مشخصه های عملکرد هواپیما

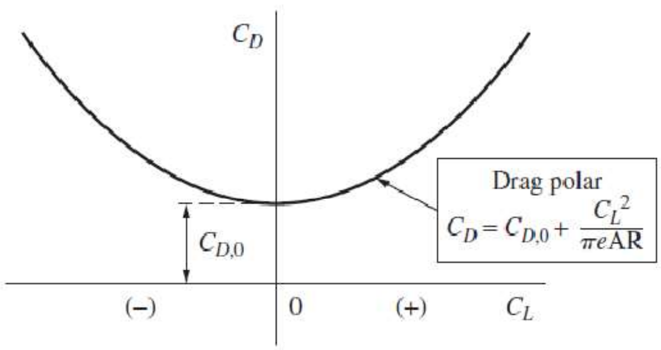
In previous chapters the physical phenomena producing lift, drag, and moments of an airplane were introduced. We emphasized that the aerodynamic forces and moments exerted on a body moving through a fluid stem from two sources, both acting over the body surface:

1. The pressure distribution.
2. The shear stress distribution.

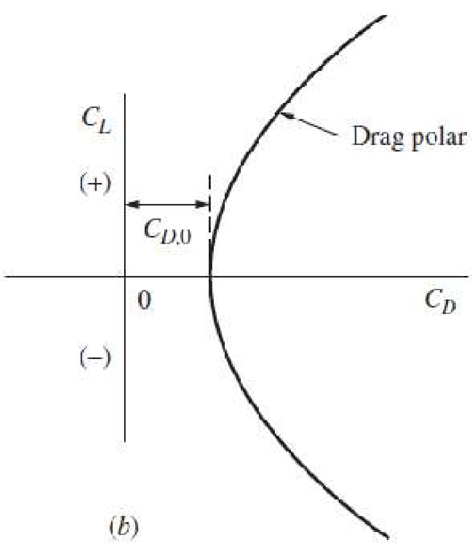
The physical laws governing such phenomena were examined, with various applications to aerodynamic flows.

در فصل قبل با استفاده از نیروهای آیرودینامیکی برای ضریب پسا به معادله قطبی پسا رسیدیم.

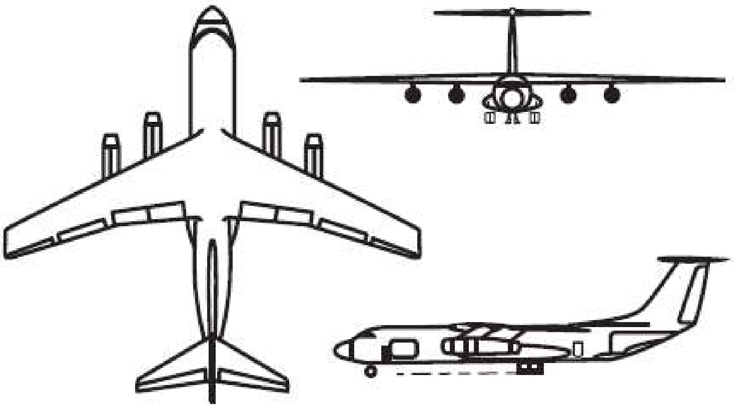
$$C_D = C_{D,e} + \frac{C_L^2}{\pi e \mathrm{AR}}$$



(a)



(b)

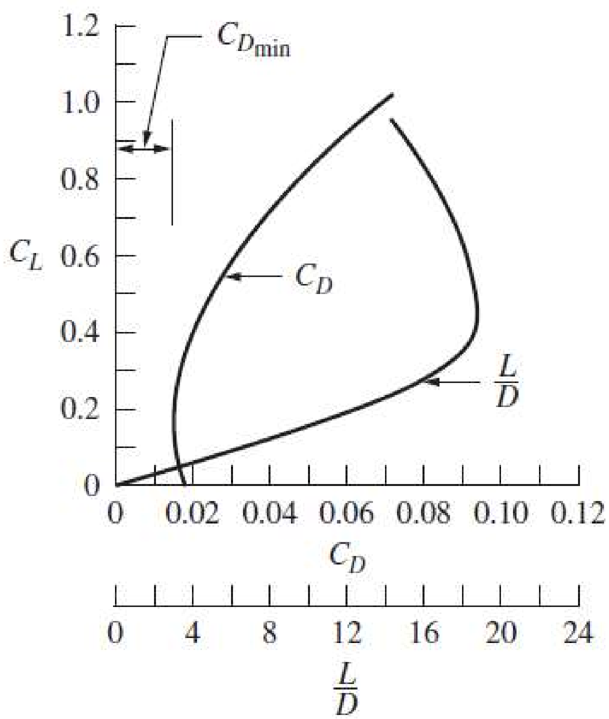


Lockheed C-141A.

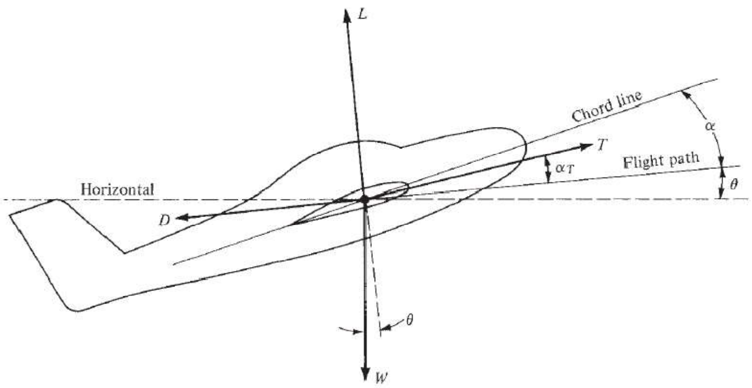
$$C_{D,\min} = 0.015$$

$$C_{L_{\text{min drag}}} = 0.16$$

$$C_{D,0} = 0.017 \text{ at } C_L = 0$$



$$C_D = C_{D,\min} + \frac{\left(C_L - C_{L_{\text{min drag}}}\right)^2}{\pi e \mathrm{AR}}$$



$$\sum F_{\parallel} = ma = m\frac{dV}{dt}$$

$$\sum F_{\perp} = m\frac{V^2}{r_c}$$

$$\sum F_{\parallel} = T\cos\alpha_T - D - W\sin\theta$$

$$\sum F_{\perp} = L + T\sin\alpha_T - W\cos\theta$$

$$\boxed{T\cos\alpha_T - D - W\sin\theta = m\frac{dV}{dt}}$$

$$\boxed{L + T\sin\alpha_T - W\cos\theta = m\frac{V^2}{r_c}}$$

consider level, unaccelerated flight.

$$\boxed{T\cos\alpha_T = D}$$

$$\boxed{L + T\sin\alpha_T = W}$$

α_T is small enough that $\cos \alpha_T \approx 1$ and $\sin \alpha_T \approx 0$.

$$\boxed{T = D}$$

$$\boxed{L = W}$$

THRUST REQUIRED FOR LEVEL UNACCELERATED FLIGHT

$$T = D = q_\infty S C_D$$

$$L = W = q_\infty S C_L$$

$$\frac{T}{W} = \frac{C_D}{C_L}$$

$$\boxed{T_R = \frac{W}{C_L/C_D} = \frac{W}{L/D}}$$

1. Choose a value of V_∞.
2. For this V_∞, calculate the lift coefficient from

$$C_L = \frac{W}{\frac{1}{2}\rho_\infty V_\infty^2 S}$$

Note that ρ_∞ is known from the given altitude and S is known from the given airplane. The C_L calculated from Eq. (6.17) is the value necessary for the lift to balance the known weight W of the airplane.

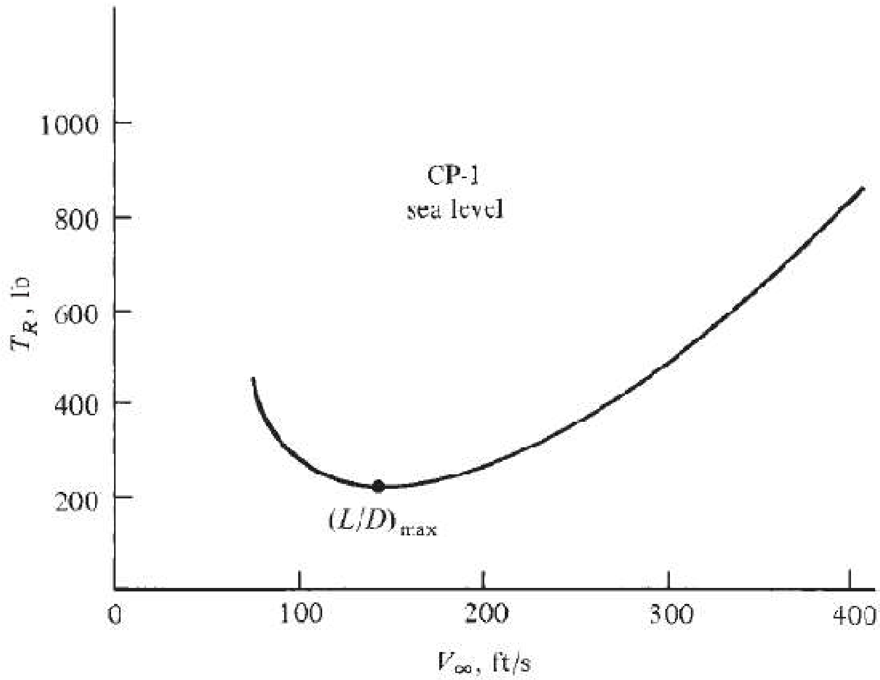
3. Calculate C_D from the known drag polar for the airplane

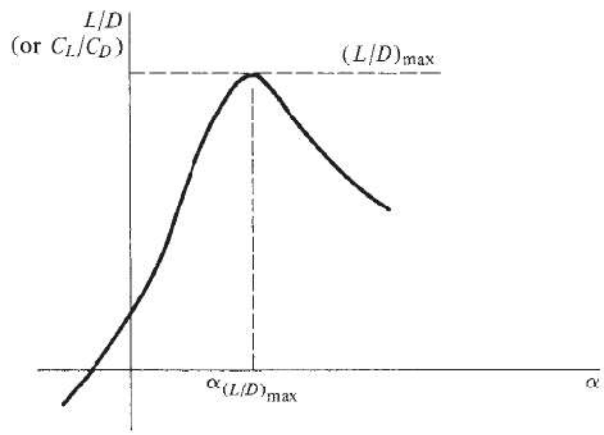
$$C_D = C_{D,0} + \frac{C_L^2}{\pi e \mathrm{AR}}$$

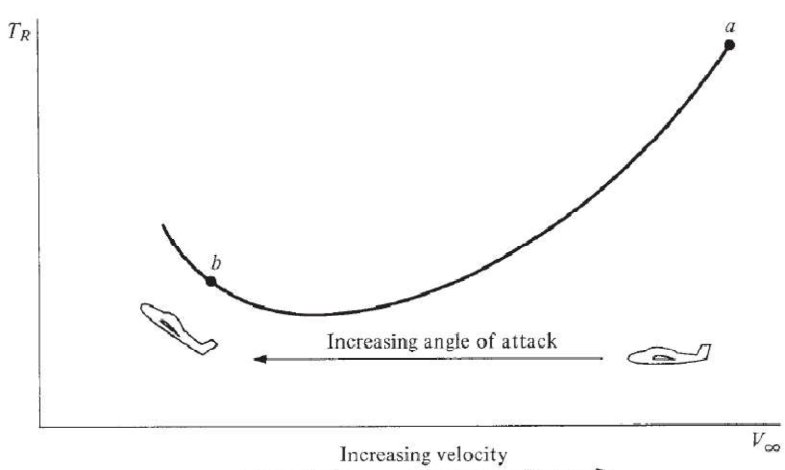
where C_L is the value obtained

4. Form the ratio C_L/C_D.

5. Calculate thrust required


1000
800
600
400
200
T_R, lb
CP-1
sea level
$(L/D)_{max}$
0
100
200
300
400
V_∞, ft/s

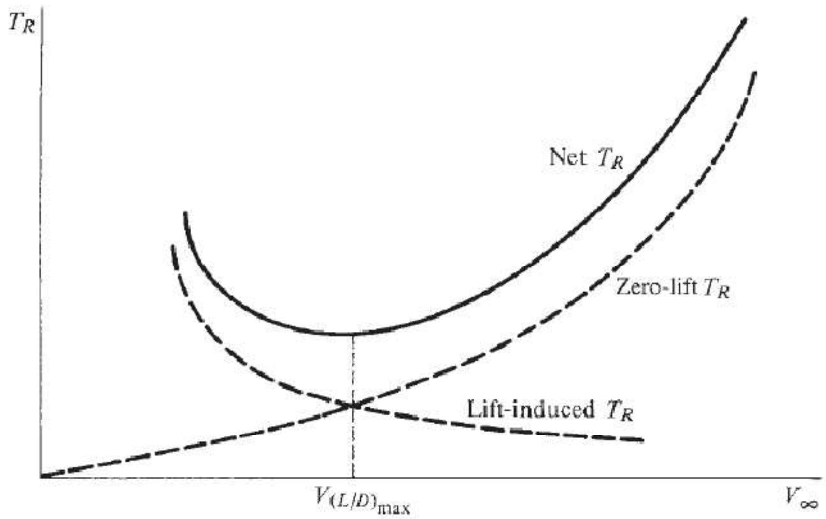

L/D
(or C_L/C_D)
$(L/D)_{max}$
$\alpha_{(L/D)_{max}}$
α


T_R
a
b
Increasing angle of attack
V_∞
Increasing velocity

$$T_R = D = q_\infty S C_D = q_\infty S (C_{D,0} + C_{D,i})$$

$$= q_\infty S \left(C_{D,0} + \frac{C_L^2}{\pi e \mathrm{AR}} \right)$$

$$\boldsymbol{T_R} = \underbrace{q_\infty S C_{D,0}}_{\text{Zero-lift } T_R} + \underbrace{q_\infty S \frac{C_L^2}{\pi e \mathrm{AR}}}_{\text{Lift-induced } \boldsymbol{T_R}}$$



$$C_L = W/(q_\infty S)$$

$$T_R = q_\infty S C_{D,0} + \frac{W^2}{q_\infty S \pi e \mathrm{AR}}$$

$$\frac{dT_R}{dq_\infty} = \frac{dT_R}{dV_\infty} \frac{dV_\infty}{dq_\infty}$$

برای محاسبه مقدار مینیمم رانش مورد نیاز می توان نوشت:

$$\frac{dT_R}{dq_\infty} = SC_{D,0} - \frac{W^2}{q_\infty^2 S \pi e \mathrm{AR}} = 0$$

$$C_{D,0} = \frac{W^2}{q_\infty^2 S^2 \pi e \mathrm{AR}}$$

$$\frac{W^2}{q_\infty^2 S^2} = \left(\frac{W}{q_\infty S}\right)^2 = C_L^2$$

$$\boxed{\begin{array}{c} C_{D,0} = \dfrac{C_L^2}{\pi e \mathrm{AR}} = C_{D,i} \\ \text{Zero-lift drag = drag due to lift} \end{array}}$$

EXAMPLE 6.1

For all the examples given in this chapter, two types of airplanes will be considered:
a. A light, single-engine, propeller-driven, private airplane, approximately modeled after the Cessna T-41A shown in Fig. 6.10. For convenience, we will designate our hypothetical airplane as the CP-1, having the following characteristics:

Wingspan = 35.8 ft

Wing area = 174 ft^2

Normal gross weight = 2950 lb

Fuel capacity: 65 gal of aviation gasoline

Power plant: one-piston engine of 230 hp at sea level

Figure 6.10 The hypothetical CP-1 studied in Ch. 6 sample problems is modeled after the Cessna T-41A shown here.
(Source: *U.S. Air Force.*)

Specific fuel consumption = 0.45 lb/(hp)(h)

Parasite drag coefficient $C_{D,0} = 0.025$

Oswald efficiency factor $e = 0.8$

Propeller efficiency = 0.8

b. A jet-powered executive aircraft, approximately modeled after the Cessna Citation X, shown in Fig. 6.11. For convenience, we will designate our hypothetical jet as the CJ-1, having the following characteristics:

Wingspan = 53.3 ft

Wing area = 318 ft^2

Normal gross weight = 19,815 lb

Fuel capacity: 1119 gal of kerosene

Power plant: two turbofan engines of 3650 lb thrust each at sea level

Specific fuel consumption = 0.6 lb of fuel/(lb thrust)(h)

Parasite drag coefficient $C_{D,0} = 0.02$

Oswald efficiency factor $e = 0.81$

By the end of this chapter, all the examples taken together will represent a basic performance analysis of these two aircraft.

In this example, only the thrust required is considered. Calculate the T_R curves at sea level for both the CP-1 and the CJ-1.

Figure 6.11 The hypothetical CJ-1 studied in Ch. 6 sample problems is modeled after the Cessna Citation X shown here.
(Source: © *RGB Ventures LLC dba SuperStock/Alamy.*)

■ **Solution**

a. For the CP-1, assume that $V_\infty = 200$ ft/s = 136.4 mi/h. From Eq. (6.17),

$$C_L = \frac{W}{\frac{1}{2}\rho_\infty V_\infty^2 S} = \frac{2950}{\frac{1}{2}(0.002377)(200)^2(174)} = 0.357$$

The aspect ratio is
$$\text{AR} = \frac{b^2}{S} = \frac{(35.8)^2}{174} = 7.37$$

Thus, from Eq. (6.1*c*),

$$C_D = C_{D,0} + \frac{C_L^2}{\pi e \text{AR}} = 0.025 + \frac{(0.357)^2}{\pi(0.8)(7.37)} = 0.0319$$

Hence
$$\frac{L}{D} = \frac{C_L}{C_D} = \frac{0.357}{0.0319} = 11.2$$

Finally, from Eq. (6.16),

$$T_R = \frac{W}{L/D} = \frac{2950}{11.2} = \boxed{263 \text{ lb}}$$

To obtain the thrust-required curve, the preceding calculation is repeated for many different values of V_∞. Some sample results are tabulated as follows:

V_∞, ft/s	C_L	C_D	L/D	T_R, lb
100	1.43	0.135	10.6	279
150	0.634	0.047	13.6	217
250	0.228	0.028	8.21	359
300	0.159	0.026	6.01	491
350	0.116	0.026	4.53	652

The preceding tabulation is given so that the reader can try such calculations and compare the results. Such tabulations are given throughout this chapter. They are taken from a computer calculation in which 100 different velocities were used to generate the data. The T_R curve obtained from these calculations is given in Fig. 6.6.

b. For the CJ-1, assume that $V_\infty = 500$ ft/s = 341 mi/h. From Eq. (6.17),

$$C_L = \frac{W}{\frac{1}{2}\rho_\infty V_\infty^2 S} = \frac{19{,}815}{\frac{1}{2}(0.002377)(500)^2(318)} = 0.210$$

The aspect ratio is $$\mathrm{AR} = \frac{b^2}{S} = \frac{(53.3)^2}{318} = 8.93$$

Thus, from Eq. (6.1*c*),

$$C_D = C_{D,0} + \frac{C_L^2}{\pi e \mathrm{AR}} = 0.02 + \frac{(0.21)^2}{\pi(0.81)(8.93)} = 0.022$$

Hence $$\frac{L}{D} = \frac{C_L}{C_D} = \frac{0.21}{0.022} = 9.55$$

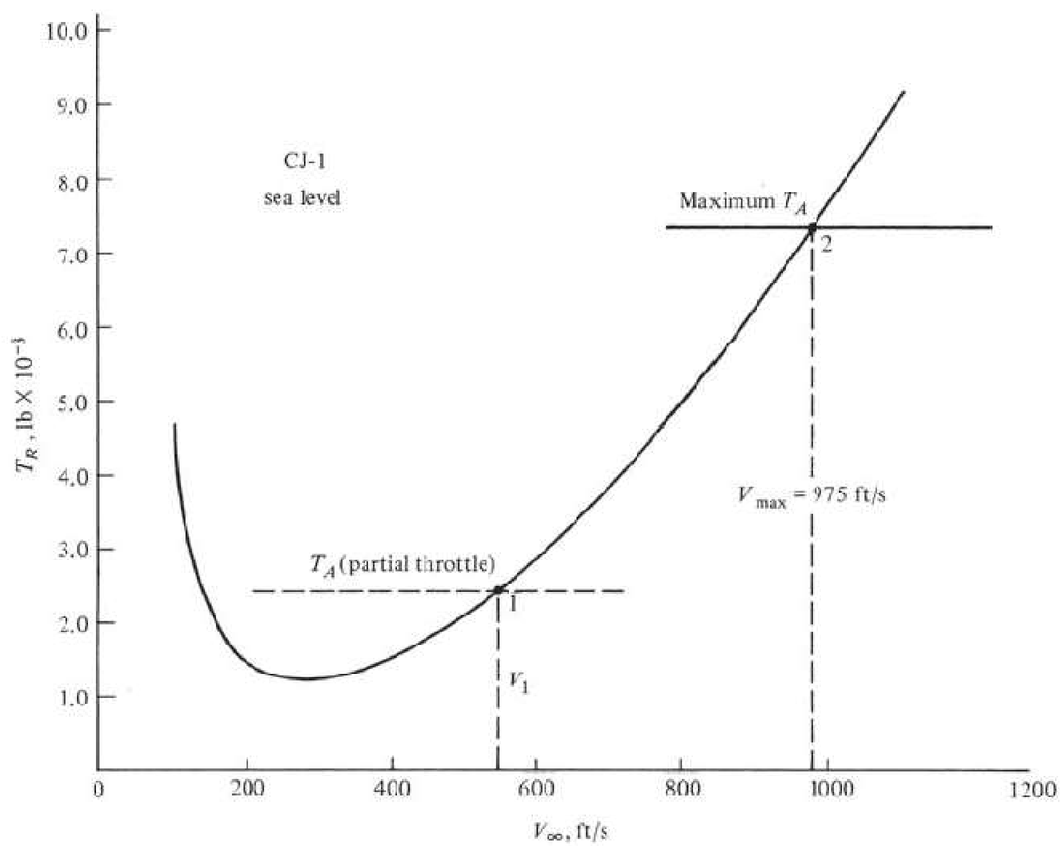


Figure 6.12 Thrust-required curve for the CJ-1.

Finally, from Eq. (6.16),

$$T_R = \frac{W}{L/D} = \frac{19{,}815}{9.55} = 2075 \text{ lb}$$

A tabulation for a few different velocities follows:

V_∞, ft/s	C_L	C_D	L/D	T_R, lb
300	0.583	0.035	16.7	1188
600	0.146	0.021	6.96	2848
700	0.107	0.021	5.23	3797
850	0.073	0.020	3.59	5525
1000	0.052	0.020	2.61	7605

The thrust-required curve is shown in Fig. 6.12.