Appendix A

Lecture 36

Performance analysis of a piston engined airplane – 2

Topics

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3 Engine characteristics

Model : Lycoming O-360-A3A.

Type : Air-cooled, carbureted, four-cylinder, horizontally opposed piston engine.

Sea level power : 180 BHP (135 kW)

Propeller : 74 inches (1.88 m) diameter

The variations of power output and fuel consumption with altitude and rpm are shown in Fig.3. For the present calculations, the values will be converted into SI units.

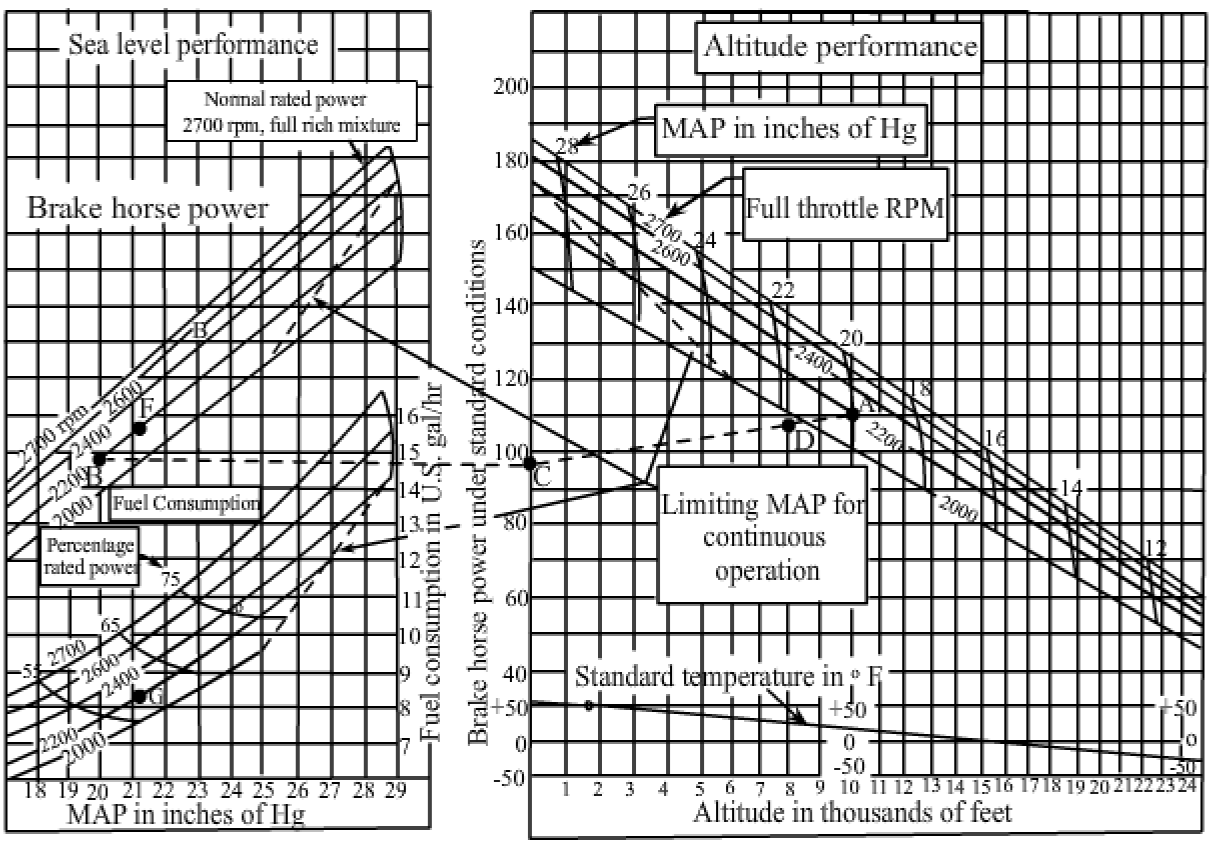

Fig.3 Characteristics of Lycoming O-360-A
(with permission from Lycoming aircraft engines)

3.1 Variation of engine BHP

The variation of available engine output (BHP_a) with altitude is assumed to be of the form:

$$BHP_a = BHP_{sealevel}(1.13\sigma - 0.13)$$

where σ is the density ratio = ρ/ρ_{sL} .

The power outputs of the engine at select altitudes are given in Table 2 and plotted in Fig.4.

Note: At a given altitude, the variation of engine BHP with flight speed is very slight and is generally neglected.

h(m)	σ	BHP_a (kW)
0	1	135.00
1000	0.9075	120.89
2000	0.8217	107.80
3000	0.7423	95.69
4000	0.6689	84.49
5000	0.6012	74.16
5500	0.5691	69.27
6000	0.538	64.52
6500	0.5093	60.14
7000	0.4812	55.86

Table 2 Variation of BHP with altitude

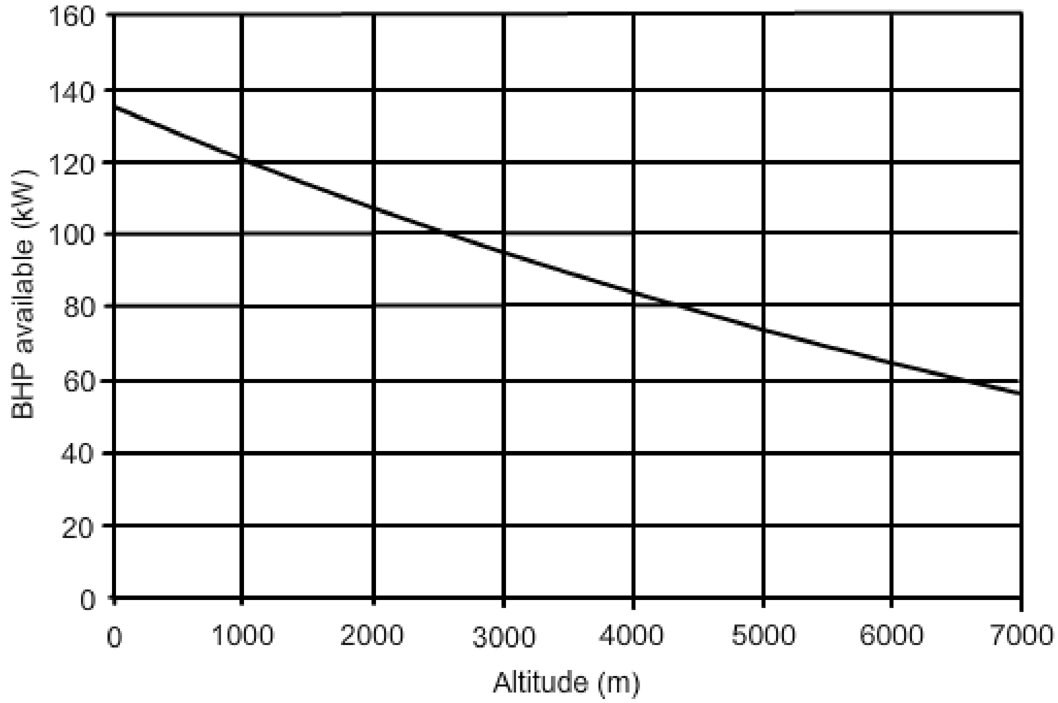

Fig.4 Variation of BHP with altitude at maximum power condition

3.2 Thrust horsepower available

The available thrust horsepower is obtained as product of $BHP_a \times \eta_p$, where η_p is the propeller efficiency. The propeller efficiency (η_p) depends on the flight speed, rpm of the engine and the diameter of the propeller. It can be worked out at different speeds and altitudes using the propeller charts. However, chapter 6 of Ref.2 gives an estimated curve of efficiency as a function of the advance ratio $(J = \frac{V}{nD})$ for the fixed pitch propeller used in the present airplane. This variation is shown as data points in Fig.5.

It may be added that this variation of η_p with J is used in chapter 6 of Ref.2, to estimate the drag of Piper Cherokee airplane from measurements in flight. In another application, in Ref.10, chapter 17, the same variation is used to compare the performance of fixed pitch and variable pitch propellers. Based on these two applications, it is assumed here that the variation of η_p with J shown in Fig.5, can be used at all altitudes and speed relevant to this airplane.

For the purpose of calculating the airplane performance, an equation can be fitted to the η_p vs J curve in Fig.5. A fourth degree polynomial for η_p in terms of J is as follows.

$$\eta_p(J) = -2.071895J^4 + 3.841567J^3 - 3.6786J^2 + 2.5586J - 0.0051668 \quad (14)$$

It is seen that the fit is very close to the data points. The dotted portions are extrapolations.

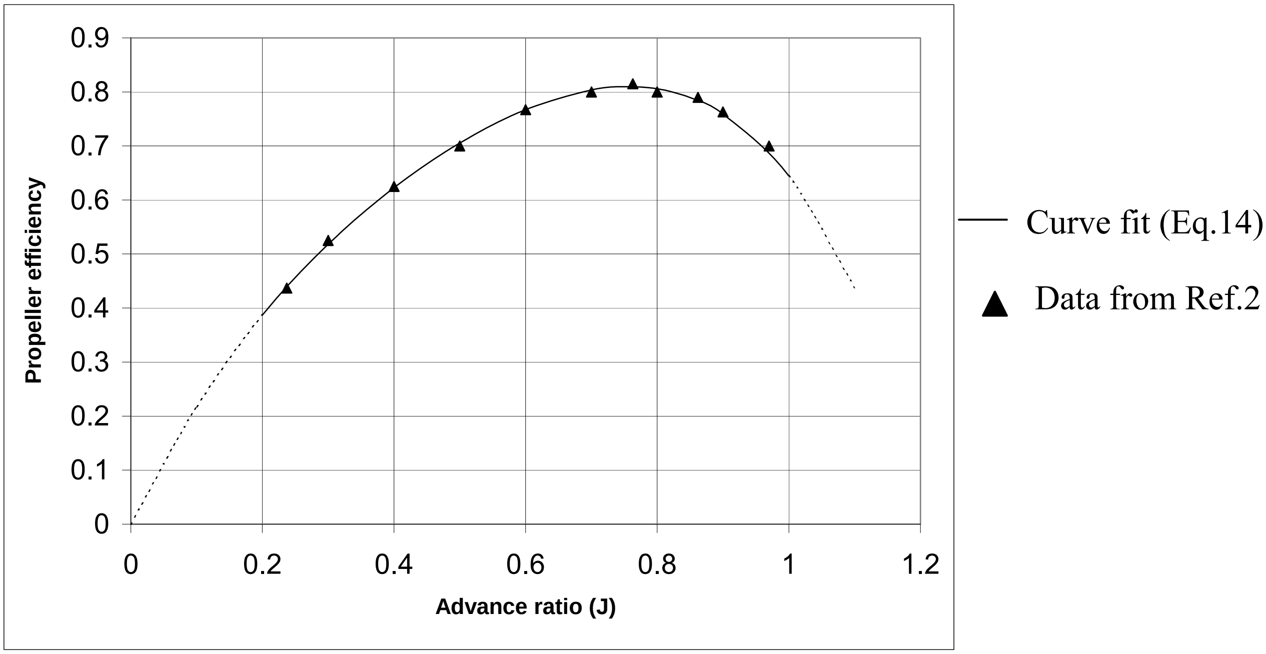

Fig.5 Variation of propeller efficiency with advance ratio

For the calculation of maximum speed, maximum rate of climb and maximum rate of turn, it is convenient to have maximum power available $(THP_a = \eta_p \times BHP)$ as a function of velocity. The maximum power occurs at 2700 rpm (45 rps). Noting the propeller diameter as 1.88m, the η_P vs J curve can be converted to η_p vs V curve (Fig.6).

The expression for η_P in terms of velocity is as follows.

$$\eta_p = -4.0447\times10^{-8}V^4 + 6.3445\times10^{-6}V^3 - 5.1398\times10^{-4}V^2 + 3.0244\times10^{-2}V - 0.0051668 \quad (15)$$

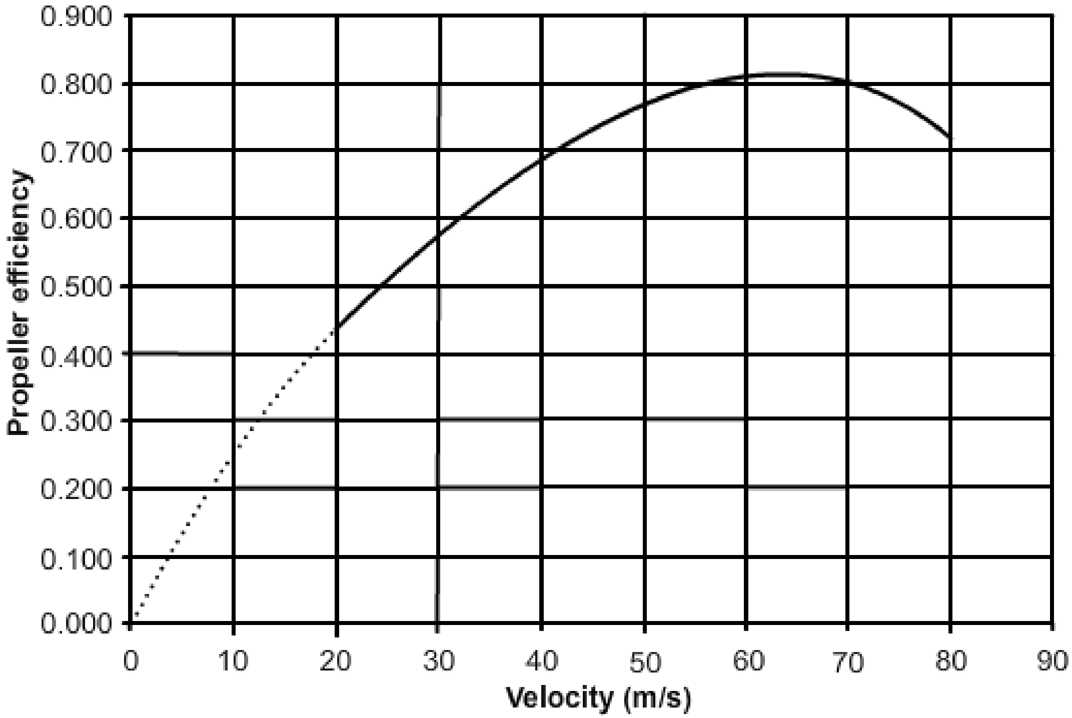

Fig.6 Variation of propeller efficiency with velocity at 2700rpm

Making use of the power available at different altitudes as given in Table 2 and the values of the propeller efficiency at different speeds given by Eq.(15), the maximum available thrust horsepower ($THP_a = \eta_p \times BHP$) can be obtained at different speeds and altitudes. The variations are plotted in Fig.7.

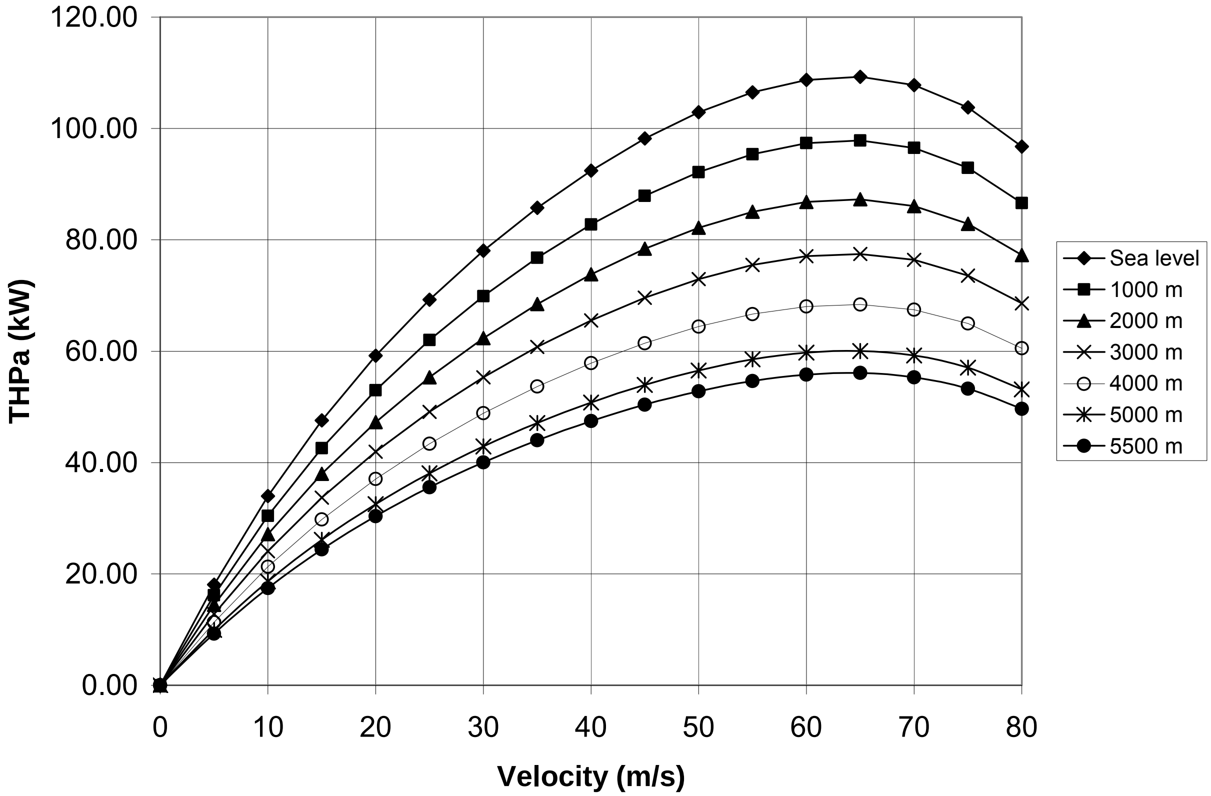

Fig.7 Variations of THP_a with altitude

4 Steady level flight

4.1 Variation of stalling speed with altitude

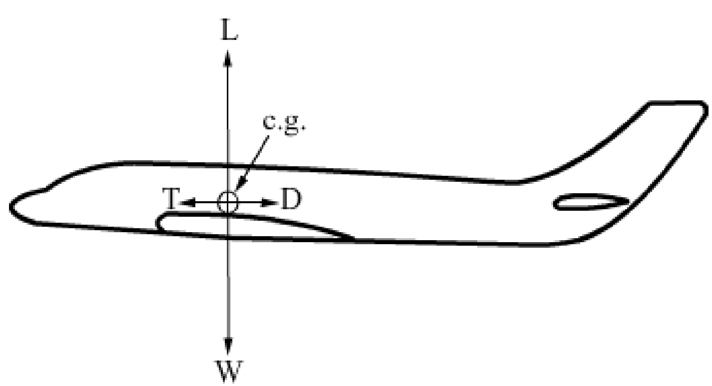

Fig.8 Forces on an airplane in steady level flight

In steady level flight, the equations of motion are:

$$T - D = 0 \quad (16)$$

$$L - W = 0 \quad (17)$$

Further,

$$L=\frac{1}{2}\rho V^2SC_L=W \quad (18)$$

$$T=D=\frac{1}{2}\rho V^2SC_D \quad (19)$$

$$V=\sqrt{\frac{2W}{\rho SC_L}}$$

Since C_L cannot exceed C_{Lmax} , there is a flight speed below which the level flight is not possible. The flight speed at which C_L equals C_{Lmax} is called the stalling speed and is denoted by V_s.

Hence,

$$V_s=\sqrt{\frac{2W}{\rho SC_{Lmax}}}$$

Since density decreases with altitude, the stalling speed increases with height.

In the present case, W = 1088 × 9.81 = 10673.28 N and S = 14.864 m^2.

As regards C_{Lmax} , Reference 2 gives the values of C_{Lmax} as 1.33, 1.42, 1.70 and 1.86 for flap deflections of 0^o ,10^o , 25^o and 40^o respectively.

Using these data, the variations of stalling speeds with altitude are presented in Table 3 and plotted in Fig.9.

H(m)	**σ**	**V_s (δ_f = 0^o) (m/s)**	**V_s (δ_f =10^o) (m/s)**	**V_s (δ_f =25^o) (m/s)**	**V_s (δ_f = 40^o) (m/s)**
0	1.000	29.69	28.73	26.26	25.10
1000	0.908	31.16	30.16	27.57	26.35
2000	0.822	32.75	31.70	28.97	27.69
3000	0.742	34.46	33.35	30.48	29.14
4000	0.669	36.30	35.13	32.11	30.70
4500	0.634	37.28	36.08	32.97	31.52
5000	0.601	38.29	37.06	33.87	32.38
5500	0.569	39.36	38.09	34.81	33.28
6000	0.538	40.46	39.16	35.79	34.22

Table 3 Stalling speeds for various flap settings

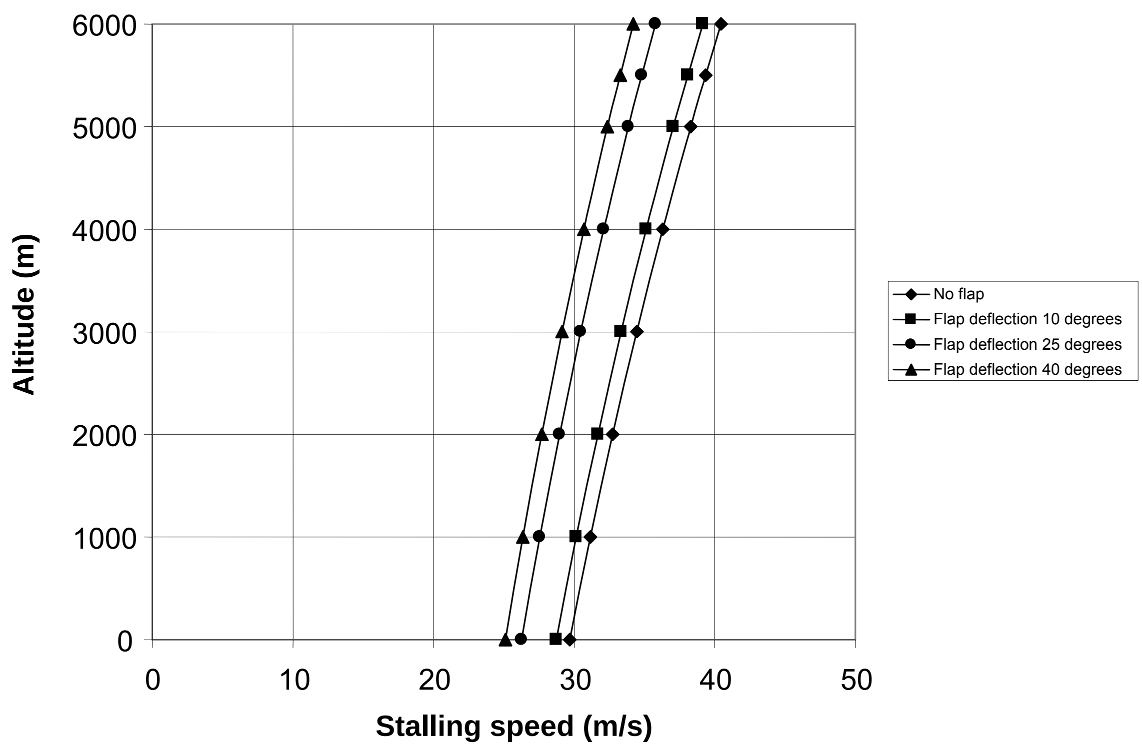

Fig.9 Variations of stalling speed with altitude for different flap settings

4.2 Variations of V_{max} and V_{min} with altitude

With a parabolic drag polar and the engine output given by an analytical expression, the following procedure gives V_{max} and V_{min}. Available power is denoted by P_a and power required to overcome drag is denoted by P_r. At maximum speed in steady level flight, required power equals available power.

$$P_a = BHP \times \eta_p \tag{20}$$

$$P_r = \frac{D \times V}{1000} = \frac{1}{2000} \rho V^3 S C_D$$

The drag polar expresses C_D in terms of C_L. Writing C_L as $\frac{2W}{\rho S V^2}$ and substituting in the above equations we get:

$$BHP \times \eta_p = \frac{1}{2000} \rho V^3 S C_{DO} + \frac{K W^2}{500 \rho S V}. \tag{21}$$

The propeller efficiency has already been expressed as a fourth order polynomial function of velocity and at a chosen altitude, BHP is constant with velocity. Their product ($\eta_p \times$ BHP) gives an analytical expression for power available. Substituting this expression on the left hand side of Eq.(21) and solving gives V_{max} and $(V_{min})_e$ at at the chosen altitude. Repeating the procedure at different altitudes, we get V_{max} and $(V_{min})_e$ at various heights. Sample calculations and the plot for sea level conditions are presented in Table 4 and Fig.10.

V (m/s)	**η_p**	**P_a (kW)**	**P_r(kW)**
0	0.000	0.000	-
5	0.134	18.086	188.983
10	0.252	33.995	94.789
15	0.352	47.549	64.053
20	0.438	59.185	49.778
25	0.513	69.259	42.753
30	0.578	78.045	40.069
35	0.635	85.735	40.615
40	0.685	92.438	43.953
45	0.727	98.184	49.947
50	0.762	102.918	58.611
55	0.789	106.503	70.040
60	0.805	108.724	84.376
65	0.809	109.280	101.792
70	0.798	107.790	122.480

Table 4 Steady level flight calculations at sea level

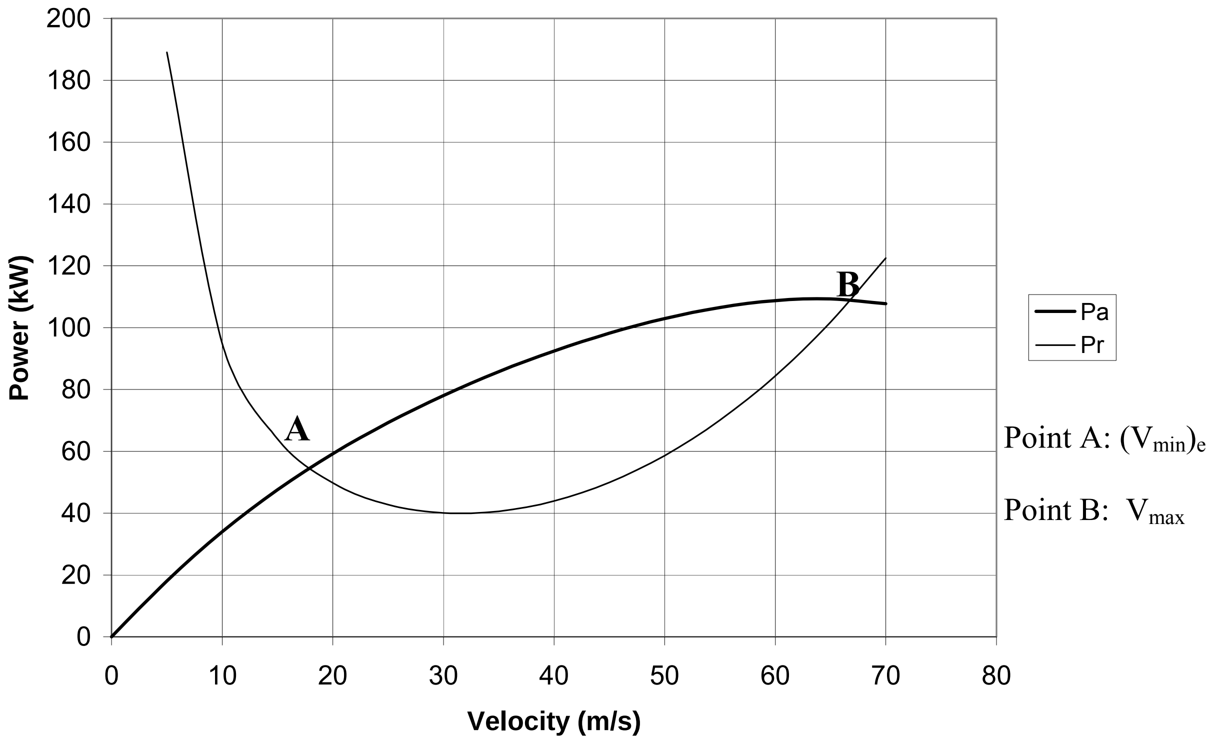

Fig.10 Sample plot for P_a and P_r at sea level

It may be noted that

- The minimum speed so obtained corresponds to that limited by power $(V_{min})_e$.
- If this minimum speed is less than the stalling speed, a level flight is not possible at this speed. The minimum velocity is thus higher of the stalling speed and $(V_{min})_e$.

The results for V_S, $(V_{min})_e$, V_{min} and (V_{max}) at various altitudes are tabulated in Table 5 and plotted in Fig.11. It may be noted that at h = 5200 m, V_{max} and $(V_{min})_e$ are same. This altitude is the maximum height attainable by the airplane and will be referred later as absolute ceiling.

h (m)	V_s(no flap) (m/s)	$(V_{min})_e$ (m/s)	V_{min} (m/s)	V_{max} (m/s)	V_{max} (kmph)
0	29.7	18	29.7	66.84	240.624
1000	31.2	20.4	31.2	65.75	236.7
2000	32.75	23.3	32.75	64.3	231.48
3000	34.46	27	34.46	62.3	224.28
4000	36.3	32	36.3	59.15	212.94
5000	38.29	41	41	52.7	189.72
5200	38.73	46.5	46.5	46.5	167.4

Table 5 V_S , $(V_{min})_e$, V_{min} and V_{max} at various altitudes

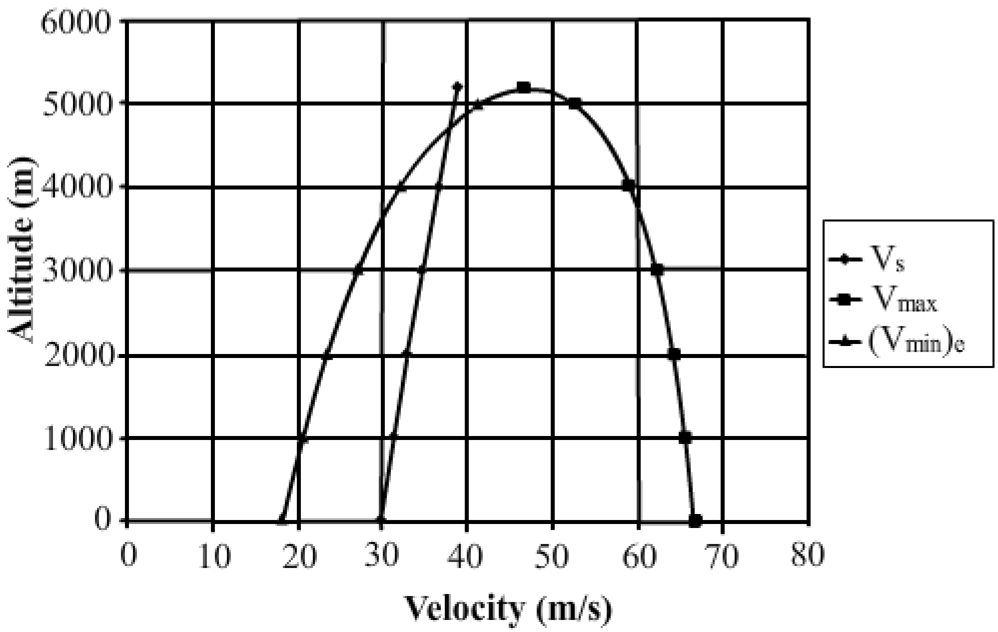

Fig.11 Variations of maximum and minimum flight velocities with altitude

Remark:

The calculated value of V_{max} of 240.6 kmph at sea level is fairly close to the value of 246 kmph of the actual airplane quoted in section 1.10.

5 Steady climb performance

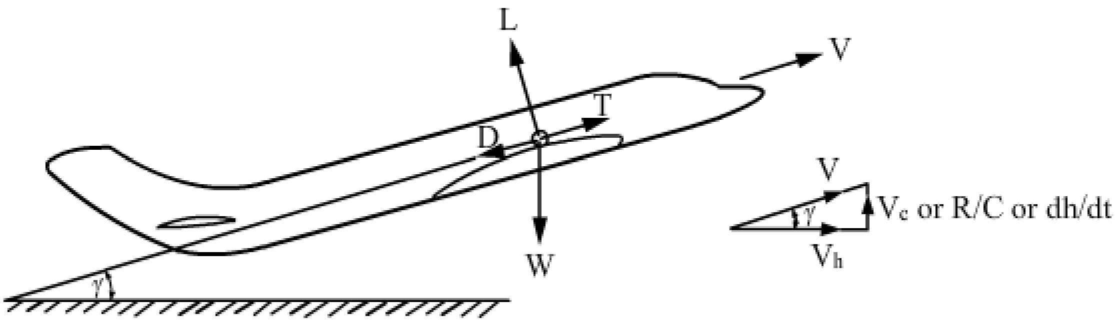

Fig.12 Forces on an airplane in steady climb

Calculation of rate of climb:

In this flight, the C.G of the airplane moves along a straight line inclined to the horizontal at an angle γ . The velocity of flight is assumed to be constant during the climb.

Since the flight is steady, acceleration is zero and the equations of motion can be written as:

$$T - D - W \sin\gamma = 0 \tag{22}$$

$$L - W \cos\gamma = 0 \tag{23}$$

Noting that $C_L = 2L/\rho V^2 S = \dfrac{2\,W\cos\gamma}{\rho S V^2}$, gives:

$$C_D = C_{Do} + K\left(\frac{2\,W\cos\gamma}{\rho S V^2}\right)^2$$

Also $V_c = V\sin\gamma$, or $\sin\gamma = \dfrac{V_c}{V}$

$$\cos\gamma = \sqrt{1 - \frac{V_c^{\,2}}{V^2}}$$

Substituting in Eq.(22) gives :

$$T_a = \frac{1}{2}\rho V^2 S\left\{C_{DO} + \frac{KW^2}{\frac{1}{2}\rho V^2 S}\left[1 - \left(\frac{V_c}{V}\right)^2\right]\right\} + W\frac{V_c}{V}$$

Or $A\left(\frac{V_c}{V}\right)^2 + B\left(\frac{V_c}{V}\right) + C = 0$

(24)

where,

$$A = \frac{KW^2}{\frac{1}{2}\rho V^2 S}, \; B = -W \text{ and } C = T_a - \frac{1}{2}\rho V^2 S C_{Do} - A \;;$$

T_a = available thrust = 1000 x P_a/V.

The available thrust horsepower is given by the following expression:

$P_a = BHP_{sealevel}(1.13\sigma - 0.13)\eta_p$

Equation 24 gives 2 values of $\frac{V_c}{V}$. The value which is less than 1.0 is chosen as appropriate.

Consequently,

$$\gamma = \sin^{-1}\frac{V_c}{V} \qquad (25)$$

$$V_c = V\sin\gamma \qquad (26)$$

The climb performance is calculated using following steps.

(i) Choose an altitude.

(ii) Choose a velocity between V_{min} and V_{max} and obtain A, B and C in Eq.(24).

(iii) Solve for $\frac{V_c}{V}$, obtain γ and V_c .

(iv) Repeat calculations, at chosen altitude, at various velocities in the range of V_{min} to $V_{max.}$

(v) Repeat steps (i) to (iv) at various altitudes.

Sample calculations at sea level are presented in Table 6.

V (m/s)	η_p	THP_a (kW)	T (N)	A	C	V_c/V	γ (deg.)	V_c (m/s)	V_c (m/min)
30	0.578	78.04	2601.49	1049.68	1265.85	0.120	6.894	3.60	216.03
35	0.635	85.73	2449.56	771.19	1289.14	0.122	7.000	4.26	255.89
40	0.685	92.43	2310.96	590.44	1212.13	0.114	6.563	4.57	274.29
45	0.727	98.18	2181.86	466.52	1071.92	0.101	5.790	4.53	272.36
50	0.762	102.91	2058.35	377.88	886.123	0.083	4.777	4.16	249.80
55	0.789	106.50	1936.42	312.30	662.971	0.062	3.568	3.42	205.35
60	0.805	108.72	1812.06	262.42	405.797	0.038	2.181	2.28	137.00
65	0.809	109.28	1681.23	223.60	115.193	0.011	0.619	0.70	42.10

Note: B = - W = -10673.28 N

Table 6 Steady climb calculations at sea level.

Repeating similar calculations at various altitudes gives the variations of γ and V_c with velocity at different altitudes. The results are plotted in Figs.13 and 14. From these figures the variations of γ_{max}, V_{cmax} or $(R/C)_{max}$, $V_{\gamma max}$ and $V_{R/Cmax}$ at various altitudes are obtained. The results are presented in Table 7 and in Figs.15, 16 and 17.

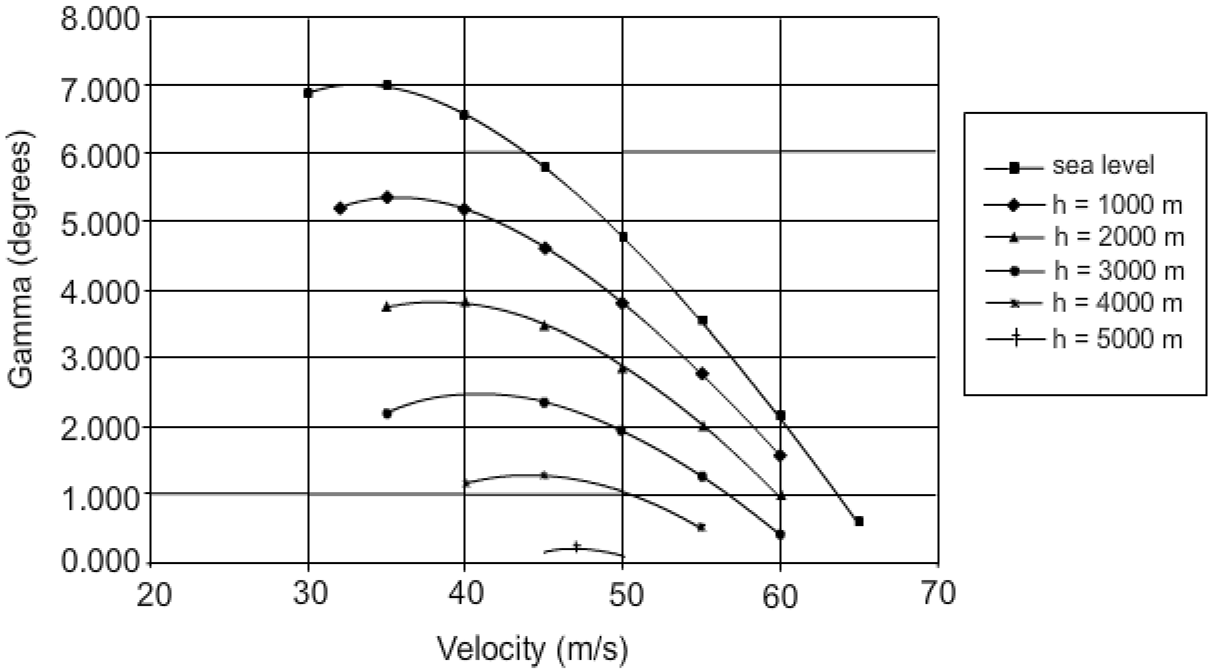

Fig.13 Variations of angle of climb with flight velocity at different altitudes

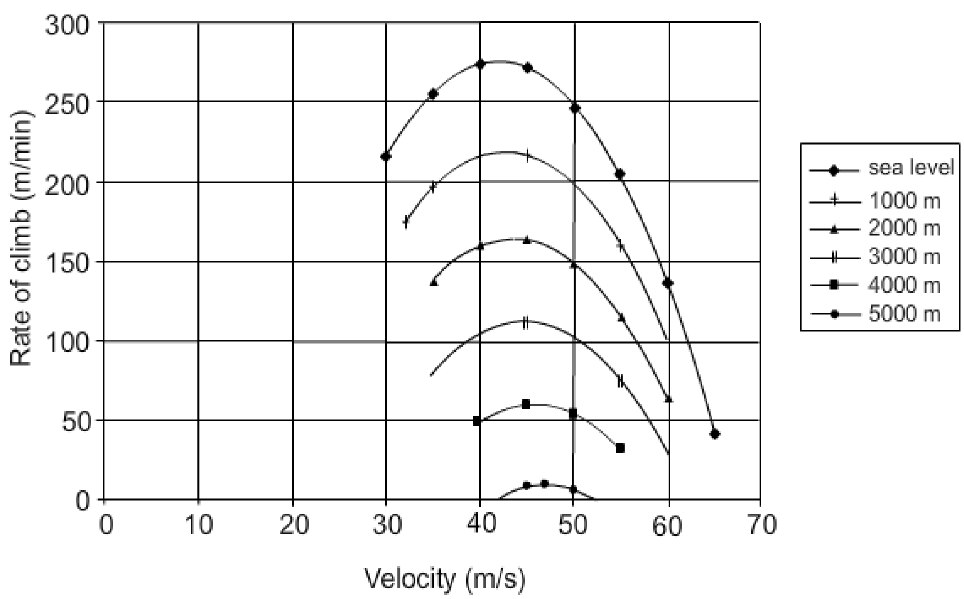

Fig.14 Variations of rate of climb with flight velocity at different altitudes

h (m)	γ_{max} (deg)	V_{cmax} (m/min)	$V_{\gamma max}$ (m/s)	$V_{R/C\,max}$ (m/s)
0	7	276	34.1	41.7
1000	5.4	219.7	35	42.6
2000	3.83	165.8	38	43.6
3000	2.5	111.7	40.9	45
4000	1.28	60.5	44	45.9
5000	0.2	10	46	46.5
5200	0	0	46.5	46.5

Table 7 Climb performance

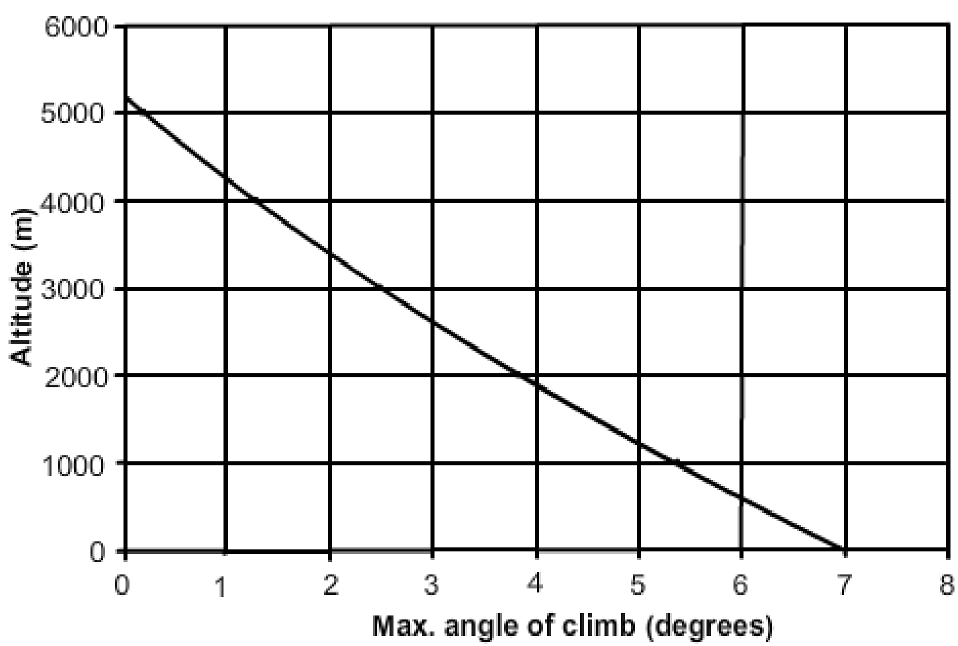

Fig.15 Variation of maximum angle of climb with altitude

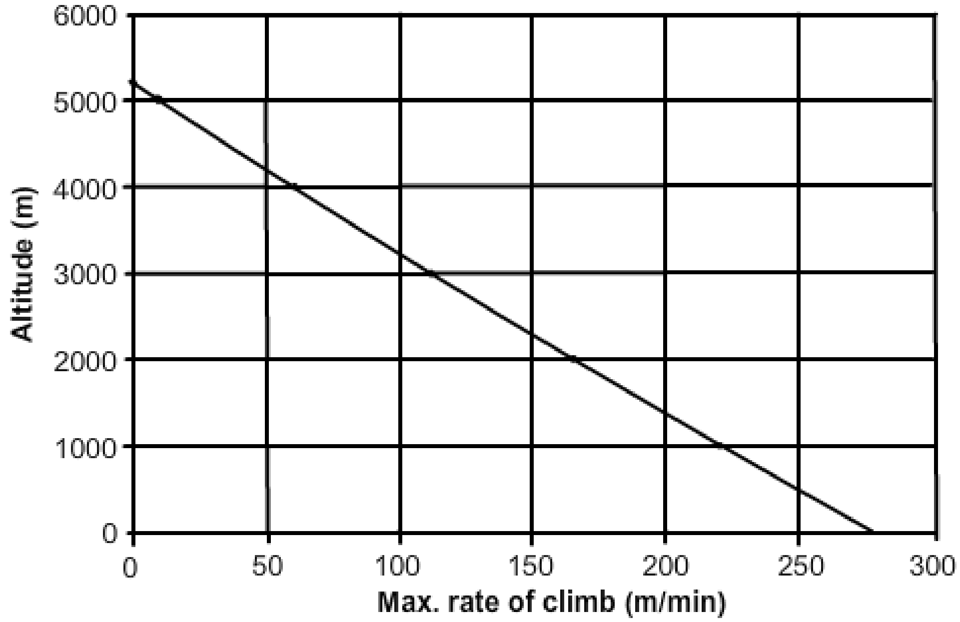

Fig.16 Variation of maximum rate of climb with altitude

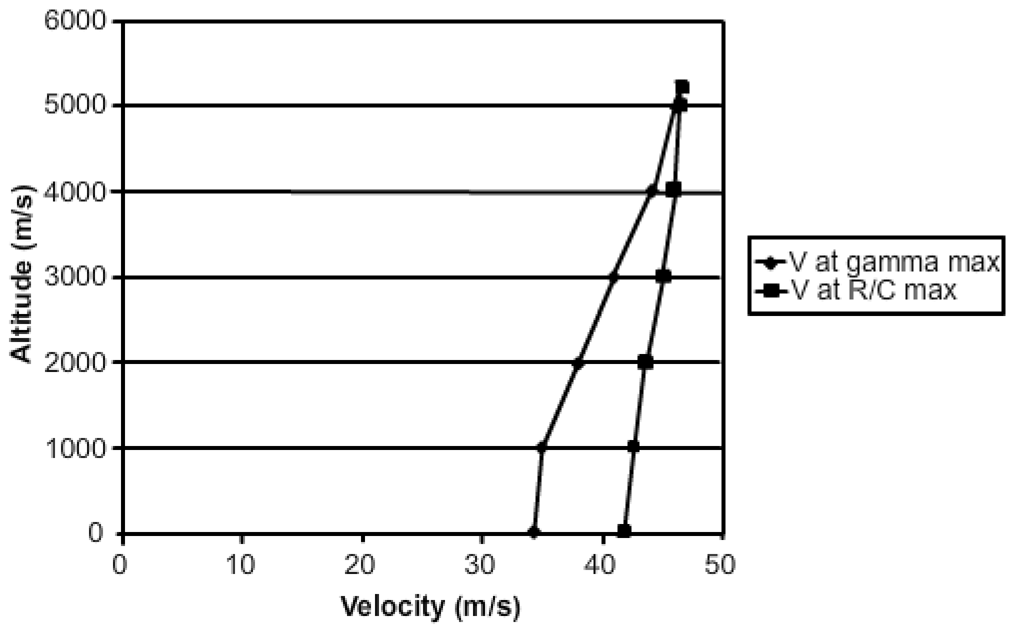

Fig.17 Variations of $V_{\gamma max}$ and $V_{(R/C)max}$ with altitude

Remark:

It is observed that the maximum rate of climb and maximum angle of climb decrease with altitude, but the velocity at which the rate of climb and angle of climb are maximum increase slightly with height.

Service ceiling and absolute ceiling

The altitude at which the maximum rate of climb becomes 100 ft/min (30.5 m/min) is called the service ceiling and the altitude at which the maximum rate of climb becomes zero is called the absolute ceiling of the airplane. These can be obtained from Fig.16. It is observed that the absolute ceiling is 5200 m and the service ceiling is 4610 m. It may be pointed out that the absolute ceiling obtained from R/C_{max} consideration and that from V_{max} consideration are same (as they should be). Further, the service ceiling of 4610 m is close to the value of 4035 m for the actual airplane quoted in section 1.10.

6 Range and endurance

6.1 Estimation of range in a constant velocity flight

It is convenient for the pilot to cruise at constant velocity. Hence, the range performance in constant velocity flights is considered here. In such a flight at a given altitude, the range (R) of a piston-engined airplane is given by the following expression (Eq.7.23 of the main text of the course).

$$R = \frac{7200\,\eta_p}{BSFC} E_{max} \tan^{-1}\left[\frac{E_1\,\zeta}{2E_{max}\,(1-KC_{L1}\,E_1\zeta)}\right] = \frac{3600\eta_p}{BSFC\sqrt{k_1 k_2}}[\tan^{-1}\frac{W_1}{\sqrt{k_1/k_2}} - \tan^{-1}\frac{W_2}{\sqrt{k_1/k_2}}] \quad (27)$$

where, $k_1 = \frac{1}{2}\rho V^2 S C_{Do}, k_2 = \frac{2K}{\rho S V^2}$, W_1 and W_2 are the weights of the airplane at the start and end of the cruise, $E_{max} = \frac{1}{2\sqrt{C_{DO}K}}$, $E_1 = \frac{C_{L1}}{C_{D1}}$, $C_{L1} = \frac{2W_1}{\rho S V^2}$, $\zeta = 1 - \frac{W_2}{W_1}$,

$C_{D1} = C_D$ corresponding to C_{L1}.

From this expression the range and endurance in constant velocity flights, can be obtained at different flight speeds, at the cruising altitude. From this information, the flight speeds which would give the maximum range and endurance can be arrived at. It may be pointed that the above expression gives the gross still air range as defined in section 7.2.3 of the main text of the course.

The following values are taken as the common data for the subsequent calculations.

Cruising altitude = 8000′ (2478 m)

W_1 = Weight at the start of range flight = maximum gross weight = 10673.28 N

Usable fuel = 178.63 litre = 1331.78 N of petrol

W_2 = Weight at the end of the flight = 10673.28 – 1331.78 = 9341.5 N

Wing area (S) = 14.864 m^2, C_{DO} = 0.0349, K = 0.0755

ρ = density at $8000'$ = 0.9629 kg/m^3

The power required during a constant velocity flight varies as the fuel is consumed. However, for the purpose of present calculations the power required is taken as the average of power required at the start and end of cruise. It is denoted as THP_{avg}. It is noted that the power required (THP_{avg}) can be delivered by the engine operating at different settings of RPM (N) and manifold air pressure (MAP). But, for each of these settings the propeller efficiency (η_p) and fuel flow rate would be different. The optimum setting, which would give the maximum range, can be arrived at by using the following steps.

(a) Select a value of N and calculate J (= V/nd); n = N/60 .

(b) Obtain η_p corresponding to this value of J from Eq.(14).

(c) Then, BHP required (BHP_r) = THP_{avg} / η_p

(d) The left hand side of Fig 4.2 of the main text, shows the BHP vs MAP and fuel flow rate vs MAP curves with rpm as parameter. Similar curves are generated for $h = 8000'$.

(e) From the curves in step (d) the sets of N and MAP values which would give desired BHP_r can be obtained.

(f) Obtain fuel flow rate for each set of MAP and N. Calculate BSFC. Subsequently Eq.(27) gives the range for chosen set of N and MAP.

(g) Repeat calculations at different value of N.

(h) The combination of N and MAP which gives longest range is the optimum setting.

The aforesaid steps are carried-out in the next three subsections.

6.2 Calculation of BHP and fuel flow rate at different RPM's and MAP's at $8000'$

Example 4.2 of the main text illustrates the procedure to obtain BHP and fuel flow rate at N = 2200 and MAP of $20''$ of Hg. Similar calculations are repeated at N = 2700, 2600, 2400, 2200 and 2000 and at MAP = 15, 16, 17, 18, 19, 20, 21 and $21.6''$ of Hg. It may be pointed out that the atmosphere pressure at $8000'$ is $21.6''$ of Hg. (see also right hand side of the engine characteristics shown in Fig.3 of this Appendix). The values so obtained are plotted and smoothed. Figure 18 shows the calculated values by symbols and the smoothed variations by curves.

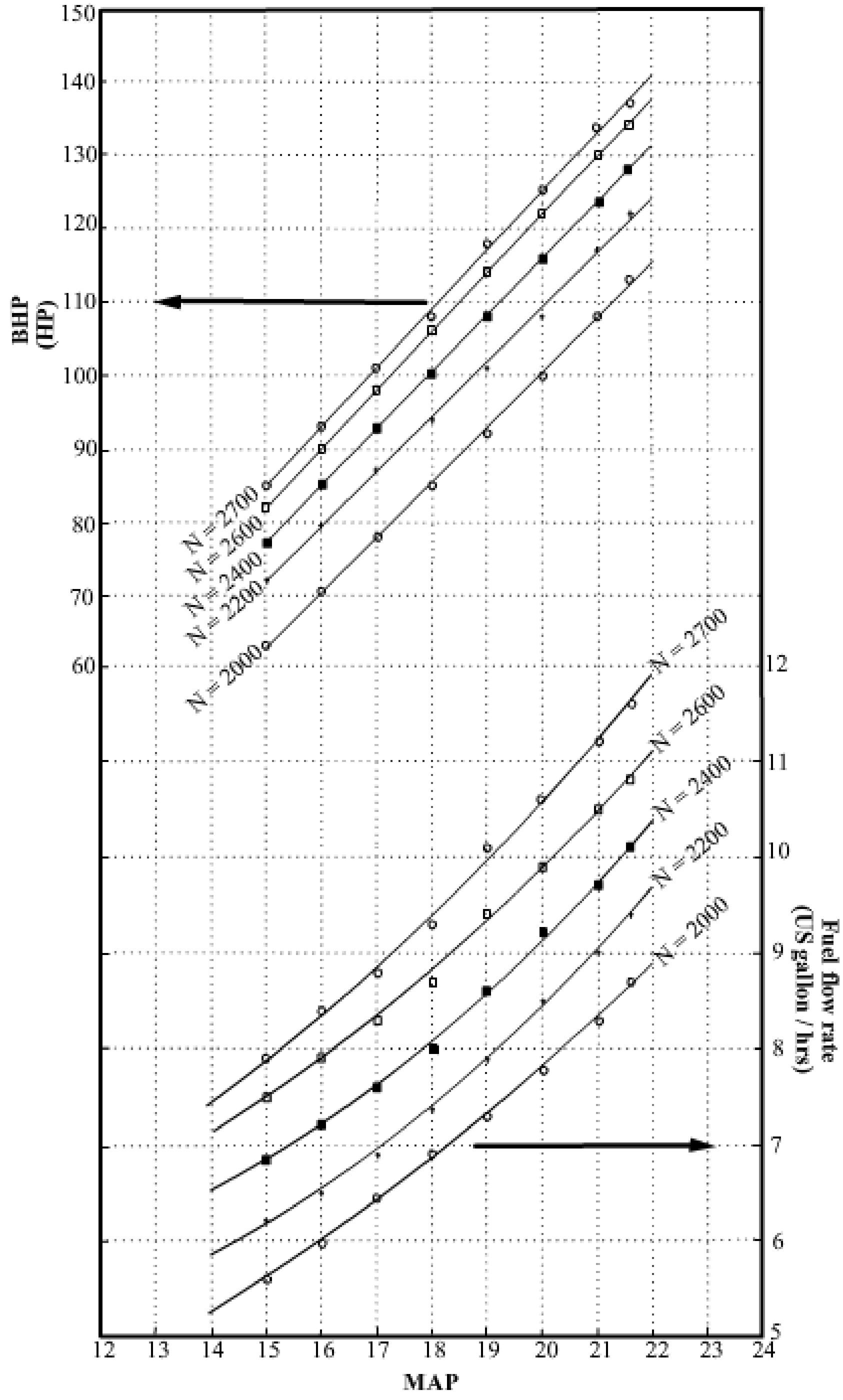

Fig.18 Variations of BHP and fuel flow rate with MAP

6.3 Sample calculations for obtaining optimum N and MAP for a chosen flight velocity (V)

For the purpose of illustration V is chosen as 50 m/s or 180 kmph

I) Calculation of THP_{avg} :

C_{L1} = Lift coefficient at start of range $= \dfrac{W_1}{\frac{1}{2}\rho V^2 S} = \dfrac{10673.28}{0.5\times0.9629\times50^2\times14.864} = 0.5966$

C_{L2} = Lift coefficient at the end of cruise $= \dfrac{W_1}{\frac{1}{2}\rho V^2 S} = \dfrac{9341.5}{0.5\times0.9629\times50^2\times14.864} = 0.5220$

$C_{D1} = C_D$ corresponding to $C_{L1} = 0.0349 + 0.0755 \times 0.5966^2 = 0.06177$

$C_{D2} = C_D$ corresponding to $C_{L2} = 0.0349 + 0.0755 \times 0.5220^2 = 0.05547$

$C_{Davg} = (0.06177 + 0.05547)/2 = 0.05862$

$THP_{avg} = \frac{1}{2}\rho V^2 S C_{Davg} / 1000 = 0.5 \text{ x } 0.9629 \text{ x } 50^3 \text{ x } 14.864 \text{ x } 0.05862/1000 = 52.43$ kW

= 70.31 HP

II) Steps to obtain highest η_p/BSFC or the range

(a) Choose 'N' from 2700 to 2000

(b) Calculate, $J = V/nd$; $n = N/60$; $d = 1.88$ m

(c) Corresponding to the value of J in step (b), obtain η_p from Eq.(14)

(d) Obtain $BHP_r = THP_{avg}/\eta_p$; THP_{avg} in HP

(e) From upper part of Fig.18, obtain MAP which would give BHP_r at chosen N. For these values of N and MAP obtain the fuel flow rate (FFR) in gallons/hr , from the lower part of Fig.18.

(f) Convert FFR in gallons per hour to that in N/hr and BHPr in HP to kW.

Obtain BSFC $= \dfrac{\text{FFR in N/hr}}{\text{BHP in kW}}$

(g) Obtain η_p/BSFC and also range from Eq.(27).

The above calculations at different values of N are presented in the table below.

N (RPM)	J	η_p	BHP (HP)	MAP	FFR (gal/hr)	FFR (N/hr)	BHP (kW)	BSFC (N/kW-hr)	η_p/BSFC	Range (km)
2700	0.591	0.762	92.22	15.90	8.32	234.58	68.77	3.410	0.223	1023.8
2600	0.613	0.773	90.88	16.10	7.92	223.44	67.77	3.297	0.234	1074.8
2400	0.664	0.794	88.54	16.47	7.38	208.07	66.02	3.151	0.251	1154.2
2200	**0.725**	**0.807**	**87.03**	**17.04**	**6.95**	**196.06**	**64.90**	**3.020**	**0.267**	**1224.9**
2000	0.797	0.806	87.22	18.25	6.97	196.56	65.04	3.021	0.266	1221.8

Table 8 Sample calculation at V = 180 kmph

It is observed from the above table that at the chosen value of V =180 kmph, the range is maximum for the combination of N = 2200 and MAP of 17.04″ of Hg. The value of R is 1224.9 km.

III Obtaining range and endurance at different flight speeds

Repeating the calculations indicated in item (II), at different values of flight speeds in the range of speeds V_{stall} from V_{max} at $8000'$, yield the results presented in Table 8a. Since the flight speed is constant, the endurance (E) is given by the following expression.

$$E \text{ (in hours)} = \frac{\text{Range in km}}{V(\text{in kmph})}$$

V (m/s)	V (kmph)	THP_{avg} (kW)	RPM_{opt}	MAP	FFR (gal/hr)	FFR (N/hr)	η_p	BHP (kW)	BSFC (N/kW -hr)	Range (km)	Endur-nce(hr)
34	122.4	41.01	2000	16.61	6.254	176.26	0.734	55.859	3.155	929.6	7.59
36	129.6	41.13	2000	16.39	6.16	173.62	0.753	54.597	3.18	999.2	7.66
38	136.8	41.64	2000	16.29	6.12	172.52	0.77	54.069	3.19	1061.1	7.76
40	144	42.51	2000	16.32	6.13	172.79	0.784	54.198	3.19	1114.3	7.74
43	154.8	44.53	2000	16.58	6.24	175.81	0.8	55.643	3.16	1176.5	7.6
46	165.6	47.37	2000	17.1	6.46	182.07	0.808	58.573	3.11	1214.5	7.33
50	180	52.43	2200	17.04	6.95	196.06	0.807	64.904	3.02	1225	6.81
52	187.2	55.53	2200	17.7	7.25	204.4	0.81	68.576	2.98	1222.1	6.52
54	194.4	58.98	2200	18.49	7.63	215.12	0.808	72.97	2.95	1205.5	6.2
56	201.6	62.81	2200	19.43	8.13	228.99	0.803	78.23	2.93	1174.1	5.82
58	208.8	67.03	2200	20.56	8.77	247.02	0.793	84.5	2.92	1127.1	5.4
60	216	71.63	2400	20.42	9.37	264.14	0.806	88.87	2.97	1090.2	5.05

Table 8a Range and endurance in constant velocity flights at $8000'$ (2438 m)

The results are plotted in Figs.19 and 20.

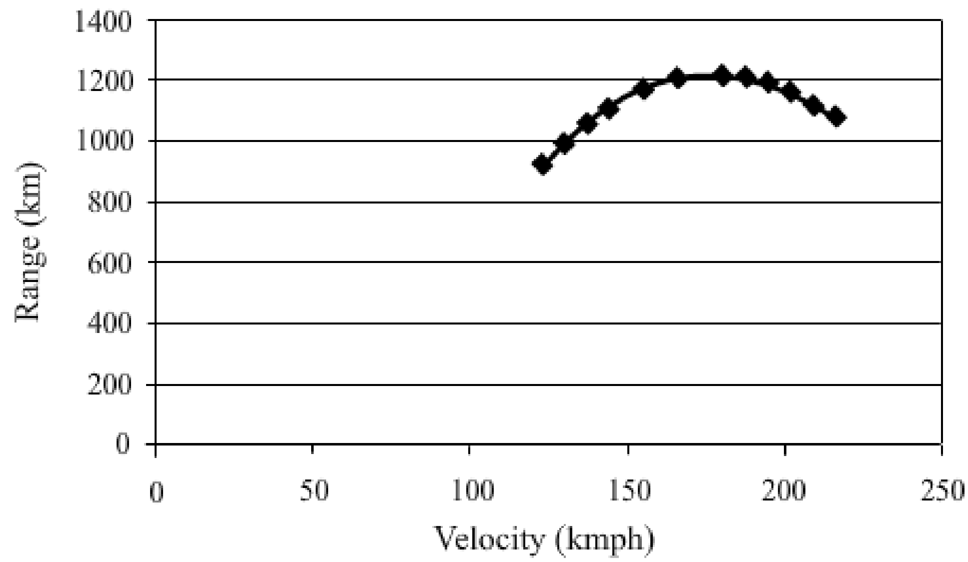

Fig.19 Range in constant velocity flights at h = 8000′ (2438 m)

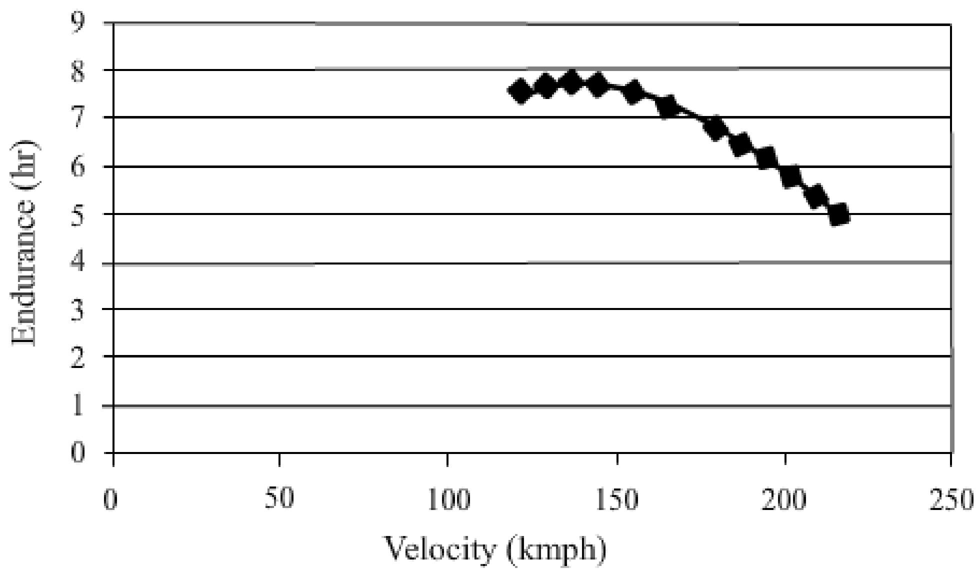

Fig.20 Endurance in constant velocity flights at h = 8000′ (2438 m)

Remarks:

i) It is seen that the maximum endurance of 7.7 hours occurs in the speed range of 125 to 145 kmph.

ii) The range calculated in the present computation is the Gross Still Air Range (GSAR). The maximum range is found to be around 1220 km which occurs in the speed range of 165 to 185 kmph.

iii) The range quoted in Section 1.10 for Cherokee PA – 28 - 181 accounts for taxi, take-off, climb, descent and reserves for 45 min. This range can be regarded as safe range. This value is generally two-thirds of the GSAR. Noting that two-thirds of GSAR (1220 km) is 813 km, it is seen that the calculated value is within the range of performance given in Section 1.10.