Chapter 3

3.1 An aircraft weighing 18,000 lbs, with wing area of 350 ft^2, flies at a true air spees (TAS) of 250 knots. Calculate the aircraft lift coefficient C_L and the equivalent airspeed (EAS) V_E if the flight altitude is:

a. sea level

b. 20,000 ft

$$a.TAS = V = 250(1.689) = 422.3\ ft/sec\ EAS = V_E$$

and

$$V_{Esl} \equiv V_{Eo} = V\sqrt{\sigma} = 250(.73) = 182.5\ kn$$

$$\begin{aligned} C_{Lo} &= \frac{2W}{\rho_o V^2 S} \\ &= \frac{2(18,000)}{.002377(422.5)^2 350} = .242 \end{aligned}$$

b. At 20,000 ft $C_{L20} = \frac{C_{Lo}}{\sigma_{20}}$=.242/.5328=.454

3.2 After the aircraft in problem 1 has consumed 3000 lbs of fuel, what value of C_L will be required to fly at the same TAS of 250 knots at 20,000 ft altitude? What is its landing speed at maximum lift coefficient of 1.5?

The lift coefficient at reduced weight may be found from

$$C_L = C_{L20}\frac{15,000}{18,000} = .454\frac{15}{18} = .378$$

The landing speed V_{Emin} is then V_{Es} and is found from

$$V_s = \sqrt{\frac{2W}{\rho_o S C_{Lmax}}} = \sqrt{\frac{2(15,000)}{.002377(350)1.5}} = 155\ ft/sec$$

3.3 It is much safer to fly at a speed marginally above V_{Es} (i.e. $V_{Eland} = 1.2V_{Es}$). For the aircraft in problem 2, determine this speed and calculate the corresponding TAS for landing at an altitude of 6000 ft.

$$V_{land} = 1.2V_s = 1.2(155) = 186\ ft/sec = 110\ knots\ \equiv V_{El}$$

$$V_{land6} = V_{El}/\sqrt{\sigma_6} = 186/.9143 = 203\ ft/sec = 120\ knots$$

3.4 The landing speed of an airplane is 10 mph greater than its stalling speed. $C_{Ll} = 1.5$, $C_{Ls} = 1.8$. Find the stalling speed.

Assuming that W, S and ρ are constant

$$\frac{C_{Ll}}{C_{Ls}} = \frac{V_s^2}{V_l^2}$$

and, from the problem statement

$$V_l = V_s + 10$$

Thus, there are two equations and two unknowns V_l and V_s. Solving then yields V_s=104.8mph.

3.5 An airplane has the following characteristics:

C_{Lmax}=1.9

$\alpha_{L=0}$=-2deg

a=.075 per deg (lift curve slope)

V_s=60 mph at 3000 ft

At what velocity will it fly at sea level if the angle of attack is $\alpha = 1\ deg$?

$$C_L = (\alpha - \alpha_{OL}) = (1 - (-2)).075 = .225$$

Wing loading W/S is obtained from stall data at 3000 ft

$$W/S = V^2 \rho_o \sigma_3 C_{Lmax}/2 = (60x1.467)^2(.002377).9151(1.9)/2 = 16\ lb/ft^2$$

and the velocity is found from

$$v = \sqrt{\frac{2W/S}{\rho_o C_L}} = \sqrt{\frac{2(16)}{.002377(.225)}} = 245 ft/sec$$

3.6 The aircraft described in Example 3.2 is a T2-B jet trainer equipped with two J60-P-6 turbojets and its empty weight is 10,000 lb. The drag polar given in the example is for a clean configuration. If the aircraft is loaded with two racks of rockets (rockets+racks=487 lbs), its parasite area is now 6.25 ft^2 and the span efficiency drops to .82. If it carries 2500 lb fuel, calculate:

a. the new drag polar

b. $\left.\frac{L}{D}\right|_{max}$

c. T_r at minimum drag at 20,000 ft

a. From the definition of the parasite area, f=C_{Do}S, and C_{Do}=6.25/255=.0245,

and k=$\frac{1}{\Pi A\!\!\!R e} = \frac{1}{\Pi(5.07).82} = .0765.$

Thus, C_D=.0245+.0765 C_L^2.

b.

$$\left.\frac{L}{D}\right|_{max} = \frac{1}{2\sqrt{kC_{Do}}} = \frac{1}{2\sqrt{.0765(.0245)}} = 11.55$$

c.

$$T_r = D_{min} = 2W\sqrt{kC_{Do}} = \frac{W}{L/D|_{max}} = \frac{10,000 + 2500 + 487}{11.55} = 1124.4\ lb$$

3.7 The aircraft in the last problem is equipped with two J60-P-6 jet engines and weighs 13,000lbs. The thrust can be approximated by

$T=2600\sigma^m$

m=.072, h <36,000 ft

m=1, $h \geq$ 36,000ft

Find: a. maximum velocity at 25,000ft, b. its ceiling.

a Since the thrust is independent of velocity, maximum velocity can be evaluated in several ways. Here, Eq.3.54 is used. Since the density ratio at 25,000 ft is .448, the thrust is $T=2(2600).448^{.72}$=2917lb. With $L/D|_{max}$=11.55, Eq.3.54 becomes

$$V = \sqrt{\frac{2917}{.002377(.448)255(.0245)}\left[1 + \sqrt{1 - \frac{1}{(11.55(2917/13,000))^2}}\right]} = 917 ft/sec$$

b Ceiling is calculated from Eq.3.49 with the assumption that it occurs at above 36,000 ft. Thus, m=1 and

$$\sigma = \frac{W}{2T_o L/D|_{max}} = \frac{1}{2(2600)11.55} = .217$$

which, from altitude tables, gives for the ceiling approximately 42,500 ft.

3.8 The following data is given for P-3C Orion ASW aircraft:

S = 1300 ft^2

f = 29.1 ft^2

$A\!\!\!R$ = 7.5

W = 128,000 lb

b = 99 ft (wing span)

span efficiency = 0.948 (low speed)

Calculate the thrust horsepower required for 15,000 ft at a speed of 200 knots.

$$P = \frac{DV}{550} = \frac{\sigma\rho_o V^3 f}{1100} + \frac{kW^2}{275\rho_o\sigma VS} \quad k = 1/(\Pi A\!\!\!R e) = 1/(7.5\Pi.948) = .0448$$

$$\begin{aligned} P &= \frac{.669(.002377)(1.69x200)^3 29.1}{1100} + \frac{.0448(128,000)^2}{275(.629).002377(1.69x200)1300} \\ &= 1527 + 4063 = 5588HP \end{aligned}$$

3.9 An aircraft has the following characteristics:

W/S = 40 lb/ft^2
S = 400 ft^2
$C_D = 0.018 + 0.062\ C_L^2$

Calculate:

a. minimum drag speed, EAS (ft/sec)
b. minimum power required speed, EAS (ft/sec)
c. minimum power required
d. minimum thrust required for steady, level flight.

a.

$$V_{DminE} = \sqrt{\frac{2W/S}{\rho}}\left(\frac{k}{C_{Do}}\right)^{.25}\sqrt{\frac{2(40)}{.002377}}\left(\frac{.062}{.018}\right)^{.25} = 250\ ft/sec$$

b.

$$V_{PminE} = \frac{V_{DminE}}{1.316} = \frac{250}{1.316} = 190\ ft/sec$$

c.

$$\begin{aligned} P_{rmin} &= D_{Pmin}V_{Pmin} = C_{DP_{min}}\rho_o V_{Pmin}^3 S/2 = 4C_{Do}\rho_o V_{Pmin}^3 S/2 \\ &= 2(.018).002377(190)^3 400 = 427\ HP \end{aligned}$$

d.

$$T_{rmin} = D_{min} = 2W\sqrt{kC_{Do}} = 2(16,000)\sqrt{.062(.018)} = 1070\ lb$$

3.10 A transport aircraft cruise weight is about 650,000 lb. Its best cruise speed is 625 mph at 42,000 ft. It is equipped with four JTD9-3 engines. Calculate its drag coefficient and estimate the thrust specific fuel consumption. Its wing area is 5500 ft^2.

Appendix D gives the thrust as a function of Mach number. Since the speed of sound at 42,000 ft is 968 ft/sec, the flight Mach number is .947. Engine data can be estimated as T=7000 lb and the specific fuel consumption SFC is about .66. The drag coefficient can be calculated, at four engines needed, as

$$C_D = \frac{4T}{\rho SV^2/2} = \frac{8(7000)}{.002377(.2245)5500(916)^2} = .0227$$

3.12 The clean drag polar of a fighter is

$$C_D = 0.04 + 0.09C_L^2$$

Its landing drag polar (flaps, slats and gear down) is

$$C_D = 0.2 + 0.32C_L^2$$

It is equipped with one engine whose thrust can be approximated by

$$T = 15,000\sigma^n \quad m = 1\ \ h > 36,000ft$$
$$m = 0.8\ \ h < 36,000ft$$

Its maximum lift coefficient is 2.16 and the takeoff speed is the same as its landing speed. Assume that the landing speed is 1.2 V_s. Calculate: