

# Flight Lab: Drag Polar Estimation using Cessna-206H

Group D

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## 1 Aim of the Experiment

To Calculate  $C_{D0}$  and  $k$  of the Cessna 206H using the Drag polar of the aircraft through Flight testing.

## 2 Introduction

The purpose of this experiment is to estimate drag polar relationship for Cessna 206H aircraft. The drag polar is the relationship between the lift on an aircraft and its drag, expressed in terms of dependence of the lift coefficient on the drag coefficient. Many performance parameters can be determined from the drag polar such as optimum rate of climb. The drag polar equation may be written as

$$C_D = C_{D0} + KC_L^2$$

where  $C_D$  is total coefficient.  $C_{D0}$  is the drag coefficient.  $C_L$  denotes coefficient of lift and  $K = \frac{1}{\pi AR e}$  where  $AR$  is aspect ratio of the wing and  $e$  is oswald efficiency factor.  $KC_L^2$  is the induced drag coefficient of a wing. and unavoidable companion of the Lift of Wing. The other drag are parasite drag and wave drag are included in the  $C_{D0}$ .

## 3 Cruise mode

In this experiment we will take the readings at cruise mode. Cruise is the level portion of the aircraft travel where flight is most fuel efficient. It occurs between ascent and descent phases and is usually the majority of a journey, Technically. cruise consists of heading changes only at a constant airspeed and altitude. for most commercial passenger aircraft. the cruise phase consumes a lot of fuel consumes the majority of fuel, typical speed is 400-500 knots. Commercial aircraft is optimized for cruise phase.

### 3.1 Instruments for this Experiment

- Airspeed Indicator

- Engine RPM Indicator
- Manifold pressure gauge
- Outside air temperature
- Altimeter
- Stopwatch

### 3.2 Calculation

Brake horse power Equation

$$\begin{aligned}
 BHP &= D \cdot V \\
 D &= \frac{\rho V^2 S C_D}{2} \\
 C_D &= C_{D0} + K C_L^2 \\
 BHP &= \frac{\rho V^2 S}{2} (C_{D0} + K C_L^2)
 \end{aligned}$$

Now using  $L = \frac{\rho V^2 S C_L}{2}$  and also knowing  $L = W$  during cruise, we get:

$$BHP = \frac{\rho S C_{D0}}{2} + \frac{2KW^2}{\rho S}$$

The Density correction is applied with the international standard ISA Table  
**Break Horse Power using Data**

$$BHP = \frac{(rpm)(MP)(Ts)(rHP)}{(rrpm)(rMP)(OAT)}$$

### 3.3 Procedure

- Wait Until altitude is reached.
- Record airspeed indicator, Engine manifold pressure, outside air temperature, RPM of engine during the cruise.
- Obtain Break Horse Power using the Recorded Data.
- Repeat for all Altitudes.
- Power required for the Steady level flight is given by

$$P_{req} = \frac{1}{2} \rho V^3 S C_{D0} + \frac{\frac{W^2}{\frac{1}{2} \rho V S}}{\pi A R e \rho S}$$

- The Above equation is modified as

$$PV = \left(\frac{1}{2}\rho SC_{D0}\right)V^4 + \frac{2W^4}{\pi A Re \rho S}$$

- The above equation is a straight line  $Y = mX + c$  with

$$Y = PV$$

$$m = \frac{1}{2}\rho SC_{D0}$$

$$e = \frac{2}{\pi A Re \rho S}$$

We can calculate the slope and intercept from data We can calculate the required parameters using above equations

## 4 Climb Mode

Following Take off Airplane Has to climb to maintain safe and economic Flight. Climb may be achieved by increasing angle of attack on the wing, or by increasing the thrust of the engine to increasing speed in some cases both techniques are used.

As lift decreases with density a climb once initiated, ends when lift is equal to weight at that point steady flight.

### 4.1 Calculations

Consider Equilibrium of Forces.

$$T - D - W \sin \gamma = 0$$

$$L - W \cos \gamma = 0$$

Now By rearranging terms we get that

$$\frac{VT - VD}{W} = V \sin \gamma$$

$$RC = \frac{dh}{dt} = V \sin \gamma$$

This can be evaluated at various speeds and the values of  $v$  corresponding to maximum rate of climb can be obtained.

## 4.2 Rate of Climb

if the time to traverse the altitude band is also recorded then we can actually calculate true rate of climb of the aircraft.

Let observed temperature is  $T_0$  and the standard temperature at the altitude is  $T_s$ , Then Pressure recorded since the pressure change is always the true pressure difference for altitude change shown by the altimeter.

$$\delta p = -\rho_s g(\delta H)_p$$

s:Standard altitude

p:Pressure altitude

$\delta H_T$  is true change in altitude, will have same temperature difference.

$$\begin{aligned}\delta p &= -\rho T g(\delta H)_T \\ \frac{\delta H_T}{\delta H_p} &= \frac{\rho_s}{\rho_T} = \frac{T_0}{T_s}\end{aligned}$$

Therefore ,

$$RC_{true} = RC_{observed} * \frac{T_0}{T_s}$$

## 4.3 Procedure

- Record the Take off Weight( $W_T$ ).
- Note the initial and final altitudes (  $h_1$   $h_2$ ) and time instants ( $t_1$   $t_2$ ).
- Repeat for different climb speeds.
- Record the weight after the landing.and consider average weight for calculations.
- Find out the rate of climb for each velocity and corresponding angle of climb  $\Gamma$  using following equations.

$$RC \approx \frac{h_2 - h_1}{t_2 - t_1} \cdot RC_{true} = RC_{observed} \times \frac{T_0}{T_s}$$

- Plot rate of climb vs speed.
- Plot angle of climb vs speed
- From the plots find Maximum rate of climb and maximum angle of climb and the corresponding velocities.

## 5 Observations Record

**Cessna 206H parameters**

<b>Parameter</b>	<b>Value</b>
Rated RPM	2700
Manifold pressure	29.92 in Hg
Sea Level Temperature	288.15K
Wing Area (S)	16.16 m <sup>2</sup>
Wing Span (b)	10.9728m
Rated Hp (rHP)	223.709 kW

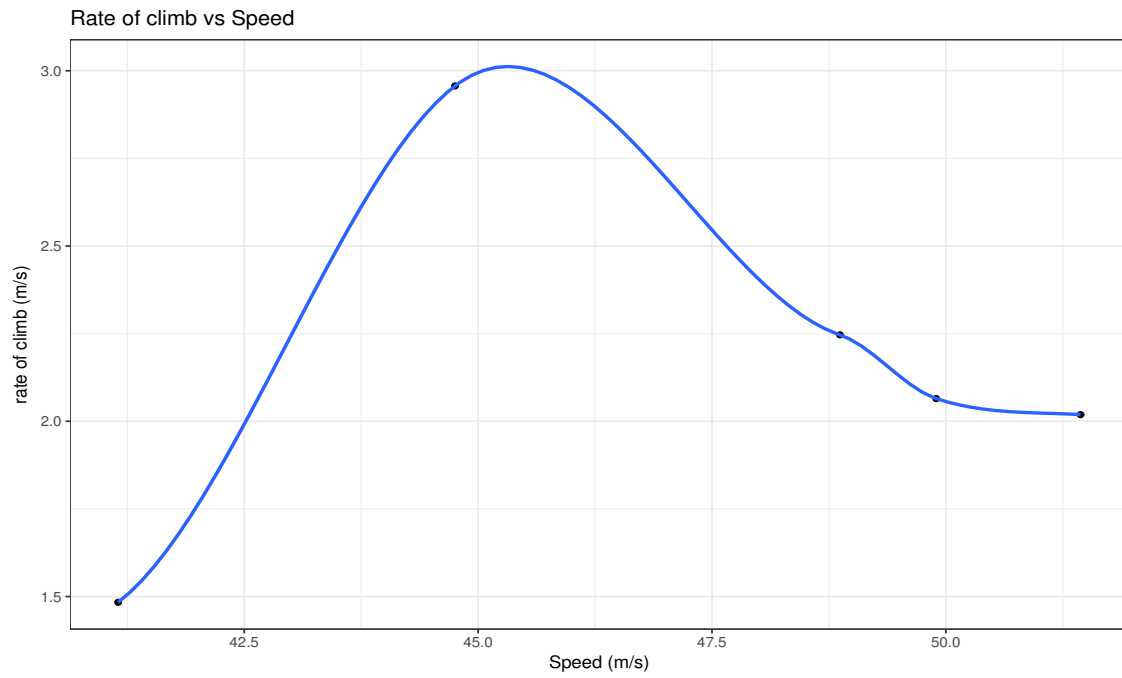
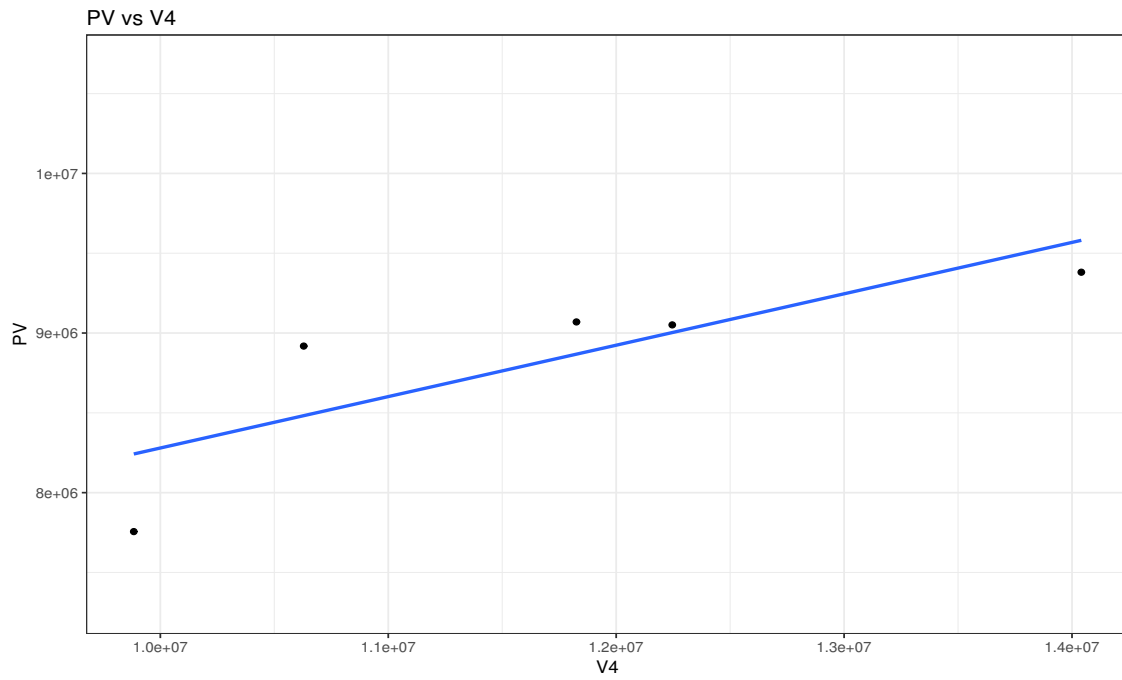
<b>V</b>	<b>MP</b>	<b>RPM</b>	<b>OAT</b>	<b>Altitude</b>
93	19.5	2270	32	500
99	21.1	2260	30	1000
95	20.9	2310	28	1600
93	22.9	2300	28	2020
90	22	2440	26	2400
94	21.2	2380	24	3040

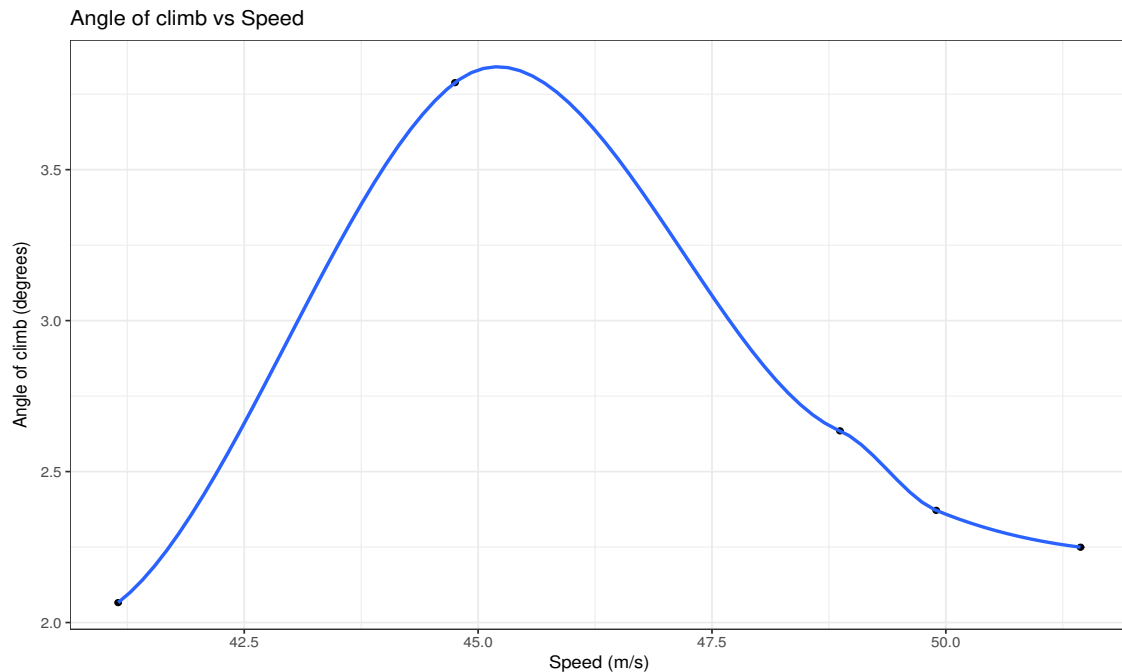
Table 1: Cruise Experiment Data

<b>V</b>	<b>RPM</b>	<b>MP</b>	<b>OAT</b>	<b>h1</b>	<b>h2</b>	<b>Time</b>
97	2530	25.2	31	500	1000	77.91
95	2370	25.3	30	1000	1500	71.37
80	2550	24.6	28	1500	2000	107.36
87	2550	24.4	27	2000	2500	53.69
100	2540	24	25	2500	3000	78.1

Table 2: Climb Experiment Data

## 6 Plot





## 7 Calculation

- The value of Aspect ratio is calculated as 7.456
- The line Equation of the Cruise equation for the graph is given by  $Y = .03220X + 5.059e + 06$
- From the Formulas above we can calculate the  $C_{D0} = 0.0326$  and  $e = 0.21565$
- the value of  $k$  is equal to 0.20

## 8 Discussions

- The Drag Polar is the relationship between the lift on an aircraft and its drag, expressed in terms of the dependence of the lift coefficient on the drag coefficient.
- Drag Polar helps to determine many performance parameter such as the power required at different altitudes and speed, the optimum speed and altitude for cruise. and optimum rate of climb.
- We can verify the rate of climb obtained from the altitude and time measurement with help of temperature readings at the corresponding altitude. Since we know the

standard atmosphere lapse rate, we can calculate the altitude from the difference between temperature readings. Since we know the time, we can get the rate of climb from the above mentioned. And hence verify the the rate of climb. Both the observed and the rate of temperature changes should have a similar profile.

- Oswald's efficiency factor is a generalized parameter connected with an aircraft's aerodynamic efficiency . Specifically, for a parabolic drag polar, there exists a dependency. The Oswald efficiency factor  $e$  reflects the airplane lifting properties deterioration caused by the distortion of an elliptical lift distribution and accounts for the non-ellipticity of the lift distribution, the increase of profile drag of the wing, fuselage, tail plane, nacelles and various interference effects with angle of attack
- There is a discrepancy between expected and actual value obtained because oswald efficiency factor depends upon the area of fuselage and the wing leading edge cross-sectional shape which can enforce the suction at the leading edge zone and decrease drag due to lift.
- All the measurements are taken from FPS system is a system of units. Here all our calculations are done in SI units the appropriate conversions are applied before calculation procedure.
- The script used for the Generation of plots can be found here.
- The average weight of aircraft is used in the formulas which is average of take off and landing weight.

## 9 Conclusions

- The coefficients  $C_{D0}$  and  $e$  are calculated.
- The Plots for the Angle of Climb and rate of climb with the variation of velocity are plotted.
- The maximum rate of climb is found around 45.2m/s.
- the maximum angle of climb is also found around 45m/s.