still $M_{overspeed}$ as the flight Mach number




A fixed-geometry inlet

■ **FIGURE P6.18**

**6.19** Consider an isentropic fixed-geometry C–D inlet, which is designed for $M_D = 1.75$. The inlet flies at an altitude where ambient (static) pressure is 20 kPa. Calculate

- **(a)** Overspeed Mach number that will start this inlet
- **(b)** The flight dynamic pressure corresponding to the altitude and $M_{overspeed}$

**6.20** An isentropic, convergent–divergent supersonic inlet is designed for $M_D = 3.0$. Assuming that the throat area is adjustable, calculate the percentage of the design throat area that needs to be opened to swallow the starting shock, i.e., $\frac{A_{th,open} - A_{th,design}}{A_{th,design}} \times 100$

**6.21** A supersonic C–D inlet is designed for a flight Mach number of $M_0 = 3.5$. This inlet starts by opening its throat (from $A_{th}$ to $A'_{th}$). Neglecting wall frictional losses, calculate

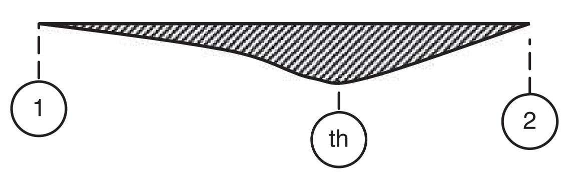
- **(a)** percentage throat opening required
- **(b)** throat Mach number after it starts (with the throat open at $A'_{th}$

**6.22** A Kantrowitz–Donaldson inlet is designed for Mach 2.0. Calculate

- **(a)** the required contraction area ratio $A_1/A_{th}$
- **(b)** the inlet total pressure recovery with the best back-pressure

**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.




■ **FIGURE P6.23**

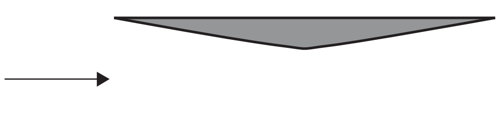
**6.24** Calculate the contraction ratio $A_1/A_{th}$ and the maximum total pressure recovery of a self-starting C–D inlet designed for $M_D = 3.2$.

**6.25** A normal-shock inlet is operating in a supercritical mode, with the shock inside the inlet. If the flight Mach number is $M_0 = 1.6$ and the shock is located at $A_s/A_t = 1.2$, calculate

- **(a)** Mach number ahead of the shock wave, $M_x$
- **(b)** percentage total pressure gain if the inlet were to operate in the critical mode

**6.26** A variable geometry isentropic supersonic inlet is designed for $M_D = 1.6$. Calculate

- **(a)** percentage flow spillage at $M_0 = 0.8$
- **(b)** percentage flow spillage at $M_D = 1.6$ before the inlet is started
- **(c)** percentage throat area increase needed to start this inlet
- **(d)** throat Mach number after the shock is swallowed, $M_{th}$
- **(e)** inlet total pressure recovery with the best backpressure (with open throat)



■ **FIGURE P6.26**

**6.27** An isentropic, fixed-geometry inlet, is designed for $M_D = 1.5$. If this inlet is to be started by overspeeding, calculate the necessary Mach number for overspeed.

**6.28** A fixed-geometry, convergent–divergent, internal-compression inlet is designed for $M_D = 2.0$ and a self-starting capability. Calculate

- **(a)** $A_1/A_{th}$
- **(b)** $M_{th}$
- **(c)** inlet total pressure recovery for the "best" backpressure

**6.29** A normal-shock inlet is flying at a Mach number of 1.8. However, due to a nonoptimum backpressure, the normal shock is inside the duct where $A_s/A_i = 1.15$. Calculate the percentage loss in the total pressure recovery due to this back pressure.

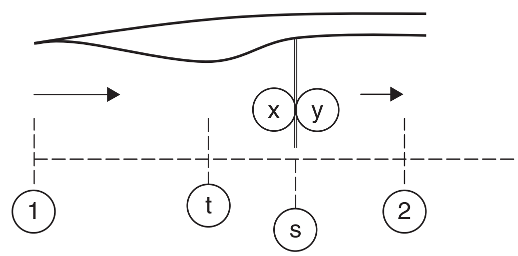
**6.30** A variable-geometry supersonic convergent–divergent inlet is to be designed for an isentropic operation (in the started mode) at $M_D = 2.6$. Calculate

- **(a)** the inlet design contraction ratio $A_1/A_{th}$
- **(b)** the percentage opening of the throat $(A'_{th} - A_{th})/A_{th}$ needed to start the inlet
- **(c)** the throat Mach number in the open position, $M'_{th}$

**6.31** A variable-geometry, internal-compression, C–D inlet is designed for $M = 2.0$. Calculate percentage opening of the throat required to swallow the starting shock.

**6.32** A 2D fixed geometry convergent–divergent diffuser is shown. Mach number at the entrance is $M_1 = 3.0$ and the static pressure is $p_t = 10$ kPa. A shock occurs at an area ratio $A_s/A_{th} = 1.25$ downstream of the throat, as shown. Calculate

- **(a)** $M_x$
- **(b)** $P_y$
- **(c)** $P_{ty}$
- **(d)** $A_2/A_1$ if the exit Mach number is 0.5, i.e., $M_2 = 0.5$



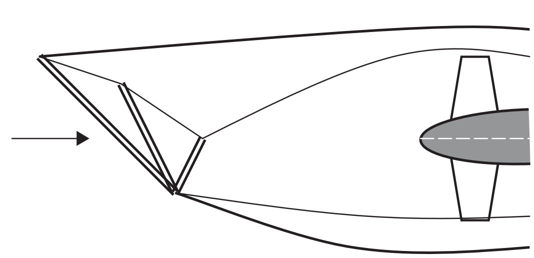

■ **FIGURE P6.32**

You may assume the throat is choked and neglect wall friction.

**6.33** A normal-shock inlet operates in a Mach 1.86 stream with ambient static pressure of $p_0 = 30$ kPa. Neglecting total pressure loss in the subsonic diffuser, calculate

- **(a)** inlet total pressure recovery with the shock at the lip, i.e., the best backpressure
- **(b)** inlet total pressure recovery when the shock is inside the duct at $A_x/A_t = 1.2$, i.e., the supercritical mode
- **(c)** inlet total pressure recovery in subcritical mode with 10% spillage, i.e., $A_{slage}/A_1 = 0.1$
- **(d)** flight dynamic pressure $q_0$

**6.34** Consider an external compression inlet with two ramps operating in a Mach-2.5 stream of air. Calculate the total pressure recovery of the inlet shock system for the case of the best backpressure for the two ramp angles of 8° and 12°, respectively, and compare it to the normal-shock inlet at the same Mach number and with the best backpressure.



■ **FIGURE P6.34**

**6.35** A supersonic flow is to be decelerated over two oblique shocks and one normal shock, similar to Problem