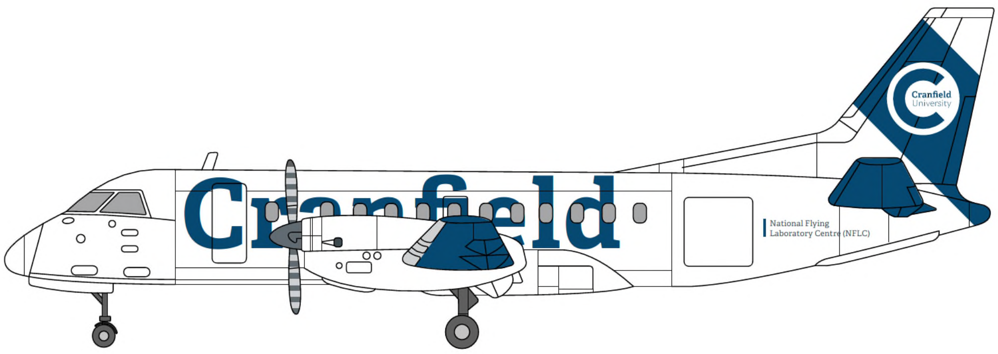
Flight Experimental Methods

Course Handbook



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With acknowledgements to Dr Alastair Cooke

Contents

1 Introduction

1.1 Background

This short course builds on the theoretical teaching given to students studying aeronautical engineering. It should demonstrate how the performance and handling qualities of a real aircraft can be measured, and how the data can be processed to a form that enables the evaluation of some of its characteristics.

Figure 1.1 Saab 340B+ G-NFLB

The aircraft is a Saab 340B+ turbo-prop, see Figure 1.1, with a maximum take-off weight of 13155 kg, powered by two GE CT7-9B engines (maximum continuous rating 1750 SHP). It has been converted into a laboratory for in-flight experimental work by the installation of an instrumentation system and seat-back displays. Your role is that of a flight test observer ('task specialist') taking measurements and analysing the resulting data. The aim of each test is to estimate some aerodynamic property of the aircraft, using standard flight test techniques, which is characteristic of either its stability, controllability or performance, or to establish the accuracy of the some of the on-board navigation systems.

Before the flight(s) there will be a review of the theory supporting the test during which it will be extended to cover the flight measurement technique(s). This is followed by a mandatory 'briefing' in which the test procedures will be explained and the data required will be described. A member of the Cranfield University academic staff ('the demonstrator') will fly in the aircraft and supervise the test as the data is gathered.

1.2 Aircraft Interior

The cabin layout in G-NFLB is configured to 33 seats, although the normal maximum complement of 24 Task Specialists will be on board, see Figure 1.2. In front of each seat is a MS Surface Pro that will display the flight parameters for each experiment.

Figure 1.2: Aircraft Interior

The on-board instrumentation system consists of signal conditioning units, power supplies, an Inertial Management Unit (IMU) and a data acquisition system. Sensors and transducers are located around the aircraft to measure:

- Flying control surface deflections (aileron, elevator, elevator trim and rudder).
- Applied control forces (elevator and rudder).
- Angle of attack and angle of sideslip.
- Aircraft attitudes, body angular rates and translational accelerations from the Ekinox D Inertial Management Unit.
- Navigation sensors: GPS, IRS, VOR/DME and ILS.

Transducer outputs are converted in the signal conditioning unit and data acquisition system into physical units for display and/or for recording, see the example shown in Figure 1.3. The combination of data and screen format shown on the seat-back displays is selected by the demonstrator. The observers panel consists of a computer generated display, which will normally be programmed to represent analogue as well as digital instruments.

The exact form of the display will be described during each briefing session(s). In most tests the measurements are made in steady flight conditions and recorded manually from the display. You will be told when to take the readings. Normally your screen will be 'frozen' so that it remains unchanged for a reasonable period of time allowing you to record all the values required. After the flight the data must be processed - the calculations required are described in the relevant analysis section and may need data found in Appendix A or B.

Figure 1.3: Observer's Panel

1.3 General Analysis

It is customary during flight tests to quote the following items which are relevant at the time of test: aircraft mass [weight], centre of gravity (CG) position [balance], pressure altitude (Hp), outside air temperature (OAT) and equivalent airspeed (Ve or EAS).

Calculation of aircraft mass (and CG position) is carried out using standardised loading forms, as included in this handbook. Starting from the basic mass of the unfuelled aircraft the masses and moments of the flight crew and special equipment are added, this gives the operational empty mass (OEM) and moment. From readings of the fuel quantity (displayed in kg) taken during test, the mass and moment of the fuel can be included in the loading table along with the mass and moments of the observers and the demonstrator. The mass and CG position of the aircraft can now be calculated for the test condition. Note that the CG is expressed as a percentage of mean aerodynamic chord (MAC or $\bar{\bar{c}}$).

Air data (airspeed, altitude, and temperature) is normally based on measurements from the pitot-static system and an air temperature sensor installed near the nose of the aircraft. The air data computer (ADC) applies all the necessary corrections to convert these measurements into their most convenient form for display. The computer-generated display will show Equivalent Airspeed (EAS or Ve), Pressure Altitude (Hp) based on an altimeter reference pressure setting of 1013 hPa. Air temperature is displayed as the static air temperature in Celsius, and labelled ‘OAT’.

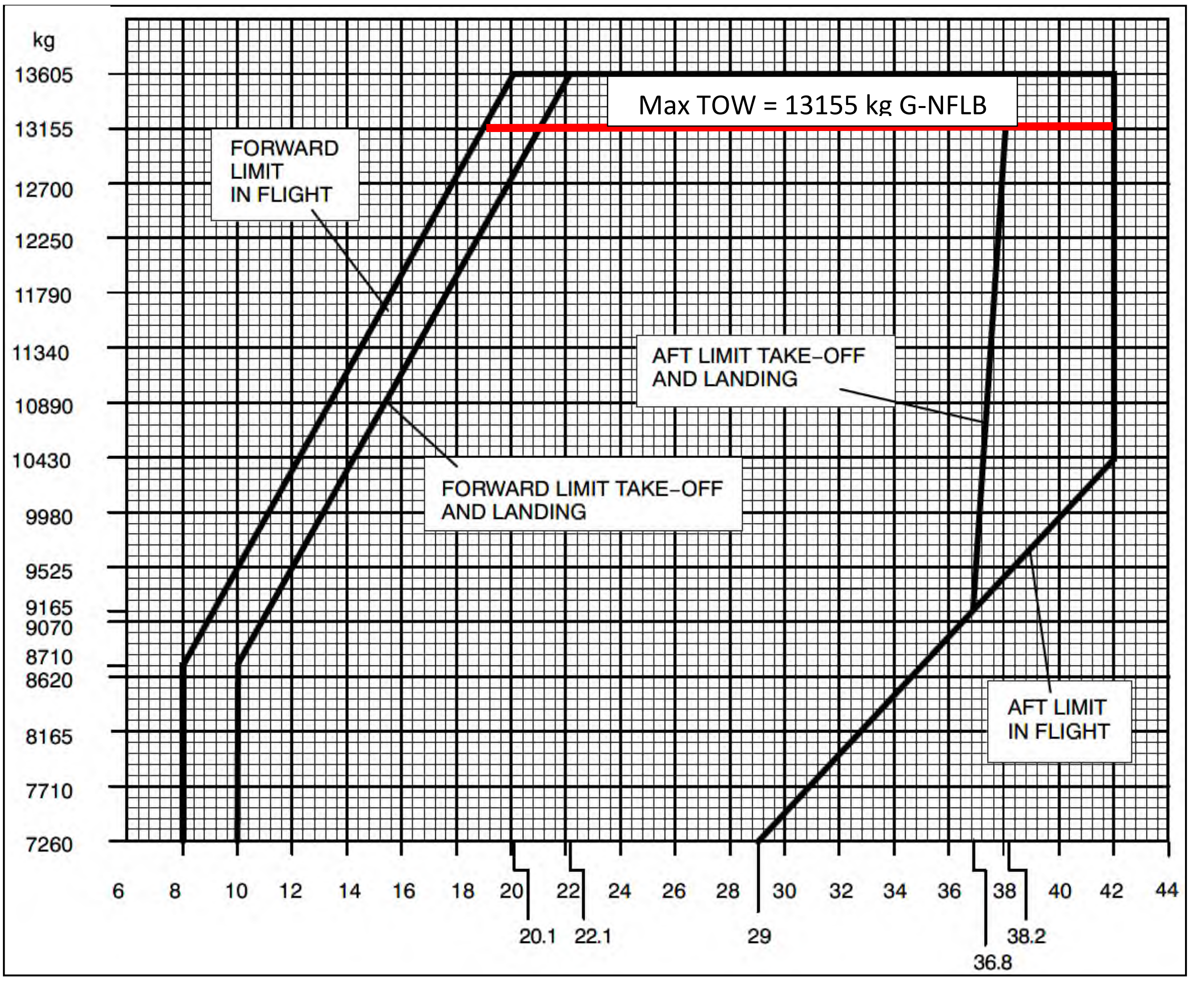


Figure 1.4 - Centre of Gravity Limits (percent MAC)

Loading Calculation Tables for Saab 340B G-NLFB

	Mass [m] (kg)
Aircraft including flight crew (Basic Empty Weight)	**8695**
Passengers including demonstrator plus *Ballast box (200kg)*	
Average fuel mass	
Σ m (Total mass)	

	Mass m (kg)	**Distance from datum (x) in m**	**Moment m.x (kg.m)**
Aircraft including crew (Basic empty weight)	**8695**	**10.69**	**92950**
Fuel		**11.18**	
Demonstrator position	**80**	**6.91**	**553**
Seat row 1		**7.87**	
Seat row 2		**8.61**	
Seat row 3		**9.34**	
Seat row 4		**10.08**	
Seat row 5		**10.82**	
Seat row 6		**11.73**	
Seat row 7		**12.49**	
Seat row 8		**13.23**	
Seat row 9		**13.96**	
Seat row 10		**14.7**	
Seat row 11		**15.49**	
Ballast box	**200**	**17.12**	
Σ m (Total mass)		**Σ m.x**	

CG position as a percentage of MAC is given by

CG position = $\frac{CG-LEMAC}{MAC} = \left[\frac{\sum m.x}{\sum m} - 10.472\right] x \frac{100}{2.085} = \qquad \% \, \bar{\bar{c}}$

2 Lift and Drag Characteristics

This section begins with an examination of typical aircraft lift and drag characteristics. These may be assessed in a number of different aircraft configurations, at different aircraft weights and altitude. Knowledge of the combined aircraft thrust allows an estimation of the total drag to be made.

Cruise performance is governed by the rate at which fuel is consumed. The fuel is required by the engines to produced thrust in order to counter the aerodynamic drag generated by the airframe.

2.1 Subsonic Lift and Drag Characteristics

Lift and drag forces acting on an aircraft are expressed in non-dimensional terms as C_L and C_D using the following equations. In general they are both functions of angle of attack of the aircraft, as shown in Figure 2.1

$$C_L = \frac{2L}{\rho_\infty V^2 S} = \frac{2L}{\rho_0 V_e^2 S} \approx \frac{2W}{\rho_0 V_e^2 S} \quad \text{and} \quad C_D = \frac{2D}{\rho_\infty V^2 S} = \frac{2D}{\rho_0 V_e^2 S} \approx \frac{2T}{\rho_0 V_e^2 S}$$

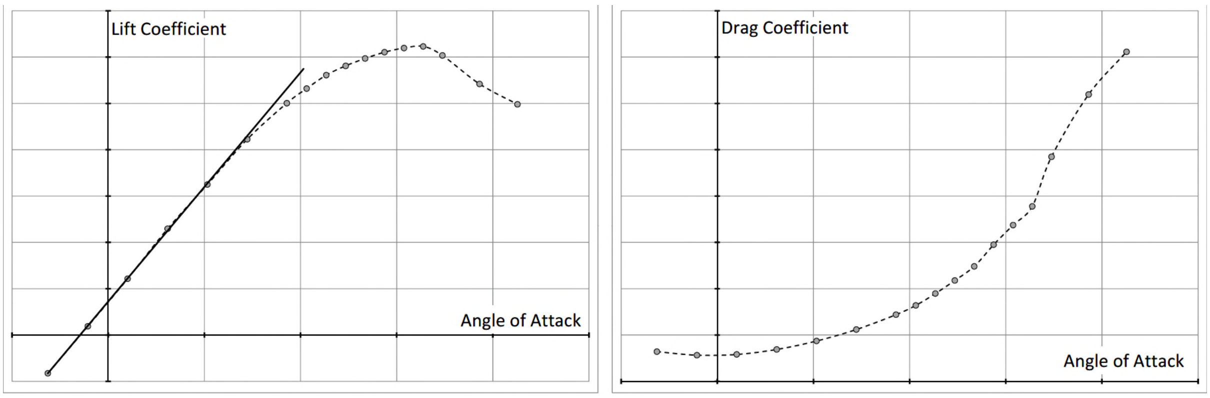


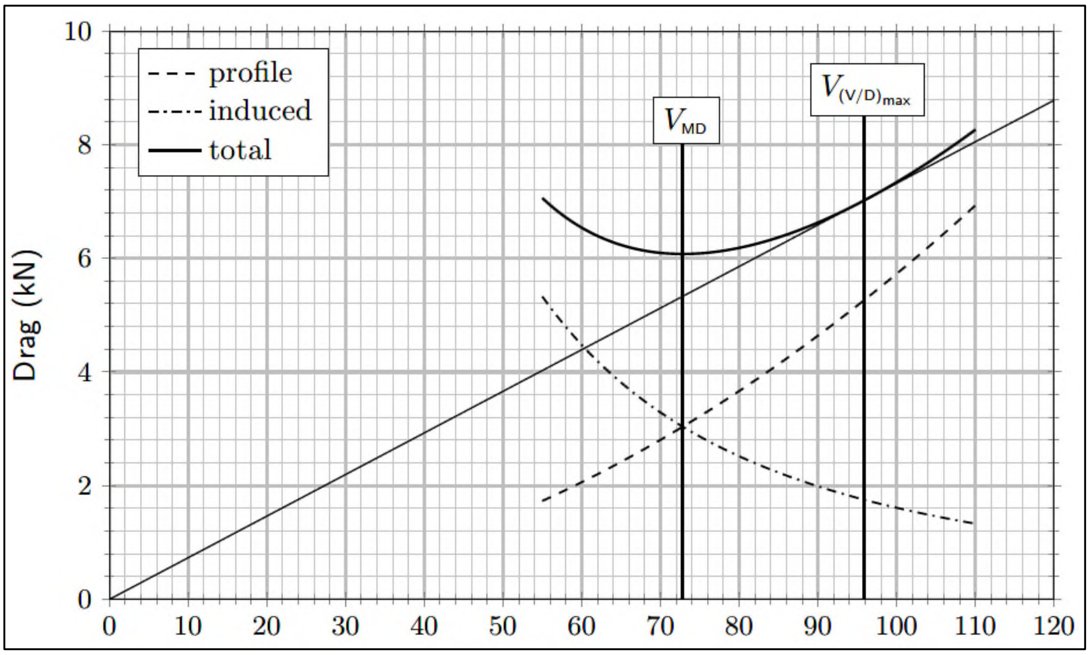
Figure 2.1: Typical variations of C_L and C_D with angle of attack

Figure 2.1 shows the **lift characteristic** as it would be plotted from wind tunnel results. The reduction of C_L at high angles of attack indicates that the wing has reached its stalling angle: a difficult area to evaluate accurately in flight. It is usual for aircraft to operate in the linear part of the curve (except for aerobatics or maybe training manoeuvres), and in this regime the lift may be expressed as follows. A plot of C_L against angle of attack will determine both a_0 and a_1.

$$C_L = C_{L_\alpha}(\alpha - \alpha_0) = C_{L_\alpha}.\alpha - C_{L_0} = a_1\alpha + a_0 \qquad (2.1)$$

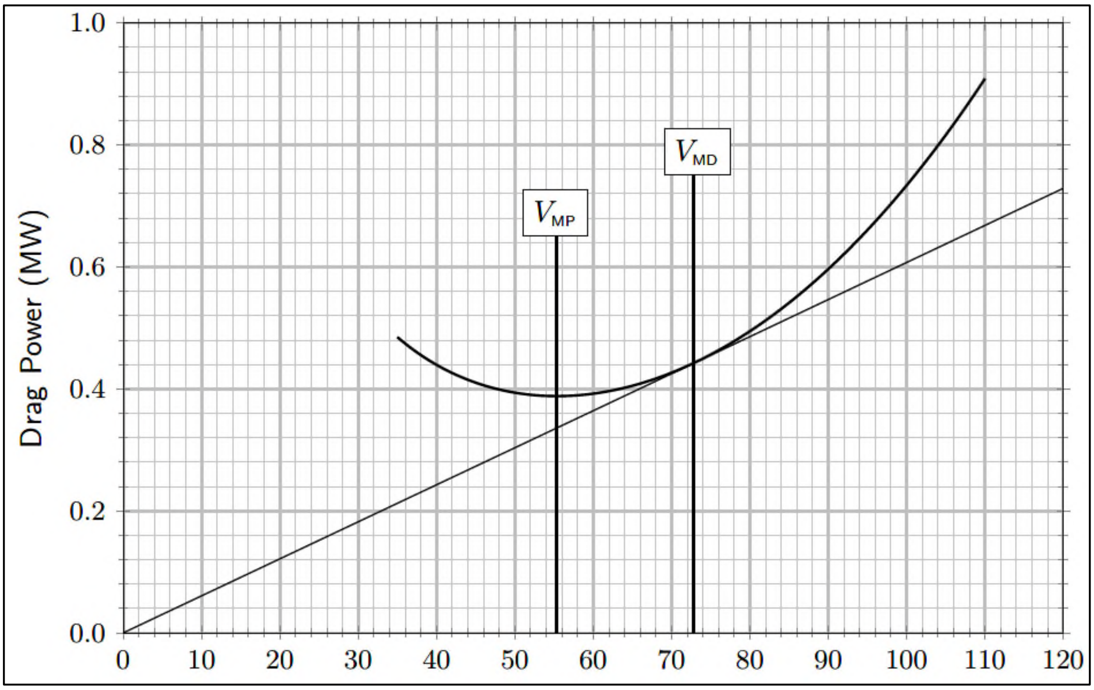
The **total drag** of a subsonic aircraft can be considered in terms of zero lift (profile) drag and lift dependent (induced) drag:

$$C_D = C_{D_0} + KC_L^2 \tag{2.2}$$



True Airspeed [V] (m/s)

Figure 2.2 – Drag Characteristics



True Airspeed [V] (m/s)

Figure 2.3 – Power Characteristics

The variation of total drag (D) and drag power (DV) with true airspeed (TAS) is shown in Figures 2.2 and 2.3 for an aircraft in level flight.

These graphs depict the most important factors in performance theory, and it is important to remember that they are valid for only one combination of weight, altitude and temperature. Three points on the curves are of considerable importance, these are the **minimum drag speed**, the **minimum power speed** and the speed that results in the **best true airspeed to drag ratio**.

2.2 Minimum Drag Speed

Since the figure is drawn for level flight[1], the lift must be constant. Therefore, the speed where the drag is a minimum (V_{MD}) defines the maximum value for L/D. It can be shown that at the minimum drag speed:

$$KC_L^2 = C_{D_0} \qquad \text{and} \qquad (C_L)_{\mathrm{MD}} = \sqrt{\frac{C_{D_0}}{K}} \tag{2.3}$$

Hence maximum L/D is given by:

$$\left(\frac{L}{D}\right)_{\mathrm{MAX}} = \frac{C_L}{C_D} = \frac{C_L}{C_{D_0} + KC_L^2} = \frac{1}{2C_{D_0}}\sqrt{\frac{C_{D_0}}{K}} = \frac{1}{2}\sqrt{\frac{1}{C_{D_0}K}} \tag{2.4}$$

2.3 Minimum Power

Power is the product of force and speed, and therefore the power required to maintain a constant speed in level flight is given by the product of drag and true airspeed. Above V_{MD} both drag and speed are increasing, so the minimum power required will be at a speed lower than the minimum drag speed. The speed at which it occurs is known as the minimum power speed (V_{MP}). As P = DV , then it is a minimum when:

$$C_L = \sqrt{\frac{3C_{D_0}}{K}} \tag{2.5}$$

[1] Fixed aircraft mass is assumed

2.4 Maximum Speed to Drag Ratio

If a line is drawn from the origin to any point on the drag plot, an angle is formed relative to the horizontal (speed) axis. The speed to drag ratio is a maximum when this angle is a minimum, and this occurs when the line is tangential to the drag curve. Taking the value of speed and drag at the point where the line touches the curve defines the best ratio of V/D. It can be shown that this occurs when:

$$C_L = \sqrt{\frac{C_{D_0}}{3K}} \tag{2.6}$$

2.5 Relationship between Speeds

Conversion of these expressions for C_L into speeds relative to V_{MD} gives:

$$V_{\mathsf{MP}} = V_{\mathsf{MD}}\sqrt[4]{\frac{1}{3}} = 0.76V_{\mathsf{MD}} \qquad \text{and} \qquad V_{\mathsf{V/D}_{\mathsf{max}}} = V_{\mathsf{MD}}\sqrt[4]{3} = 1.32V_{\mathsf{MD}}$$

Likewise

$$V_{e_{\mathsf{MP}}} = 0.76V_{e_{\mathsf{MD}}} \qquad \text{and} \qquad V_{e_{\mathsf{V/D}_{\mathsf{max}}}} = 1.32V_{e_{\mathsf{MD}}}$$

3 Aircraft Performance - Range and Endurance

3.1 Introduction

The values of the current endurance and the available range are functions of the instantaneous fuel flow. Consequently a measurement of the fuel flow required to achieve steady flight over the full speed range of the aircraft can be used to determine the airspeeds which maximize both these quantities.

If the requirement is to remain airborne for as long as possible, this is achieved by choosing a speed which results in the minimum fuel flow: the best endurance speed (V_{BE}). If, however, the requirement is to achieve the maximum range using the available fuel, then the ratio of air distance travelled per unit of fuel used must be maximized. This ratio is more precisely calculated, from flight test data, using the rate of change (with respect to time) of both quantities; the ratio of true airspeed (V) to fuel flow (rate), G. The ratio is called the “Specific Air Range” (SAR), as below. The maximum value defines the speed (V_{BR}) which will give the best range (in still air conditions).

$$SAR = \frac{V}{G} \tag{3.1}$$

In ideal conditions a plot of fuel flow rate (FFR) against true airspeed (TAS) can be used to determine both speeds: the minimum point indicating V_{BE}, and the point at which a tangent from the origin touches the curve indicates the minimum fuel flow to speed ratio and therefore V_{BR}.

The fuel flow of an aircraft engine is dependent on many variables, but it is possible to make some simple estimations based on these assumptions. These include the aircraft being at constant weight, altitude, and temperature. An additional assumption is that whatever the engine control does in the short term, the long-term effect will be to change the speed of the aircraft, assuming appropriate adjustments in elevator deflection are made, with an associated change in fuel flow.

The power output from a piston aero-engine is derived from burning a fuel/air mixture in the cylinders. The power is transmitted to a crankshaft and then to a propeller. The power produced by the engine is proportional to the fuel consumed, and is converted into thrust by the propeller. If the propeller operates at constant efficiency (a good assumption for a variable pitch propeller over a reasonable speed range), then fuel flow is proportional to the power absorbed by the propeller. The best true air speed to power ratio is found by drawing a line from the origin which is tangential to the power curve, see Figure 3.1. The point of contact with the curve defines the speed for the best ratio and is the same speed as the minimum drag speed (as power is speed x drag).

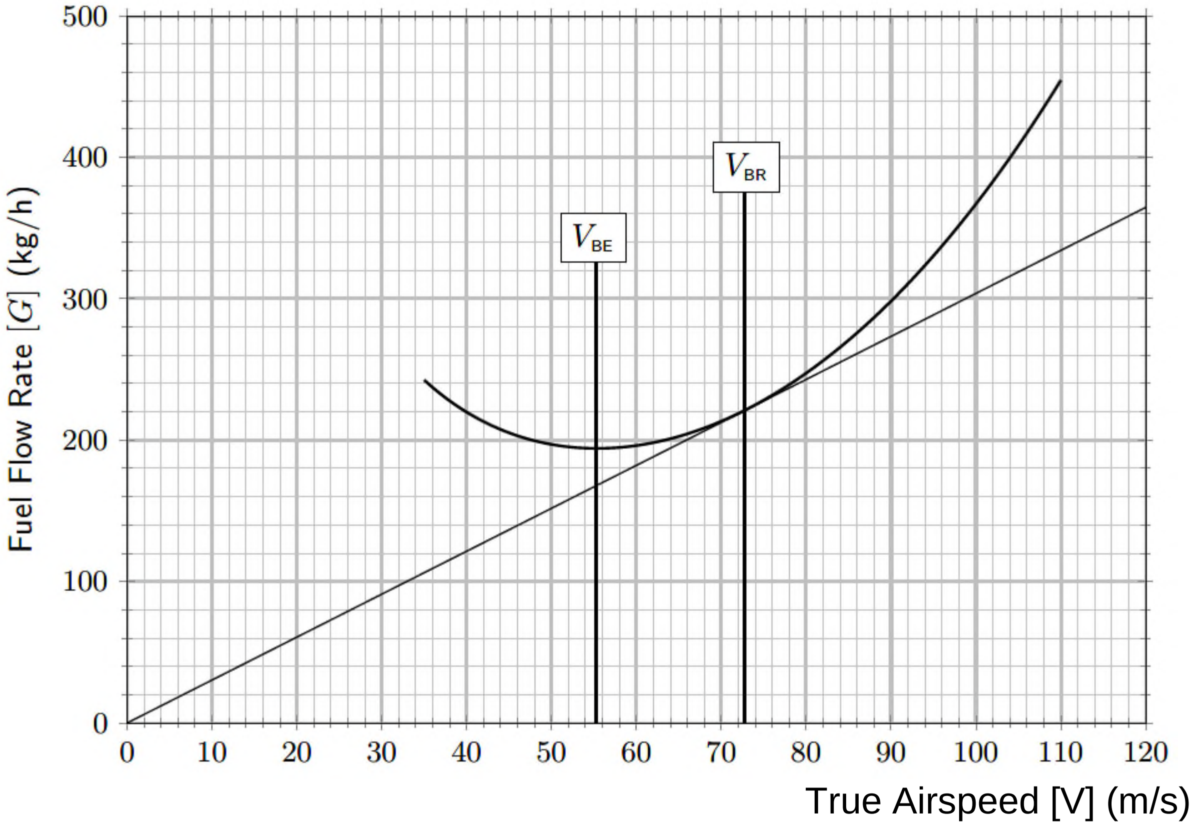


Figure 3.1 – Variation of FFR with TAS in Level Flight

A turbojet engine compresses air continuously and mixes fuel with some of this air. The mixture is burnt and, when mixed with the unburned air, forms a mass of very hot gas at high velocity. This gas passes through a turbine (which drives the compressor) and is ejected from the jet pipe to produce thrust directly. The fuel consumption rate depends on the turbine rpm and the thrust produced, but for an engine operating in the region of its optimum rpm, the thrust produced is proportional to fuel flow.

If the broad assumptions hold true it is possible to make two statements about the operation of aircraft based on the drag and power characteristics:

1. For piston-engined aircraft (power available approximately proportional to fuel flow), V_{BE} is the minimum power speed ($0.76V_{MD}$), and V_{BR} is the speed that maximises the speed to power ratio and that is minimum drag speed (V_{MD}).
2. For jet-engined aircraft (thrust available approximately proportional to fuel flow), V_{BE} is the minimum drag speed (V_{MD}), and V_{BR} is the speed that maximises the speed to drag ratio ($1.32V_{MD}$).

This argument rests on the assumption that the drag polar for the aircraft can be adequately represented by the parabolic relationship given in Eqn 2.2. In practice this is often the case over a reasonable C_L range for a particular aircraft, but it may not give a good fit to measured data at very high or very low values of C_L. For this reason, simple predictions may be inaccurate when compared with real flight data.

For any given aircraft/power plant configuration, one or more of the assumptions may not be true and a plot of measured fuel flow at a particular altitude, temperature, and weight will provide the correct speeds under the specified conditions. Figures 3.2 and 3.3 shows typical **Jetstream** data when tested at the same altitude on the same day.

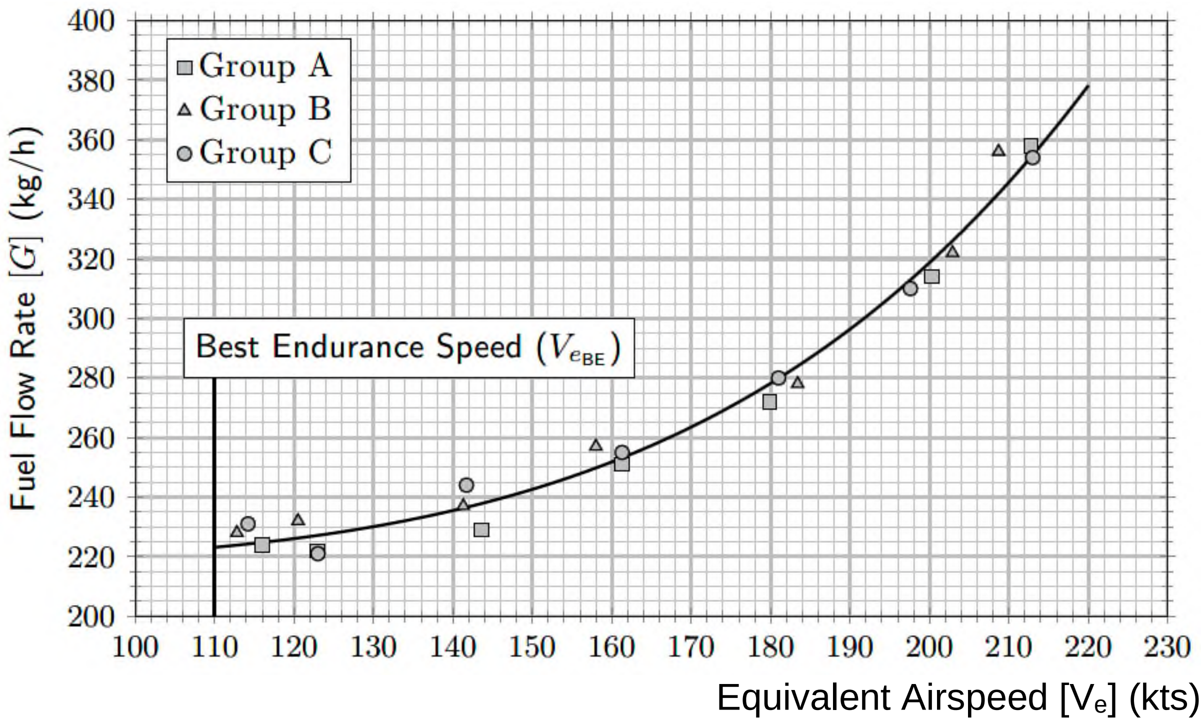


Figure 3.2: Typical Flight Test Data (Endurance)

Normally, the speeds recommended in the flight manual will be faster than the speeds deduced from graphs such as these. The best endurance may be at a speed where the aircraft handling is poor, and so an improvement by increasing the speed is desirable and will not be too detrimental to the endurance, see Figure 3.2. An allowance for weight and altitude change may also be recommended.

The increase in recommended range speed does not significantly reduce the range but produces a faster flight. A head or tail wind component will change the ground speed of the aircraft, and therefore the actual range achieved should take this into consideration. However, the effects on the best range speed are small and unless the head wind exceeds 25% of TAS or the tail wind 33% of TAS, then adjustments to the best range speed need not be made. (Note that the total fuel required must allow for the head/tail wind.)

The experiment will derive the drag characteristic for the **Saab**, and deduce the classic "performance" speeds from it based on the assumption that: $C_D = C_{D0} + KC_L^2$. Fuel flow rate (FFR) will also be recorded so that a comparison may be made between these performance speeds and the measured best range/endurance speeds at a particular weight, altitude, and temperature. Note that the **Saab** is fitted with turboprop engines which fit into neither the piston engine nor jet engine category.

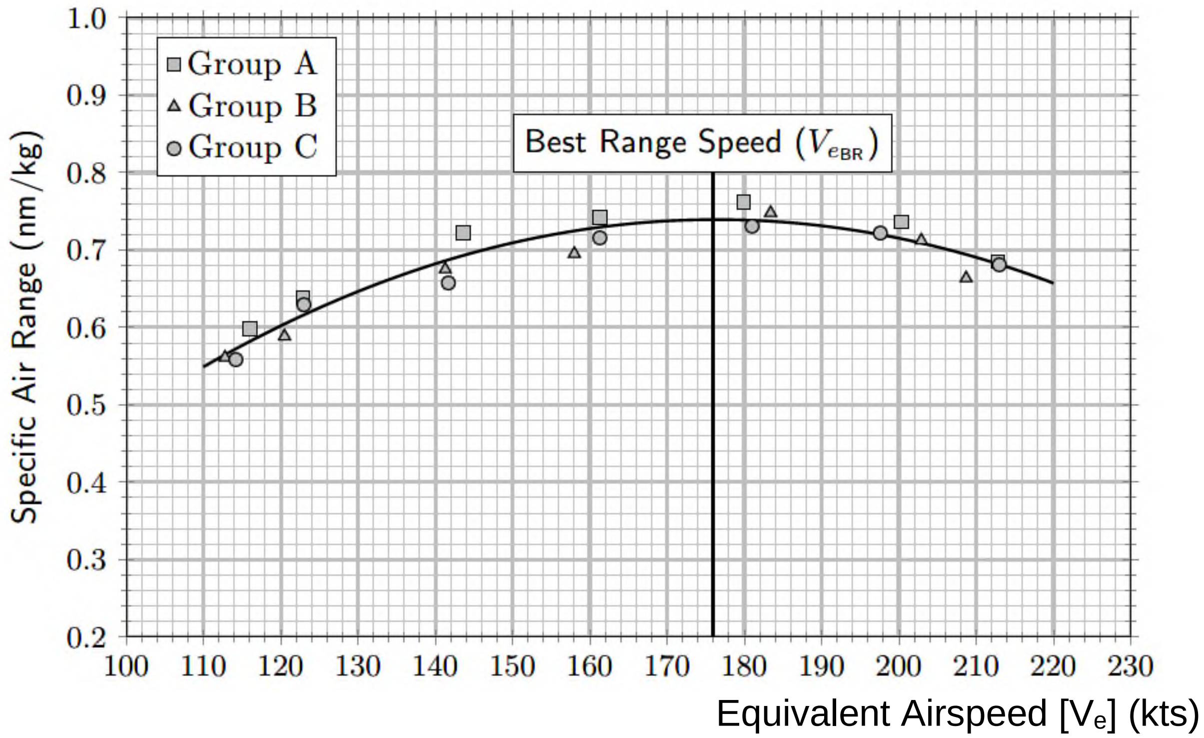


Figure 3.3: Typical Flight Test Data (Range)

3.2 Test Plan - Estimate of Drag and Cruise Performance

Objectives:

- To estimate the aircraft drag in the cruise configuration.
- To determine best endurance and best range speeds and compare with those for thrust producing powerplants and power producing powerplants.

Test Method: ***Measurements in steady, level flight.***

1. At a suitable height, clear of turbulence, set the aircraft in cruising flight at the chosen speed with the elevator trim tab set to give zero force at the yoke/wheel. When steady, record values for:

    - Pressure Altitude, Hp
    - Air Temperature (OAT), T
    - Fuel quantity,
    - Airspeed (EAS), Ve
    - Fuel flow (left and right), FFR_L and FFR_R
    - Thrust (left and right), T_L and T_R

2. At the same altitude change to another speed, and when the aircraft is steady, record values for:

    - Airspeed (EAS), Ve
    - Fuel flow (left and right), FFR_L and FFR_R
    - Thrust (left and right), T_L and T_R

3. Repeat this process to cover the speed range of the aircraft, noting fuel quantity when trimmed for the last data point.

Drag and Cruise Performance Measurement

Altitude .... (ft) OAT .... (°C)

Airspeed [EAS] kt V_e	Fuel Flow [left] (kg/h) FFR_L	Thrust [left] (kN) T_L	Thrust [right] (kN) T_R	Fuel Flow [right] (kg/h) FFR_R	Fuel Quantity (kg)
				Average fuel (kg)	

Table 3.1: Drag & Cruise Performance - Test Data

Data Analysis – Drag and Performance

1. Using the load sheet, calculate the mass of the aircraft during the test (use an average value for the fuel).

2. Calculate C_L and C_D at each speed, using Equations B.1 and B.2,

3. Plot C_D against C_L^2 *(Figure 3.4)* and from the best straight line fit, deduce values for the constants C_{D0} and K in:

   $$C_D = C_{D_0} + KC_L^2$$

   C_{D0} = ...   K = ...

4. Determine the C_L for minimum drag, using *Equation 2.3*: C_L = ...

5. Calculate the *equivalent* airspeed for *minimum* drag for your test weight using a version of Equation B.4:

   V_{eMD} =.............

6. Plot C_D against C_L (*Figure 3.5*), determine the best L/D ratio and compare with the result obtained using Equation 2.4:

   $$\left(\frac{L}{D}\right)_{MAX} = \ldots\ldots\ldots\ldots$$

V_e (kts)	C_L	C_L^2	Drag (N)	C_D

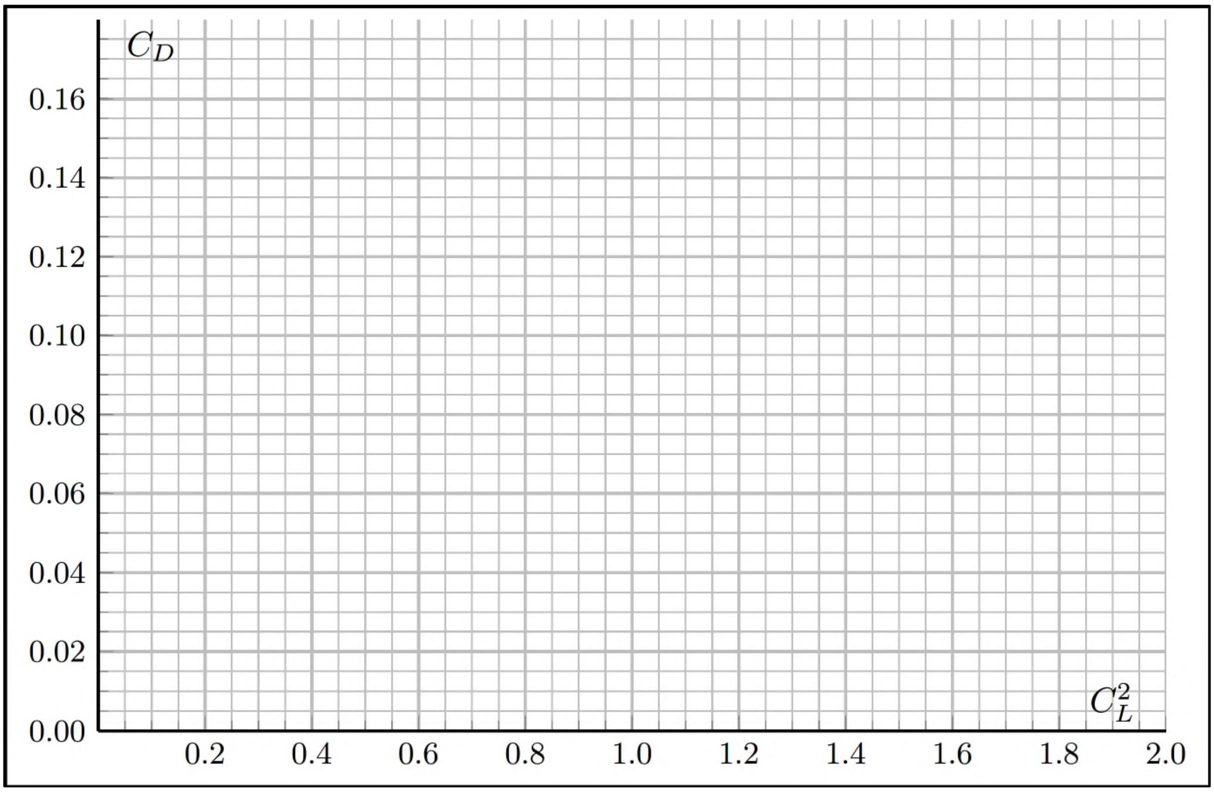


Figure 3.4: Linearised Non-dimensional Drag Characteristic

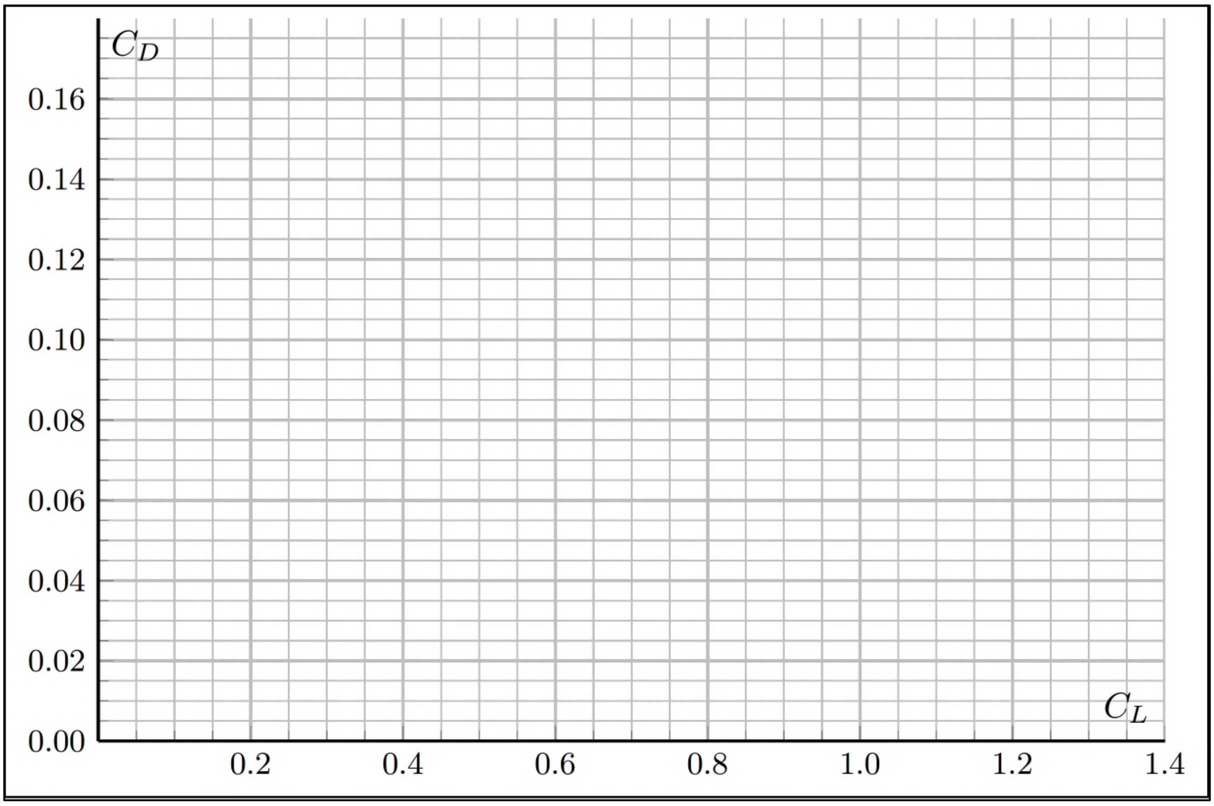


Figure 3.5: Drag Polar

7. Obtain the pressure ratio for the test altitude δ = ...

8. Calculate the temperature ratio θ = ...

9. Determine the true airspeed using Equation B.3

10. Calculate **total** fuel flow at each test point, plot it against Ve and determine the equivalent air speed for best endurance (*Figure 3.6*)

V_{eBE} = ...

11. Calculate SAR (using Eqn 3.1), plot it against Ve and determine the equivalent air speed for best range (*Figure 3.7*)

V_{eBR} = ...

12. Obtain the following speeds and compare with the best endurance and best range speeds:
    (a) minimum drag speed (from Step 5) V_{eMD} = ...
    (b) minimum power speed V_{eMP} = ...
    (c) the speed for the best V/D ratio $Ve_{\ V/Dmax}$ = ...

V_e (kt)	V (kts)	FFR (kg/hr)	SAR (nm/kg)

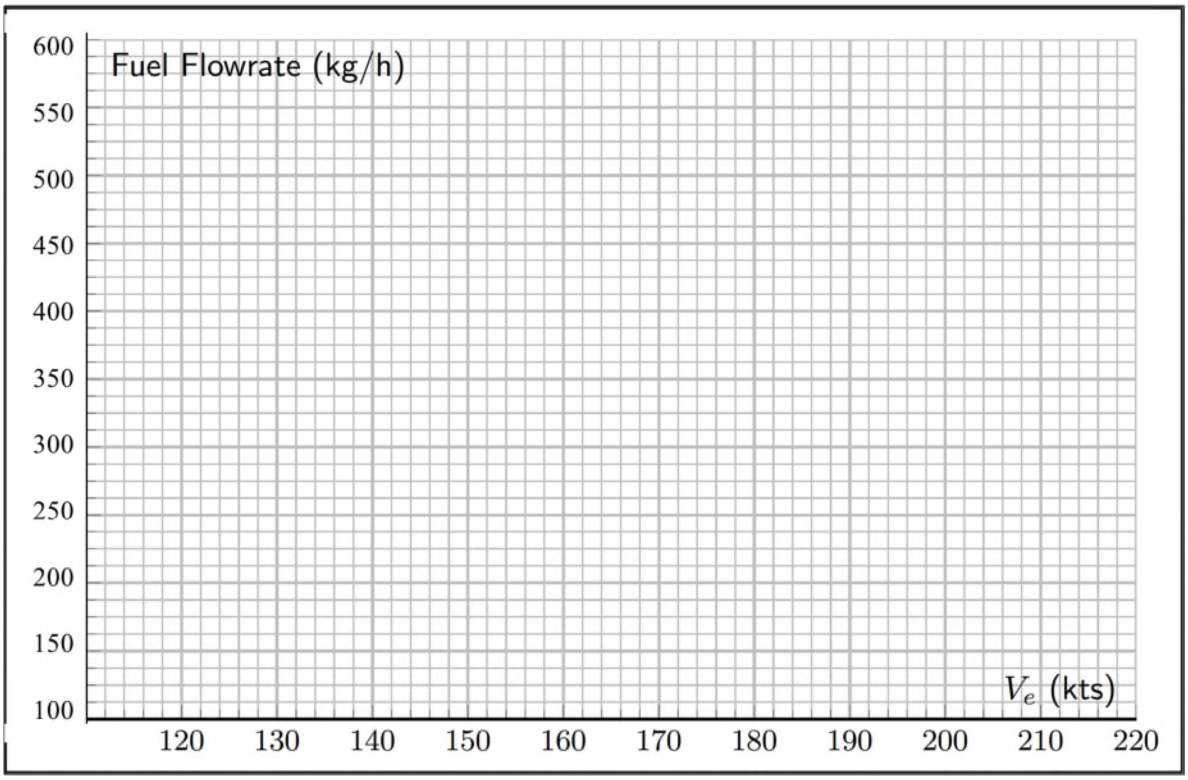


Figure 3.6: Endurance Performance

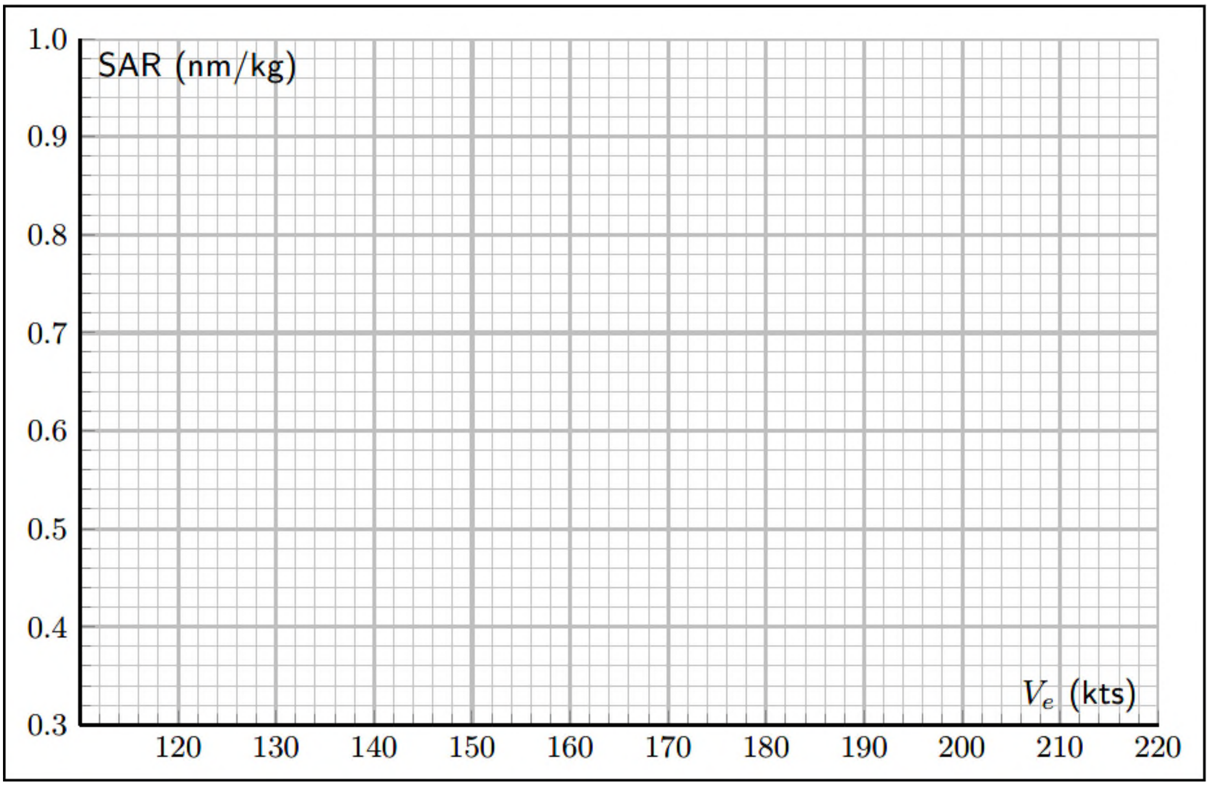


Figure 3.7: Range Performance

4. Longitudinal Static and Manoeuvre Stability

4.1 Longitudinal Static Stability

The longitudinal handling qualities of the aircraft involve motion in the pitching sense. Any disturbance will result in an effective change in the angle of attack (AoA), α. If the aircraft is to exhibit static stability, any disturbance in the AoA must produce a pitch response that will tend to restore the aircraft to the original AoA. Consequently a nose up (+ve) disturbance in AoA should produce a nose down (-ve) pitching moment, M, as a response, that is:

$$\frac{\mathrm{d}M}{\mathrm{d}\alpha} = \mathring{\mathrm{M}}_{\alpha} < 0 \tag{4.1}$$

If an aircraft is stable a simple test can be performed to show the changes of pitching moment relative to changes in AoA. The aircraft is held steady by setting the overall pitching moment to zero. In this state the aerodynamic pitching moment produced by the aircraft is balanced by the moment from the elevator, thus:

$$\mathring{\mathrm{M}}_{\alpha}.\alpha + \mathring{\mathrm{M}}_{\eta}.\eta = 0 \tag{4.2}$$

where $\mathring{\mathrm{M}}_{\alpha}$ is the aerodynamic pitching moment derivative of the aircraft, $\mathring{\mathrm{M}}_{\eta}$ is the aerodynamic pitching moment derivative of the elevator and η is the elevator deflection. This situation is also illustrated in Figure 4.1.

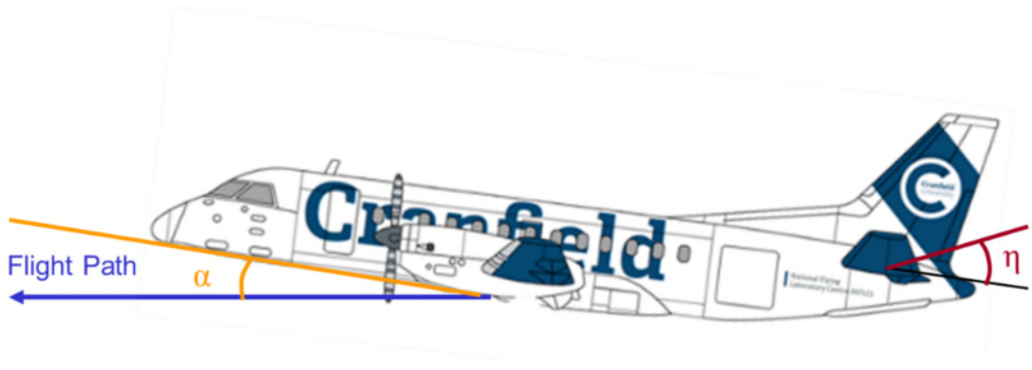


Figure 4.1: Equilibrium Flight with pitch rate (q) = 0

Sign convention - Elevator angle (η) down is positive, up is negative

4.2 Static Stability, Controls Fixed

If the AoA is increased by a small amount, to reduce the airspeed, then a stable aircraft should produce an aerodynamic nose down pitching moment in an attempt to return to the former angle of attack. Thus to hold a stable aircraft at the new AoA will require the elevator to produce an increased nose up moment; this implies that it will have to be moved to a more trailing edge up (negative) position. The new elevator angle will therefore be more negative. This implies that stability is indicated by a negative gradient of the curve of elevator deflection versus AoA, see Figure 4.2.

In practice it is difficult to measure α accurately and so the curve is plotted in terms of the weight coefficient, C_W, since this can be found easily from equivalent airspeed and aircraft weight. Since the curves are measured by reference to the 'static' forces and moments acting on the aircraft this is known as Static Stability. For historic reasons relating to the original test method, when the static stability is measured in terms of the control position changes it is known as *controls fixed* or *stick fixed static stability*.

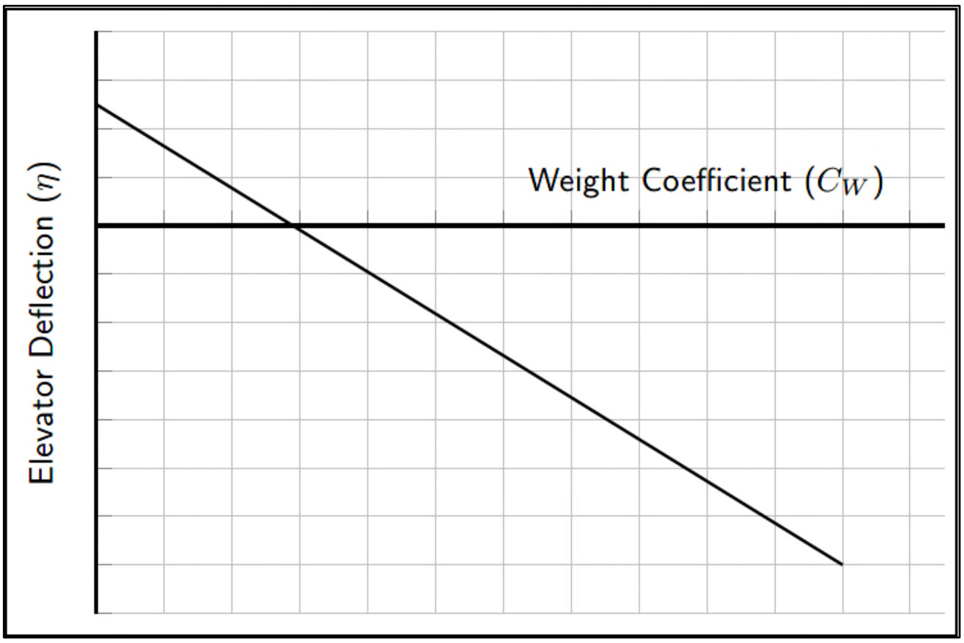


Figure 4.2 Stick Fixed Static Stability

It is found that as the centre of gravity, CG, of the aircraft moves aft the stability decreases since the restoring moment produced by a change in angle of attack is reduced. As a consequence of the reduced moment a smaller change of elevator angle with CW will be required and the slope of the curve will be reduced.

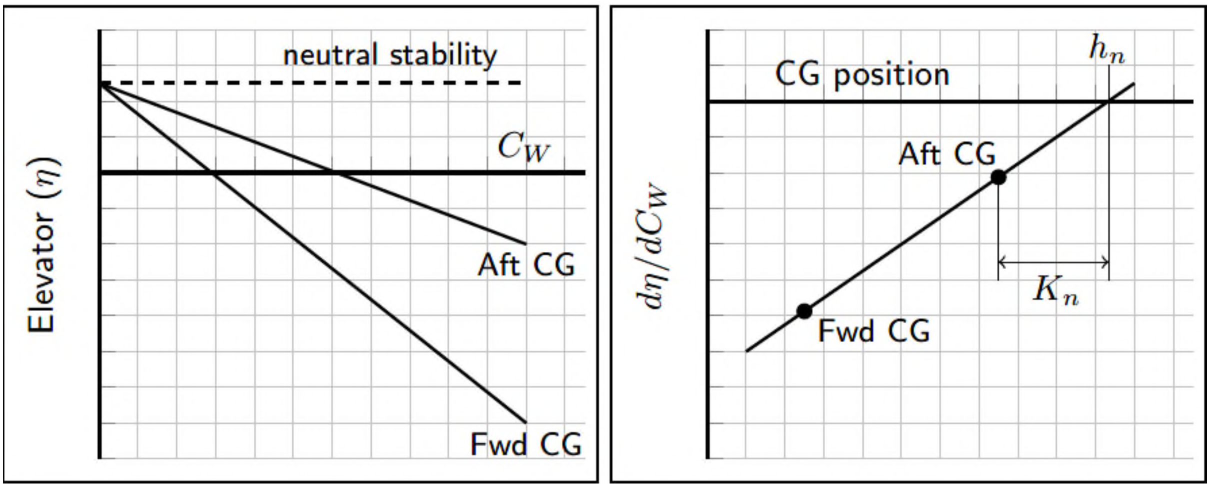


Figure 4.3 Determining the Neutral Point (Stick Fixed)

A CG position will exist at which the slope is zero as no restoring moment is produced, making the aircraft neutrally stable. This situation arises because both the weight and the lift act effectively at the same point. This condition also marks the limit of stability and the CG position is called the *Neutral point, controls fixed*, $\mathbf{h_n}$. The distance between an actual CG position and the Neutral point is known as the static margin, $\mathbf{K_n}$. This Neutral point can be found by plotting the slopes of curves obtained at two, or more, CG locations against the CG location at the time of test; extrapolating to $d\eta/dC_W = 0$ indicates the Neutral point, see Figure 4.3.

Should the CG move further aft than the neutral point the aircraft will become statically unstable. This means that if a disturbance to the steady state of flight occurs then the aircraft will diverge; for example, if the angle of attack decreases by a small amount then it will continue to decrease unless the pilot intervenes to apply a corrective moment through the elevator control. In its stable state the aircraft would maintain its trimmed angle of attack without pilot intervention.

Although discussed in terms of angle of attack; longitudinal static stability (stick fixed) is more readily seen by the pilot in terms of the change in elevator angle, or stick position, necessary to cause a change in speed. In practice such changes in stick position will only be evident to the pilot if they are large.

4.3 Static Stability, Controls Free

As the control movements can be very small; an associated effect, changes in control (or stick) force, is usually more important in determining the ease of airspeed control for the pilot. A simplified version of the elevator control circuit is shown in Figure 4.4: deflection of the elevator control surface will produce a change in the aerodynamic loading on it and hence on the control hinge moment, H_η.

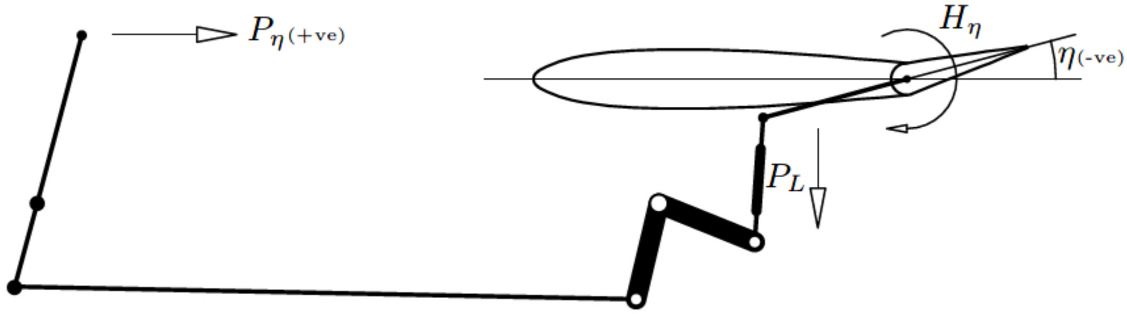


Figure 4.4 Elevator Control Circuit

In a reversible, manual control system this hinge moment is fed back by the linkages to the pilot and felt as a control force, P_η. Thus any movement of the control will require a change in the force applied by the pilot. The force needed from the pilot can be made zero by operating the trim system so that the elevator hinge moment, H_η, is balanced entirely by a hinge moment generated using trim tab deflection, $H\beta$, thus:

$$H_\eta + H_\beta = 0 \qquad (4.3)$$

A 'trim' curve can be drawn either in terms of elevator hinge moment (in coefficient form, C_H) or tab deflection β, against C_W in a similar manner to the curve for the elevator deflection. This is known as the *controls free or stick free static* stability curve or the trim curve since when the test data is recorded the elevator and tab hinge moments balance allowing the elevator to 'float' freely at the required defection. By the general convention of control forces; a positive (pull) force should be required to reduce airspeed. This achieved if the elevator moves to a more negative position. Consequently a positive slope of β (or C_H) with C_W ($d\beta /dC_W$) will indicate stability, see Figure 4.5.

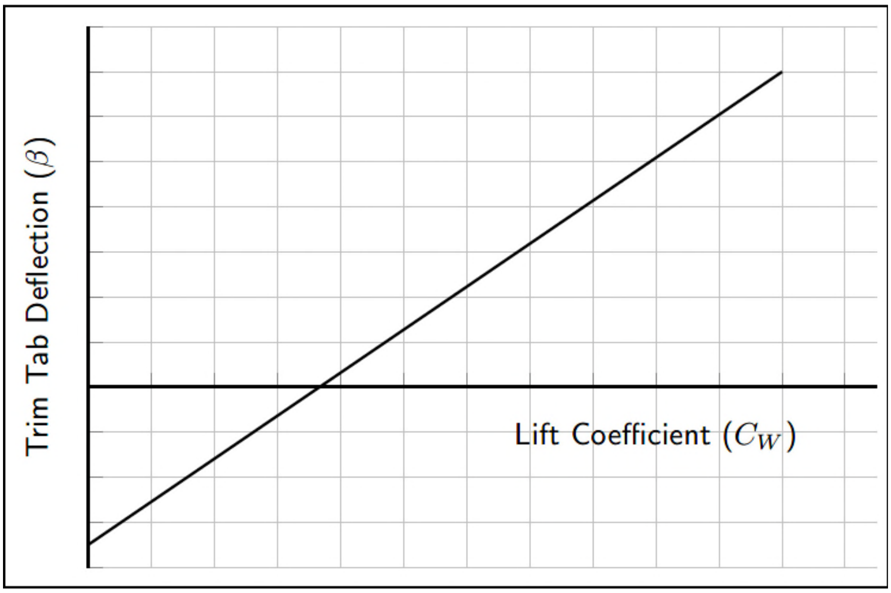


Figure 4.5 Stick Free Static Stability

The CG position also affects the rate of change of trim tab deflection with lift coefficient; decreasing the slope as it moves aft. A CG position will occur at which the trim curve slope is zero indicating neutral static stability, controls free. The Neutral point (**h_n'**) can be found in a similar manner to the controls fixed neutral point, see Figure 4.6. Usually the two neutral points will occur at different CG locations; this arises from the aerodynamic balancing of the elevator control and the lost 'force' due to friction in the control circuit. Thus typically:

$$h'_n \leq h_n \tag{4.4}$$

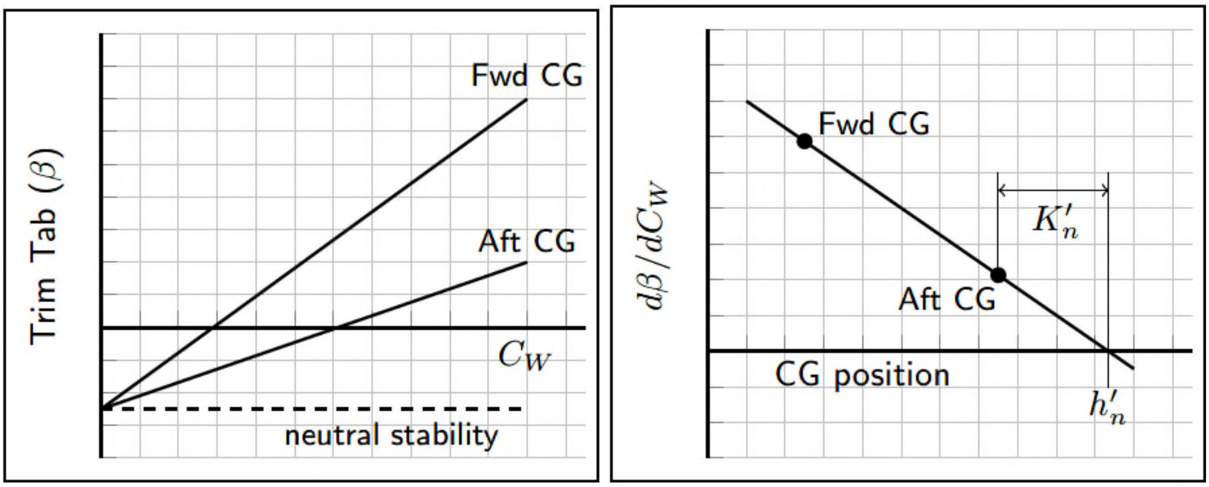


Figure 4.6 Determining the Neutral Point (Stick Free)

Controls free static stability is more apparent to the pilot in terms of the stick force necessary to produce a change in speed. To maintain the correct 'feel' the slope of the control force against speed curve must be negative and be greater than a minimum value. This requirement is stated in CS 25.173:

> ***<u>CS 25.173 Static longitudinal stability</u>***
>
> "...*Under the conditions specified...[cruise speed with flaps and landing gear retracted, the CG in the most adverse position and the most critical weight between MTOW and MLW]...the characteristics of the elevator control forces (including friction) must be as follows:...*
>
> *(c) The average gradient of the stable slope of the stick force versus speed curve may not be less than 1 pound for each 6 kts, [7.4 N/10 kts]..."*

and from AMC 25.173(c):

> "*The average gradient is taken over each half of the speed range between 0.85 and 1.15 Vtrim*."

2.4 Static Stability Measurement

Static stability testing represents a means of assessing some of the handling qualities of an aircraft in a simple and economical manner. It is only necessary to fly the aircraft at a number of steady speeds and to measure the elevator defection, trim tab deflection and airspeed. The neutral points can be determined if this is done at two or more CG locations.

Although the data is recorded when the aircraft is in a stick-free state we can also obtain the stick-fixed data by assuming that the trim tab deflection does not affect the elevator deflection required.

4.5 Manoeuvre Stability

A manoeuvre is defined as the change from one state of flight to another; therefore a manoeuvre will be required if the aircraft is to change its flight path, γ. In the longitudinal plane the manoeuvre is characterised by a pitch rate, q. Since the aircraft has a forward speed, the pitch rate will cause a change in the AOA which will produce a change in the force normal to the instantaneous direction of flight. This force is usually referred to as the 'g-force' felt by the aircraft occupants during the manoeuvre.

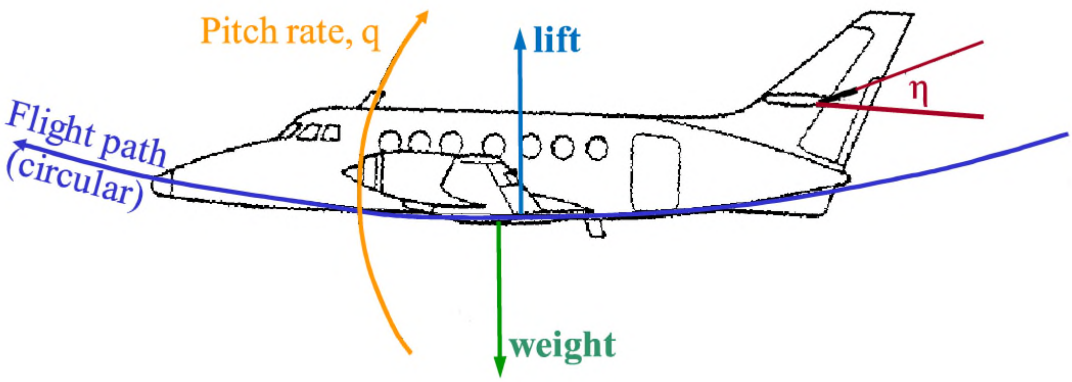


Figure 4.7 Manoeuvring Flight

Sign convention - Elevator angle (η) down is positive, up is negative

In a conventional aircraft the manoeuvre is produced by an input to the elevator control, see Figure 4.7. Pulling the stick back to a new position by applying an positive increment to the stick force, $\Delta P\eta$, changes the elevator angle by an amount $\Delta\eta$ and imparts a nose-up pitching moment to the aircraft as well as an increase in the angle of attack, $\Delta\alpha$. The aircraft will accelerate in pitch until the pitch damping produced by the tailplane along with the moment caused by the increase in incidence equals the pitching moment from the elevator:

$$\mathring{M}_q.q + \mathring{M}_\alpha.\alpha + \mathring{M}_\eta.\eta = 0 \quad \text{or} \quad \mathring{M}_n.n + \mathring{M}_\eta.\eta = 0 \qquad (4.5)$$

where $\mathring{M}_q$; $\mathring{M}_\alpha$; $\mathring{M}_\eta$; $\mathring{M}_n$ are pitching moments due to (respectively) pitch rate; angle of attack; elevator deflection; normal acceleration.

The pitch rate, and hence the manoeuvre, then becomes steady; this is a very rapid process which is strongly influenced by the short period pitching mode of the aircraft. The steady pitching manoeuvre approximates to circular motion in the vertical plane. The rate of pitch, and hence the 'g' force experienced during the manoeuvre, will depend on the elevator input, a small input will produce a low pitch rate and hence a low ‘g’ force increment whereas a large input will produce a high pitch rate and hence a high increment.

The relationship between the pilot input, in terms of both the control movement and the applied force, and the resultant manoeuvre is very important in the handling qualities of the aircraft since it determines the stick travel per g and the stick force per g. A result of their connection with static stability, these attributes are referred to as the controls fixed manoeuvre stability and the controls free manoeuvre stability, respectively. Positive manoeuvre stability is shown by the slope of the curves of incremental input (negative deflection or positive force) and increased g produced, see Figures 4.8 and 4.9.

As the pitching moment is affected by the CG position the manoeuvre characteristics of the aircraft will be CG dependent. Both the stick travel per g and the stick force per g will decrease as the CG moves aft. Points will be reached at which either will become zero; these are known as the ***manoeuvre points***. The distance between the manoeuvre point and the CG is known as the manoeuvre margin; this indicates the strength of the tendency for the aircraft to return to ‘1-g’ flight after a disturbance. If the CG is behind the manoeuvre point then a control input will not result in a steady manoeuvre but in a divergent manoeuvre with an increasing rate of pitch.

The manoeuvre characteristics of the aircraft are measured by pitching to increase and hold the g-loading then measuring the control input required. The manoeuvre can be a *pull-up* producing a pure pitching motion but this is both difficult to perform accurately and time consuming. It is more usual to perform a turn in which the g-force is increased by the component of the rate of turn in the pitching plane. Turns can be either *steady state or wind-up* turns in which the turn rate is continuously increased. The process of measurement is to fly the aircraft in trim, 1-g conditions, and to measure the control position and applied force, (which should be zero if trimmed). The aircraft is then manoeuvred to increase the g-force and this is recorded together with the control position; this is repeated for several levels of g.

Plotting the control position and applied force against g will give the control travel per g and the control force per g respectively. By repeating the process for alternative CG positions the control travel per g and control force per g can be plotted against CG position to give estimates of the manoeuvre points, stick fixed and stick free respectively, see Figures 4.8 and 4.9. The manoeuvre points will be **aft** of the neutral points, that is **h_m > h_n**, by an amount which is a function of the pitch damping of the aircraft.

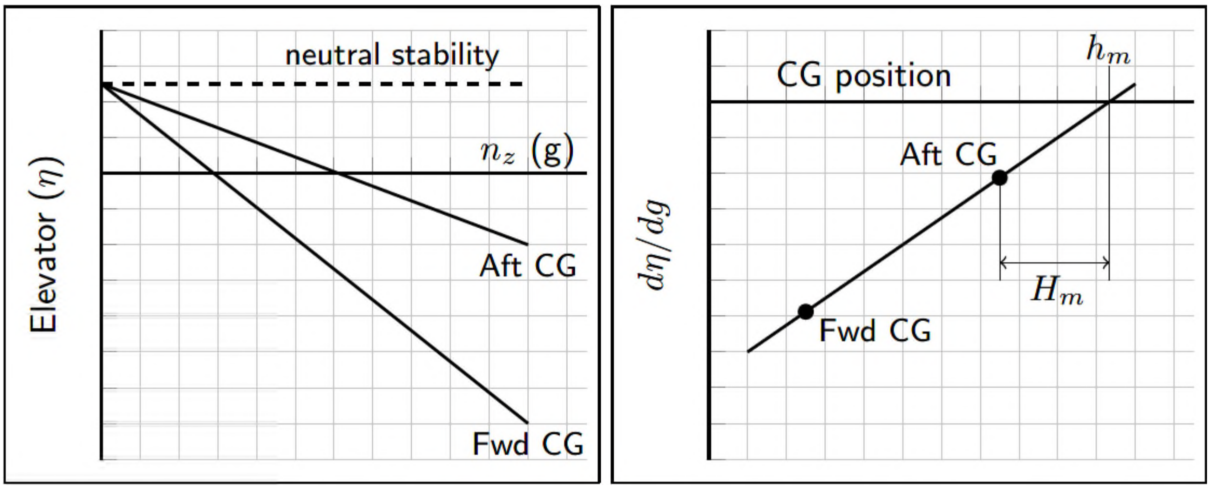


Figure 4.8 Determining the Manoeuvre Point (Stick Fixed)

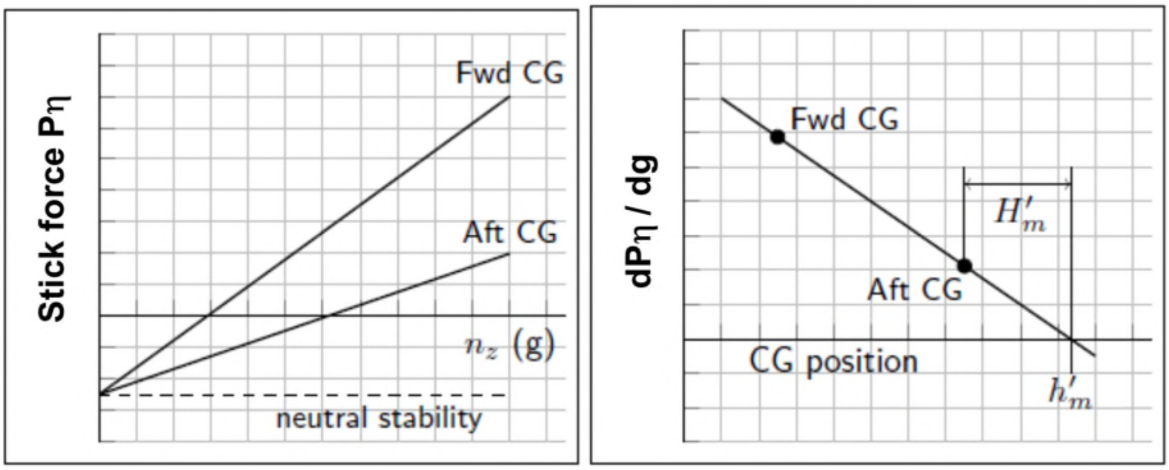


Figure 4.9 Determining the Manoeuvre Point (Stick Free)

4.6 Stick Force Requirements

Stick force per g is a very important handling characteristic. Large transport aircraft are designed to operate at low load factors, typically 2.5g, since they do not need to manoeuvre aggressively and must be efficient structurally. To avoid over-stressing these aircraft high levels of stick force per g are engineered, of the order of 100 lb per g. On the other hand a fighter aircraft with a manoeuvre envelope of say +9g to -3g needs to be manoeuvred very aggressively and the stick force per g will need to be much lower, say around 10 lb per g. The requirement for large transport aircraft is stated in CS 25.143 (g):

> *"When manoeuvring at a constant airspeed or Mach number (up to V_{FC}/M_{FC}), the stick forces and the gradient of the stick force versus manoeuvring load factor must lie within satisfactory limits. The stick forces must not be so great as to make excessive demands on the pilots strength when manoeuvring the aeroplane (see AMC No.1 to CS 25.143 (g)), and must not be so low that the aeroplane can easily be overstressed inadvertently."*

where AMC No.1 to CS 25.143 (g) states that:

> *"An acceptable means of compliance with the requirement that stick forces may not be excessive when manoeuvring the aeroplane, is to demonstrate that, in a turn for 0.5g incremental normal acceleration* (0.3g above 6096 m (20000 ft)) *at speeds up to V_{FC}/M_{FC}, the average stick force gradient does not exceed **534 N** (120 lbf)/g."*

Note that AMC No.2 to CS 25.143 (g) states that:

> "**2 *Minimum Stick Force to Reach Limit Strength***
>
> *2.1 A stick force of at least 222 N (50 lbf) to reach limit strength in steady manoeuvres or wind up turns is considered acceptable to demonstrate adequate minimum force at limit strength in the absence of deterrent buffeting. If heavy buffeting occurs before the limit strength condition is reached, a somewhat lower stick force at limit strength may be acceptable. ...*
>
> *2.2 This minimum stick force applies in the en-route configuration with the aeroplane trimmed for straight flight, at all speeds above the minimum speed at which the limit strength condition can be achieved without stalling. ..."*

4.5 Measurement of Pitch Control Force: G-NFLB

The force applied by the pilot is measured, on the test aircraft, by a strain gauge located near to the control column. The measured force can be converted to an equivalent 'stick' force ($P\eta$) which is then displayed on the relevant screen. As the force is measured on the elevator control rod, there is no need to include the mechanical gearing in the control circuit. There is a constant force P_{spring} due to a nose-down spring included the elevator control system.

Test Plan

Evaluation of Longitudinal Static Stability

Objectives:

- To assess the longitudinal static stability of the aircraft
- To estimate the neutral points, controls fixed and controls free, and to suggest an aft limit for the CG range.

Test Method: ***Change of angle of attack from cruising flight datum***

1. At a suitable height, clear of turbulence, set the aircraft in cruising flight at **180 knots**, flaps up, landing gear up, with the elevator trim tab set to give zero force at the yoke/wheel. When steady, record values for:

- Elevator angle, η;
- Elevator trim tab angle, β;
- Airspeed (EAS), Ve;
- Fuel quantity.

2. Reduce speed by 10 knot decrements down to **160 knots**. At each speed maintaining the same power setting, change the trim tab position to zero the control force. When steady at each speed, record values for:

- Elevator angle, η;
- Elevator trim tab angle, β;
- Airspeed (EAS), Ve;

3. Increase speed by 10 knot increments up to **200 knots**. At each speed maintaining the same power setting, change the trim tab position to zero the control force. When steady at each speed, record values for:

- Elevator angle, η;
- Elevator trim tab angle, β;
- Airspeed (EAS), Ve;
- Fuel quantity when at last test point.

This test will be performed at two or more CG positions so that estimates of the neutral points (h_n and h'_n) can be made.

Longitudinal Static Stability Measurement

Airspeed [EAS] kts V_e	Elevator Angle (deg) η	Elevator Tab Angle (deg) β	Fuel Quantity (kg)
		Average fuel (kg)	

Table 4.1a: LSS Test Data

Mass ……………….kg   CG position ……………(%)

V_e (m/s)	C_w	η (deg)	β (deg)

Table 4.1b: LSS Test Calculated Values

Test Plan - Evaluation of Manoeuvre Stability

Objectives:

- To assess the manoeuvre stability of the aircraft.
- To estimate the location of the manoeuvre points, controls fixed and controls free.
- To suggest an aft limit for the CG range

Method: ***Steady turns from a cruising flight datum***

1. At a suitable height, clear of turbulence, set the aircraft in cruising flight at 160 knots, flaps up, landing gear up, with the elevator trim tab set to neutral. When steady, record values for:

   - Elevator angle η
   - Elevator force $P\eta$
   - Normal acceleration n_z
   - Airspeed (EAS) V_e, for reference
   - Fuel quantity, for reference.

2. Maintaining the same power setting and speed, without retrimming, perform steady turns with bank angles up to 60°; when steady, record values for:

   - Elevator angle η
   - Elevator force $P\eta$
   - Normal acceleration n_z

This test will be performed at two or more CG positions so that estimates of the manoeuvre points (h_m and h_m') can be made. The pitch trim will be set to the same datum position for all flights.

Manoeuvre Stability Measurement

Elevator Angle (deg) η	Stick Force (N) Pη	Normal Acceleration (g) n_z

Table 4.2: LMS Test Data

Mass _______________ CG position ______________

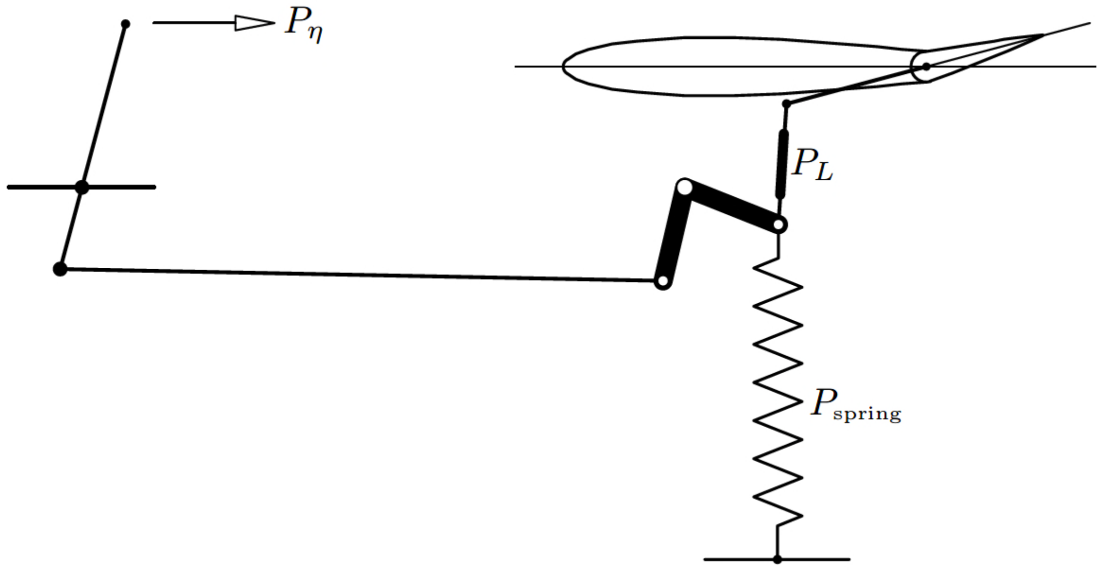


Figure 4.10 Simplified Elevator Control Circuit

Data Analysis

Longitudinal Static Stability

S1. Estimate the weight and CG location using the mean fuel quantity.

S2. Calculate the weight coefficient at each point using:

$$C_W = \frac{2mg}{\rho_0 V_e^2 S}$$

S3. Plot elevator angle η against C_W for each CG position.

S4. Plot the slope of the line ($d\eta$ / dC_W) against CG position and determine the ***stick- fixed*** neutral point using similar data from other groups.

S5. Plot trim tab deflection β against C_W for each CG position.

S6. Plot the slope of the line ($d\beta$ / dC_W) against CG position and determine the ***stick- free*** neutral point using similar data from other groups.

Manoeuvre Stability

M1. Plot the elevator angle η against g. Determine the gradient of the line, making use of a common intercept on the y-axis.

M2. Plot the slope of the line ($d\eta$ /dg) against the CG position, obtained in (S1) above and determine the stick fixed manoeuvre point using similar data from other groups.

M3. Record the 'stick' force, $P\eta$, at each data point.

M4. Plot the stick force $P\eta$ against g. Determine the gradient of the line, making use of a common intercept on the y-axis.

M5. Plot the slope of the line ($dP\eta$ /dg) against CG position, obtained in S1 above and determine the stick free manoeuvre point using similar data from other groups.

M6. What is the stick force per g with the CG at 16% and 37%? How do these figures compare with the CS requirements?

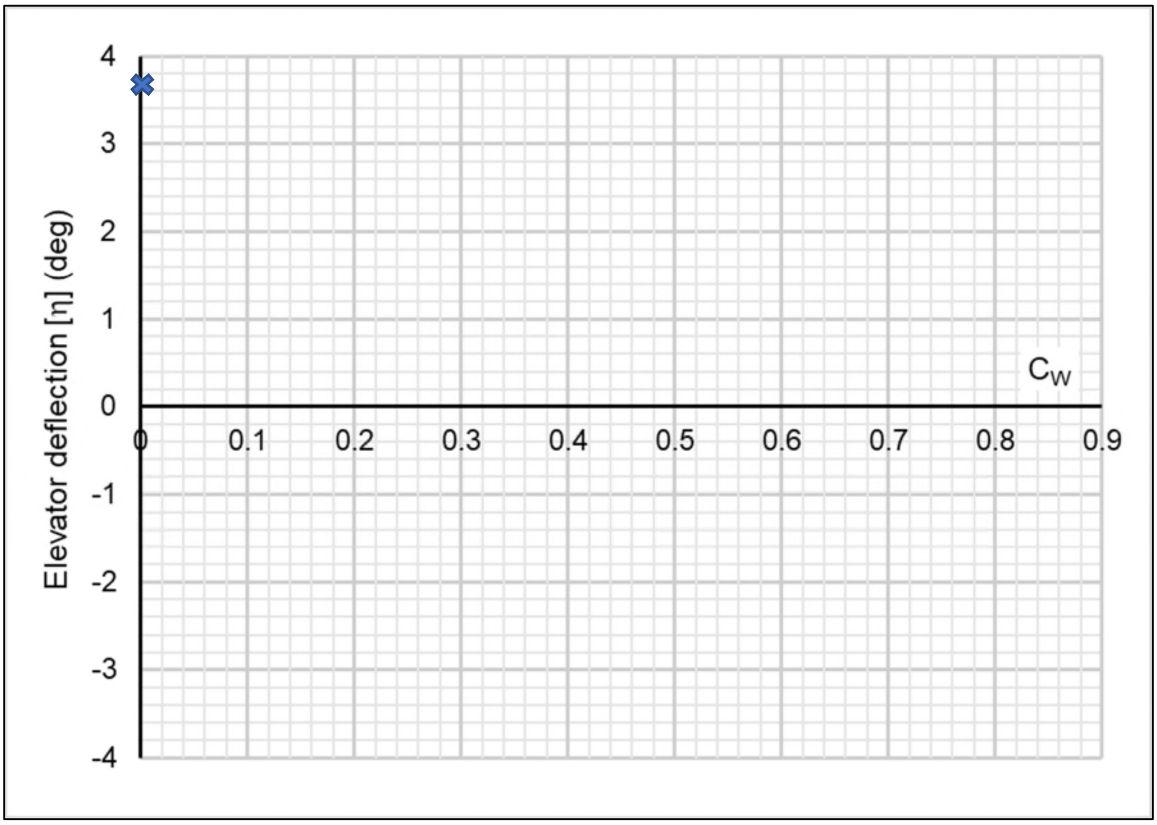


Figure 4.11 ***Elevator angle vs C_W - Static Stability, Controls Fixed***

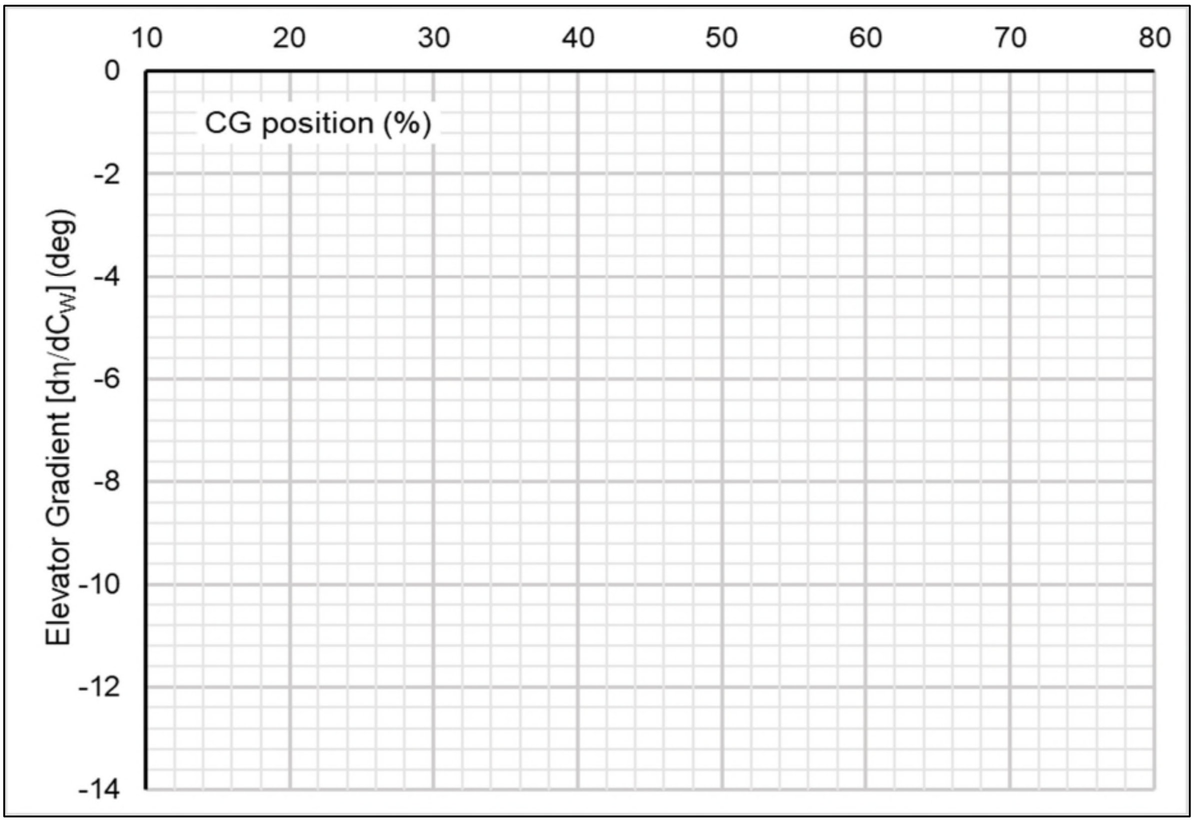


Figure 4.12 ***Graph to Determine Neutral Point, Controls Fixed***

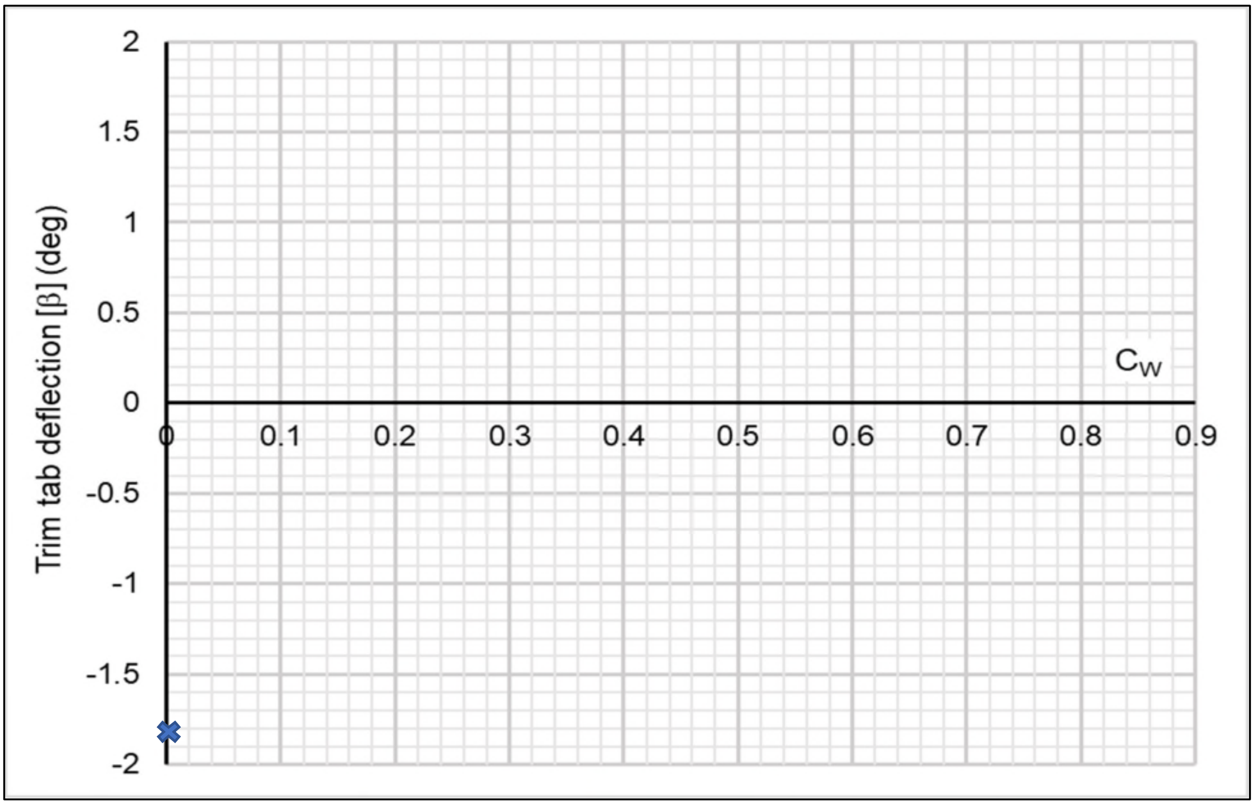


Figure 4.13 ***Elevator tab angle vs C_W - Static Stability, Controls Free***

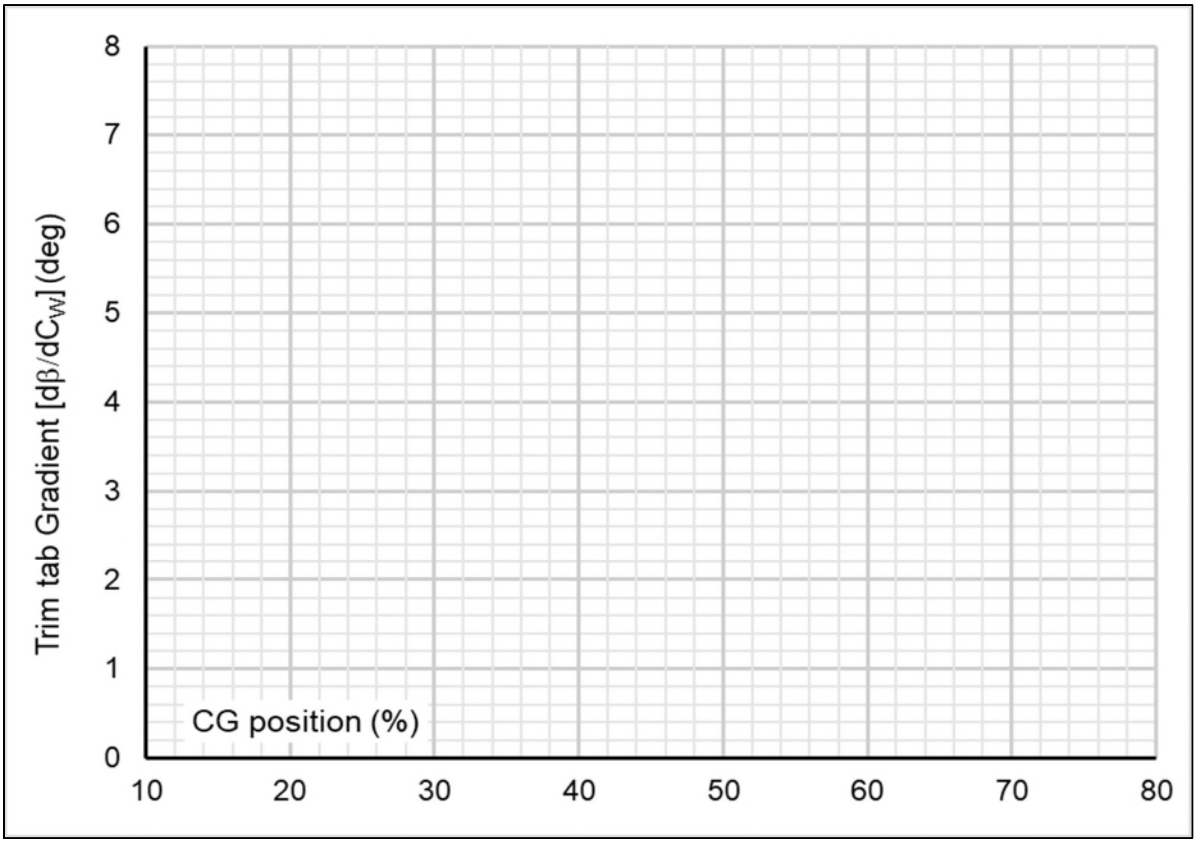


Figure 4.14 ***Graph to Determine Neutral Point, Controls Free***

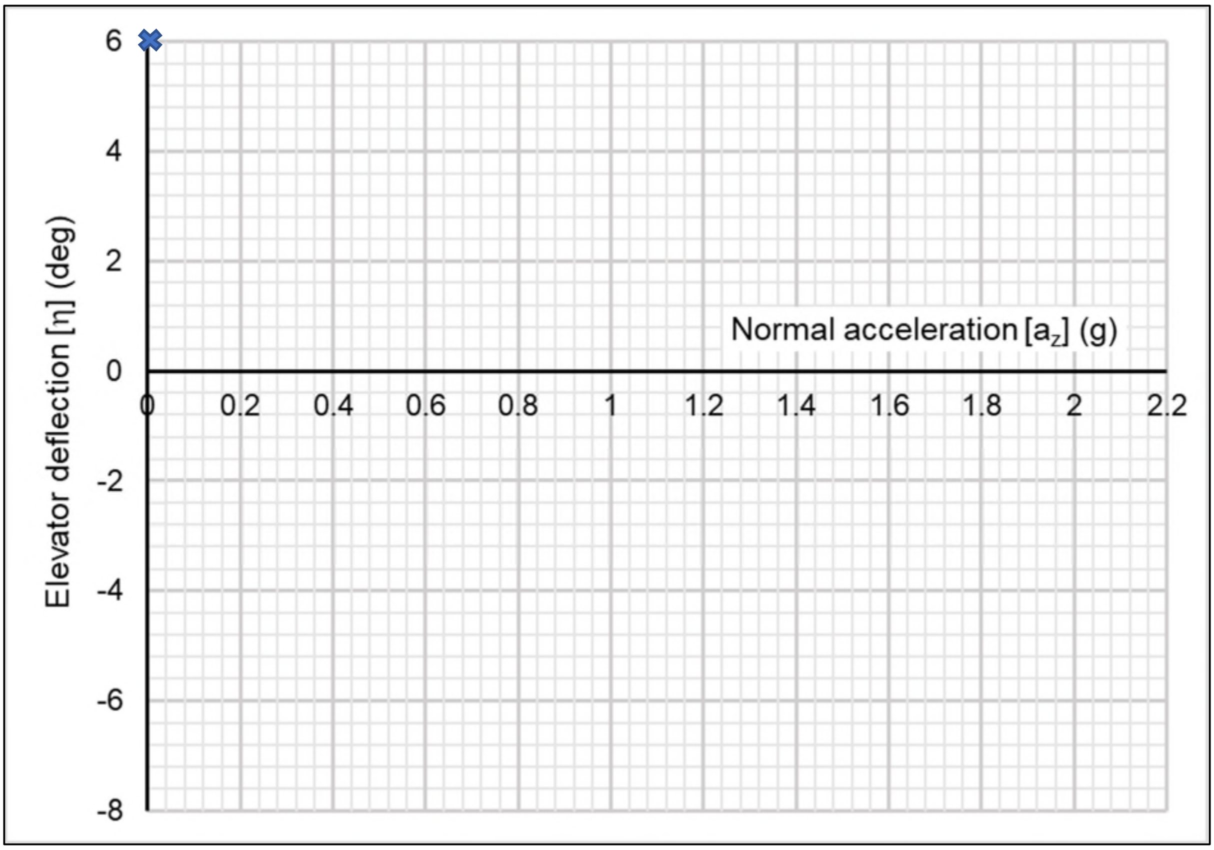


Figure 4.15 ***Manoeuvre Stability, Controls Fixed***

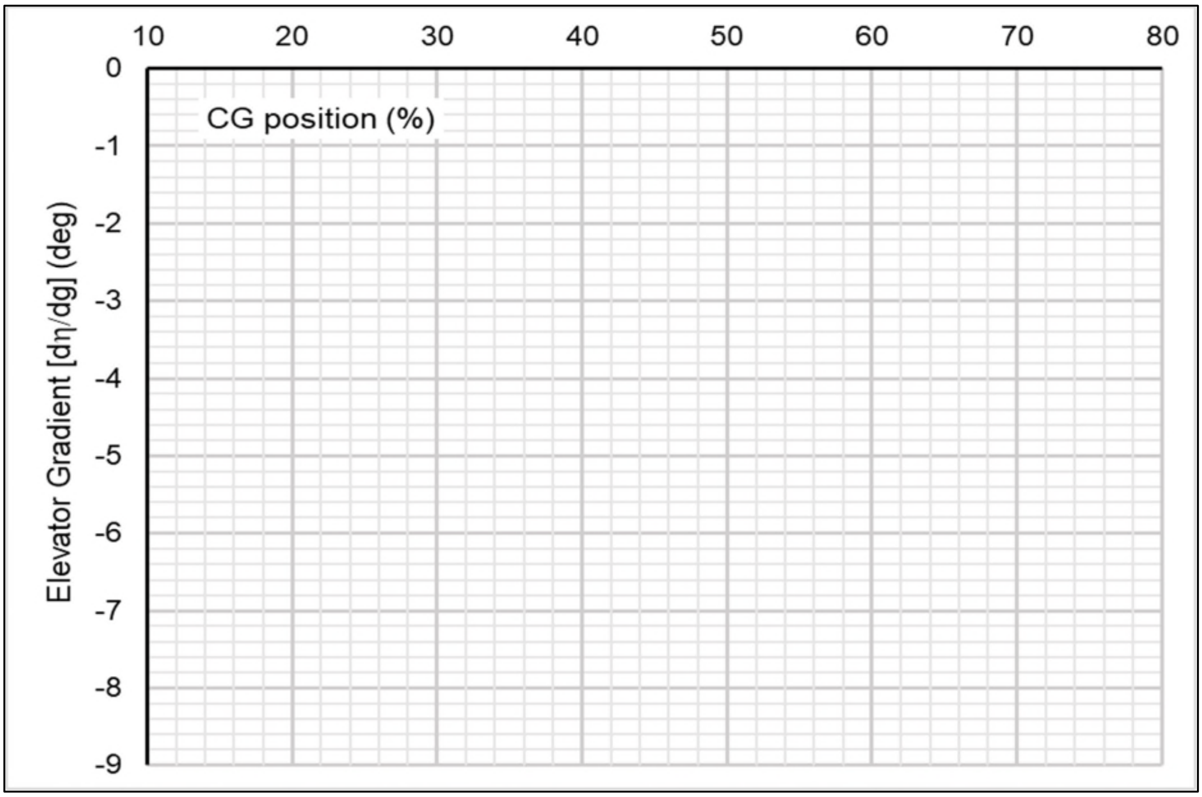


Figure 4.16 ***Graph to Determine Manoeuvre Point, Controls Fixed***

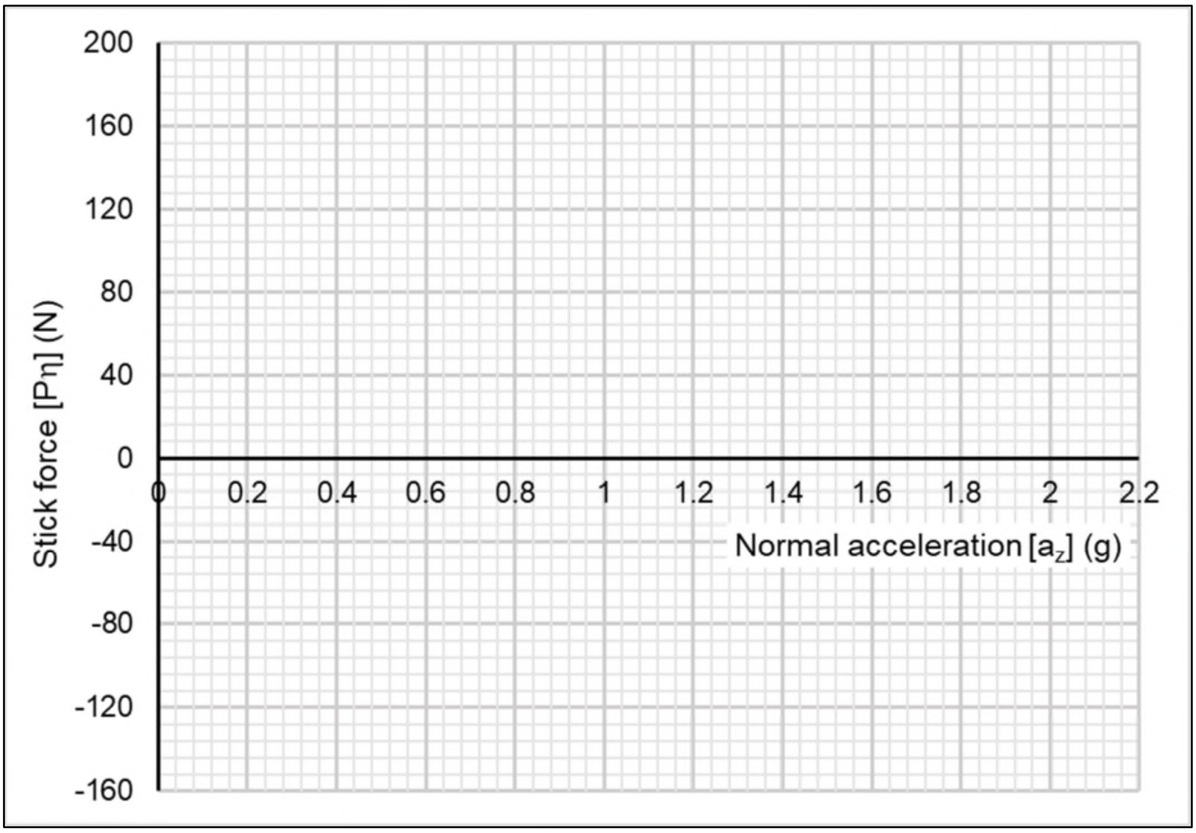


Figure 4.17 ***Stick Force vs a_z - Manoeuvre Stability, Controls Free***

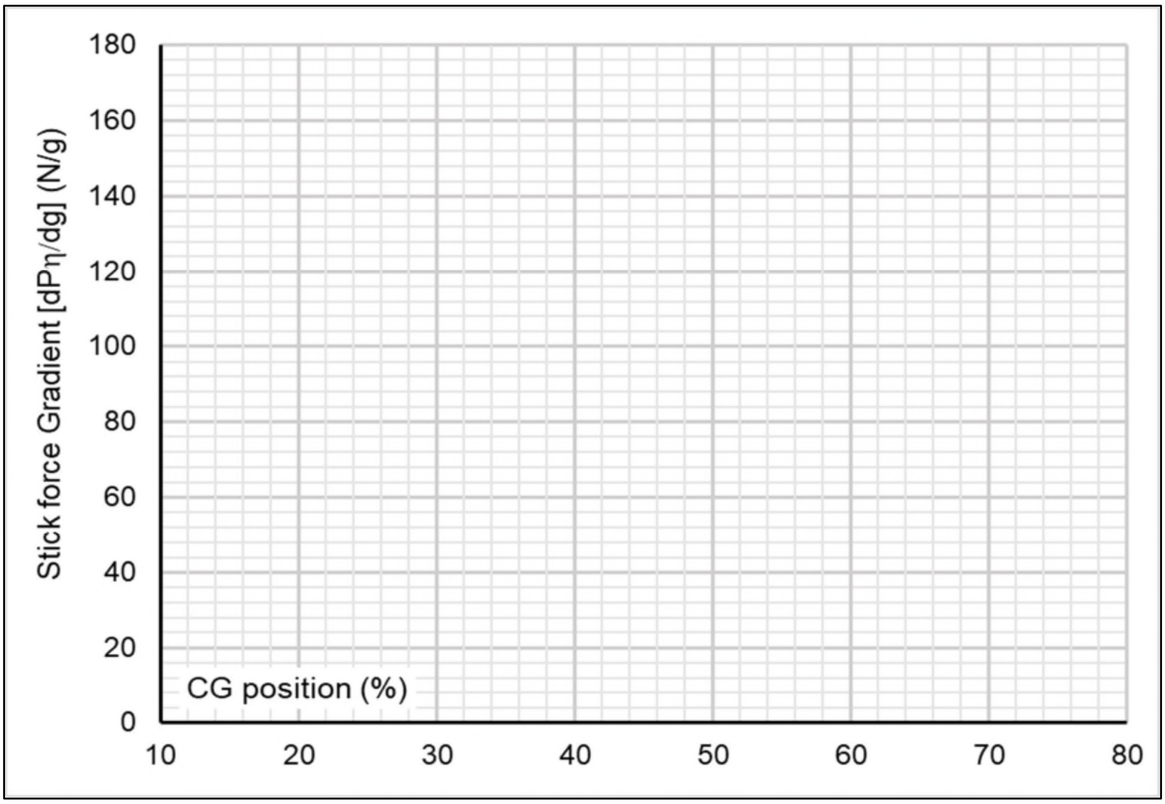


Figure 4.18 ***Graph to Determine Manoeuvre Point, Controls Free***

5. Lateral-Directional Static Stability

In simple terms, the lateral static stability of an aircraft concerns its tendency to return towards a wings level state from a steady sideslip whilst the directional static stability concerns the tendency to reduce the sideslip angle to zero and align its longitudinal axis with the relative airflow. Whilst these conditions are stated separately they are, in fact, very much interrelated and it is not possible to completely separate them.

In the case of longitudinal static stability it was seen that stability is indicated by the slope of the 'trim' curves with respect to a change in angle of attack α. In effect, the trim curve indicates the magnitude of the restoring moment generated by the aircraft when it is forced into an out-of-trim state by application of the elevator control.

In the lateral-directional case a similar analysis can be used to show that the application of aileron and rudder control will provide trim curves that indicate whether the aircraft exhibits static stability with respect to a disturbance in sideslip angle β, which can be regarded as a "lateral angle of attack".

Sideslip can be quantified either in terms of a lateral velocity component, v, or a lateral 'angle of attack' β. Note that when sideslipping the longitudinal axis of the aircraft is no longer coincident with its velocity vector (flight path) and the aircraft flies in a crabwise manner.

5.1 Directional Static Stability

Positive directional static stability comes almost entirely from the fin; most other parts of a conventional aircraft tend to be directionally destabilising. When a directionally stable aircraft is held in a steady sideslip it will produce a yawing moment due to the sideslip, $\mathring{N}_{\beta}.\beta$, which will tend to yaw the aircraft out of the sideslip back into symmetric flight.

To maintain the sideslip, yaw control must be used to overcome the stabilising yawing moment and so the rudder deflection z can be used as a measure of the magnitude of the stabilising moment in yaw. By flying a number of sideslips and recording z, applied force, P_z and indicated sideslip b, the 'trim' curves of directional static stability can be drawn for both the 'controls fixed' and 'controls free' states.

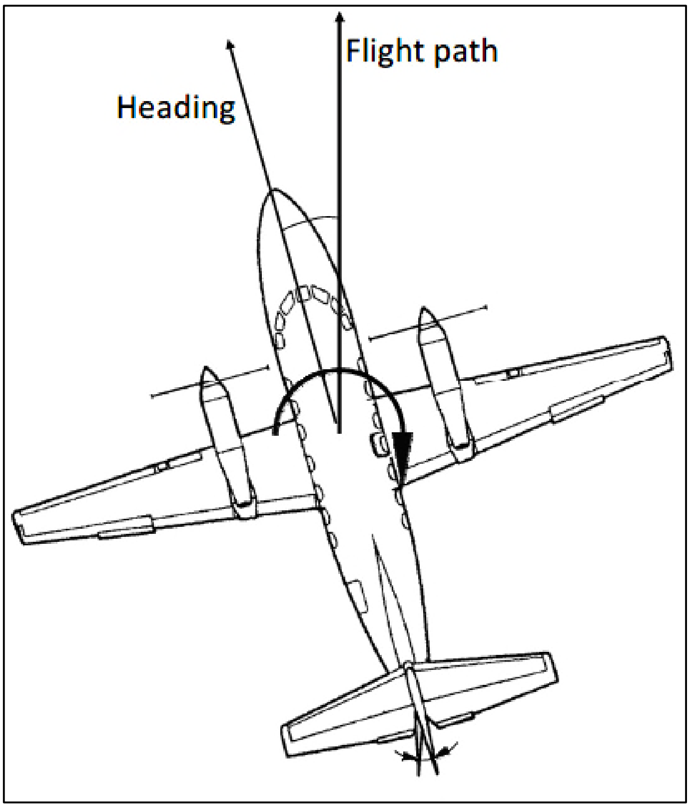


Figure 5.1 Plan view showing angle of sideslip

The slopes of the curves indicate the degree of directional stability; under the normal sign convention a positive slope of ζ against β, and a negative slope of P_ζ against β, shows that the aircraft exhibits controls fixed and controls free static stability respectively. Note that the yawing moment due to sideslip, $\mathring{N}_v$ must be positive if the aircraft is stable. Alternatively directional static stability can be gauged directly by noting the rudder deflection direction required for a given sideslip angle: ***left*** rudder for right sideslip indicates that the aircraft is directionally stable.

5.2 Lateral Static Stability

The lateral stability of the Saab 340B comes mainly from the dihedral angle of the wings; on other aircraft sweepback and high wing configurations also have a stabilising effect. When a laterally stable aircraft is held in a steady sideslip it will produce a rolling moment, $\mathring{L}_\beta.\beta$ which will tend to roll the aircraft away from the sideslip towards wings level flight. To maintain the sideslip aileron control must be used to balance the stabilising moment and so the aileron deflection ξ can be used as an indicator of the magnitude of the moment.

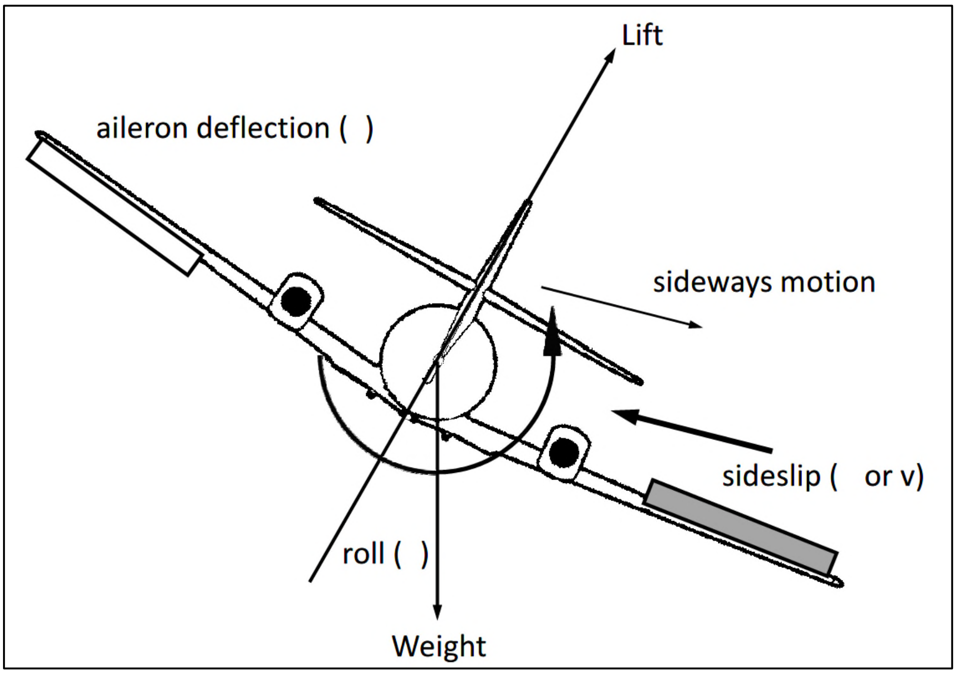


Figure 5.2 Rear view showing angle of roll

By flying a number of sideslips and recording the aileron angle ξ (and applied force, Pξ) and indicated sideslip β the trim curves of lateral static stability can be drawn for the 'controls fixed' and 'controls free' states.

The slopes of the curves indicates the degree of lateral stability; under the normal sign convention a negative slope of ξ against β, and a positive slope of Pξ against β, shows that the aircraft exhibits controls fixed and controls free static stability respectively. Note that the stability derivative, $\mathring{L}_v$ will be negative if the aircraft is stable. Alternatively the lateral static stability can be gauged directly by noting the aileron deflection direction required for a given sideslip angle: right aileron for right sideslip indicates that the aircraft is laterally stable.

5.3 Lateral/Directional Stability in Action

A sideslip will occur when the aircraft is disturbed from symmetric flight by, for example, a small angle of bank ϕ, or a lateral gust. The horizontal component of lift force will produce a lateral velocity component, or sideslip, v. If the aircraft is to regain its symmetric flight it must produce a rolling moment $\mathring{L}_v.v$ to restore the wings level state (lateral stability) and a yawing moment $\mathring{N}_v.v$ to eliminate the sideslip (directional stability).

These moments arise from the aerodynamic characteristics of the aircraft in sideslip: note that both moments will disappear when the sideslip vanishes. Ideally the aircraft will be back to wings level flight although the heading will often not equal that being flown before the disturbance.

5.4 Assessment of Lateral/Directional Static Stability

The method of assessment of lateral-directional static stability is to force the aircraft into a steady sideslip by using the aileron and rudder controls. By recording the control positions (and applied forces) along with the sideslip angle the data can be plotted to show whether the aircraft produces moments which would tend to restore it to symmetric flight.

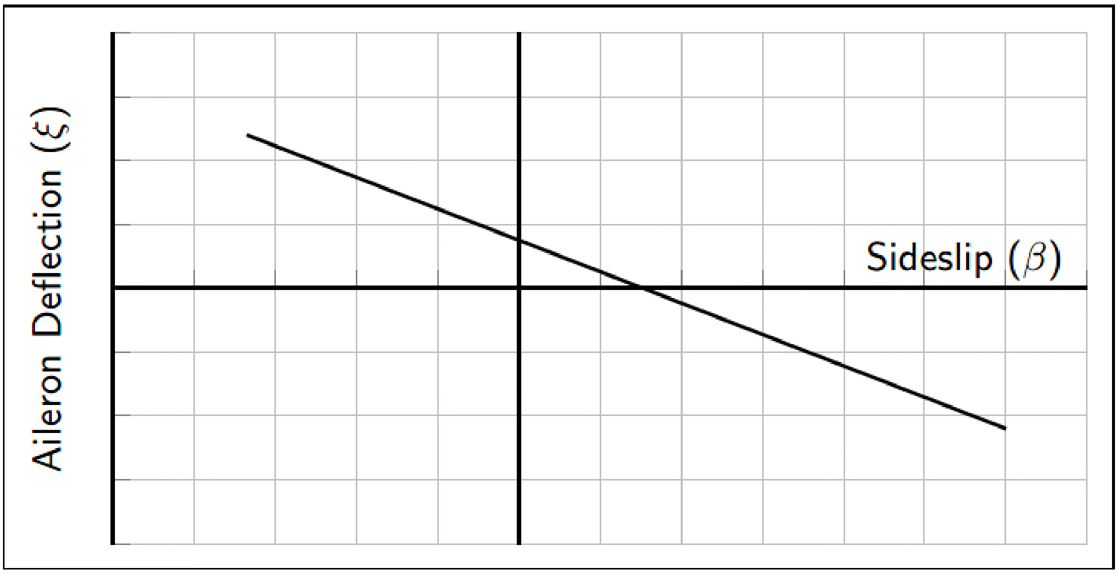


Figure 5.3 ***Controls Fixed Lateral Static Stability***

Note that the aileron and rudder controls will not produce a steady state sideslip independently; each control produces both rolling and yawing moments. In practice the lateral and directional stabilities must be considered together rather than as separate entities. The relevant airworthiness requirements, CS 25.177(c), states that:

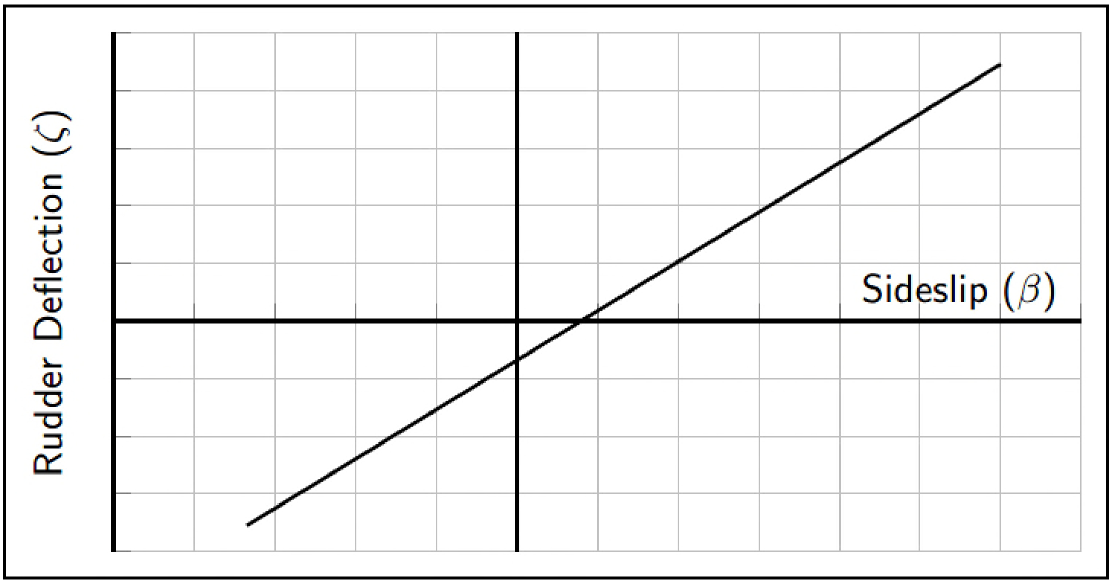


Figure 5.4 Controls Fixed Directional Static Stability

> "*In straight, steady, sideslips over the range of sideslip angles appropriate to the operation of the aeroplane, the aileron and rudder control movements and forces must be substantially proportional to the angle of sideslip in a stable sense. The factor of proportionality must lie between limits found necessary for safe operation. The range of sideslip angles evaluated must include those sideslip angles resulting from the lesser of:*
> *1. one-half of the available rudder control input; and*
> *2. a rudder control force of 801 N (180 lbf).*"

AMC 25.177(c) indicates that the limiting sideslip for the proportionality requirement is given by $\pm \sin^{-1}(30 / Vc)$. Although:

> *"Experience has also shown that a maximum sideslip angle of 15° is generally appropriate for most transport category aeroplanes even though the equation may provide a higher sideslip angle. However, limiting the maximum sideslip angle to 15 ° may not be appropriate for aeroplanes with low approach speeds or high crosswind capability."*

NOTE - Vc is *Calibrated airspeed* in knots, which can be approximated by *Equivalent airspeed* (VEAS) in this case.

Test Plan - Measurement of Lateral-Directional Static Stability

Objectives:

- To prepare trim curves (controls fixed) for the aircraft with respect to a sideslip disturbance.
- To show that the aircraft is laterally and directionally statically stable.

Method: Steady heading sideslips.

1. At a suitable height, clear of turbulence, trim the aircraft into steady, level, symmetric flight with engines matched for equal power. When steady, record values for:

    - Aileron angle ξ
    - Rudder angle ζ
    - Bank angle ϕ
    - Sideslip angle β
    - Airspeed (EAS) Ve for reference
    - Fuel quantity for reference.

2. Yaw the aircraft using rudder so that it takes up a sideslip angle; 'check' the tendency for the aircraft to roll with aileron, When steady record values for:

    - Aileron angle ξ
    - Rudder angle ζ
    - Bank angle ϕ
    - Sideslip angle β

3. Repeat for three sideslips to left and for three sideslips to right.

4. Perform a set of sideslips about another test condition/ configuration.

Steady Heading Sideslip Measurements

Datum Airspeed .... (kts) Fuel State .... (kg)

UC UP/DWN FLP ……..

Aileron angle (deg)	Rudder angle (deg)	Sideslip angle (deg)	Bank angle (deg)

Datum Airspeed .... (kts) Fuel State .... (kg)

UC UP/DWN FLP ……..

Aileron angle (deg)	Rudder angle (deg)	Sideslip angle (deg)	Bank angle (deg)

Data Analysis

1. For each configuration plot roll angle against sideslip angle and comment on the changes with airspeed.

2. For each configuration plot rudder angle and aileron angle against sideslip angle, to determine whether the aircraft is statically stable laterally and directionally and if it meets the requirements of CS 25.177(c).

3. Compare the slopes of the trim curves in each configuration and comment on the relative values of the lateral and directional static stabilities

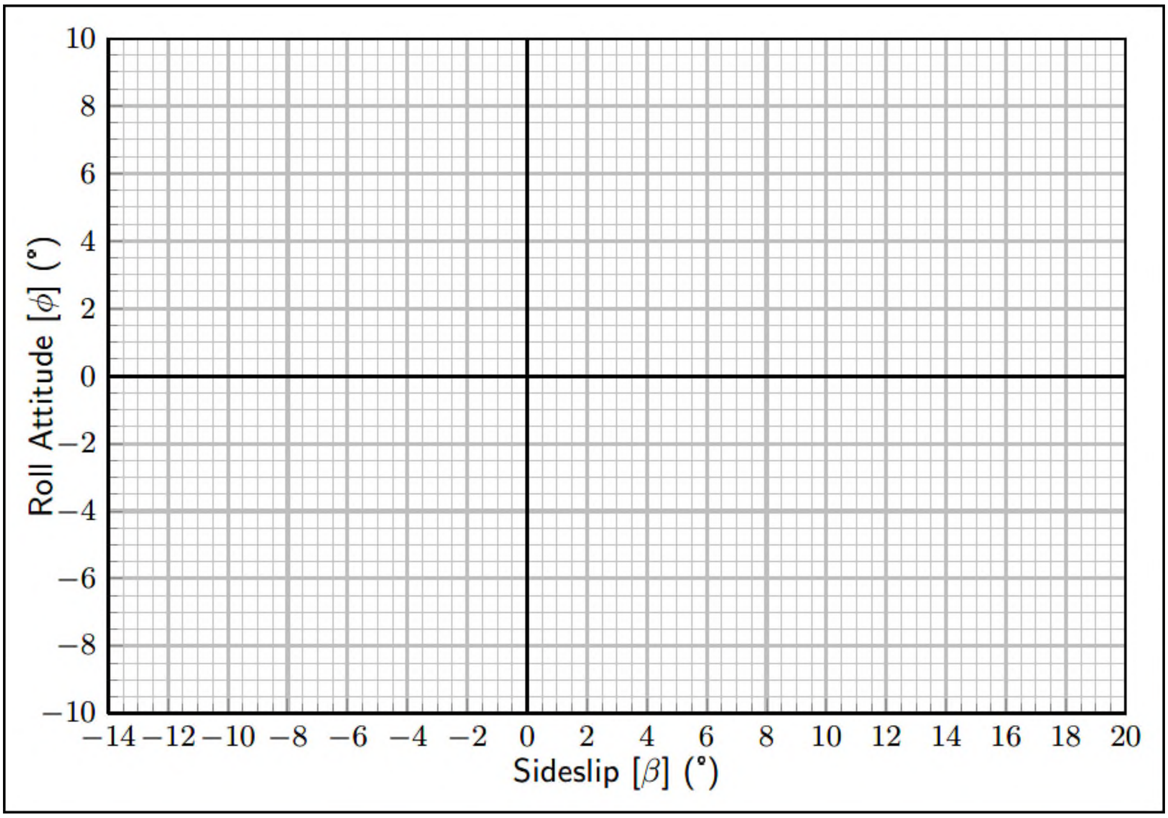


Figure 5.5 ***Sideforce Characteristics***

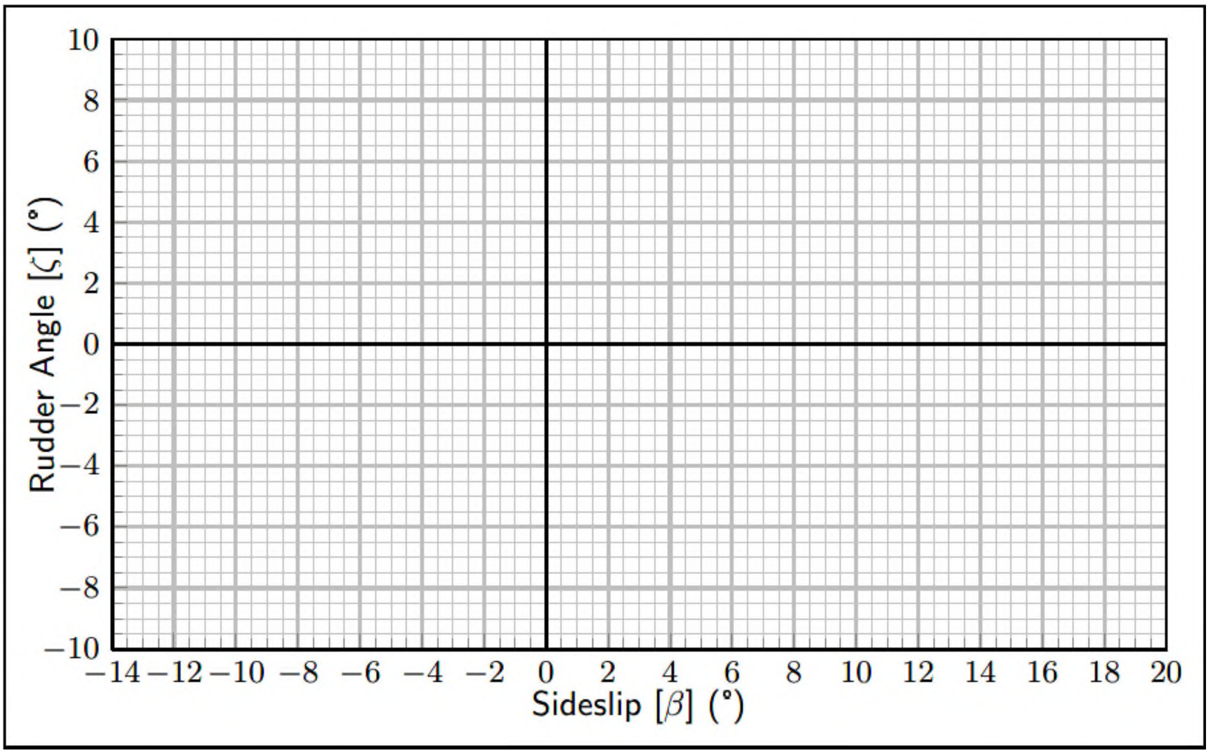


Figure 5.6 ***Directional static stability – Control Fixed***

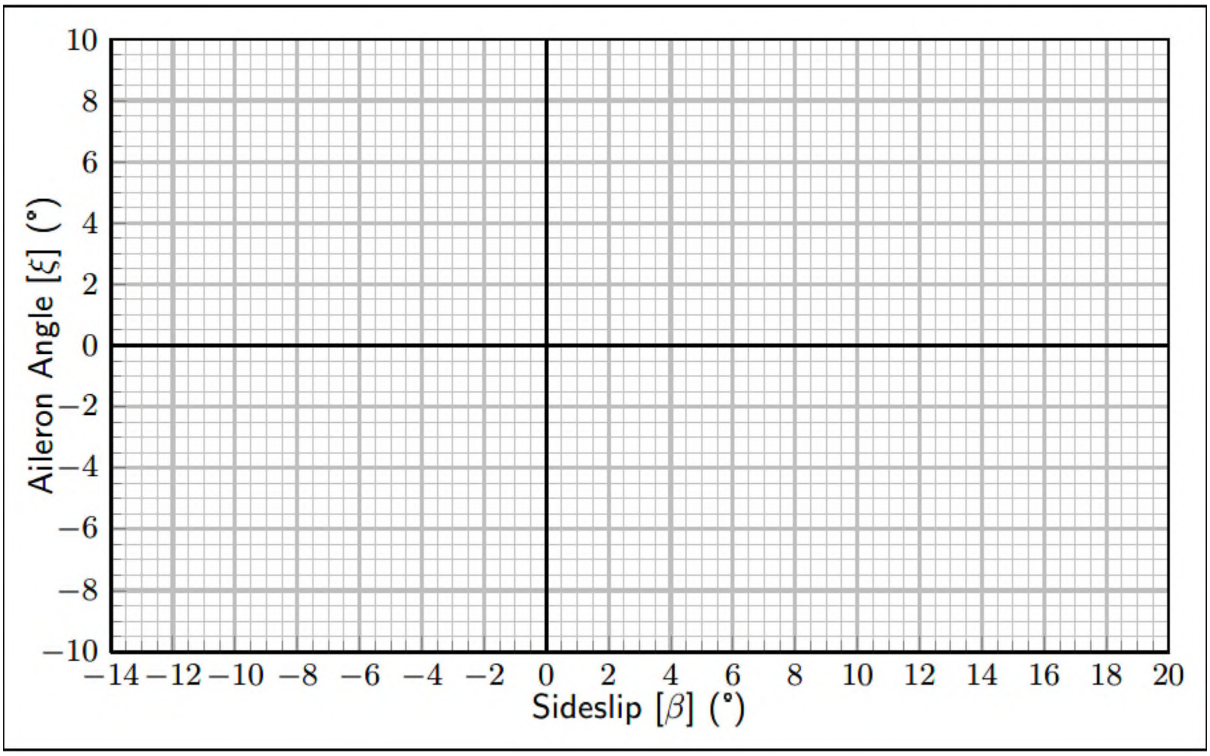


Figure 5.7 ***Lateral static stability – Control Fixed***

6. Longitudinal Dynamic Stability

6.1 Short Period Pitching Oscillation

The short period pitching oscillation is the most important dynamic mode of all and consists of a damped oscillation about the aircraft pitch axis. Whenever the aircraft is disturbed from its pitch equilibrium, the mode is excited and manifests itself as an oscillation in which the principal variables are pitch rate and angle of attack. Typically the frequency of the mode is 0.5 to 2.0 Hz, which is in the region of the natural frequency of human pilot. It is therefore essential that the short period mode be well damped, otherwise severe handling problems can arise. This requirement is encapsulated in CS 25.181 Dynamic stability which states that:

"(a) *Any short period oscillation, not including combined lateral-directional oscillations, occurring between 1.13* V_{SR} *and the maximum allowable speed appropriate to the configuration of the aeroplane must be heavily damped with the primary controls:*

> *1. Free; and*
> *2. In a fixed position."*

Flight Demonstration

The aircraft is "trimmed" to maintain constant height and speed, then the mode is excited by applying a short duration disturbance in pitch. This can be achieved by an ***impulse*** applied to the elevator. The disturbance sets the aircraft at a new angle of attack by the time that the elevator returns to its original position. A plot such as Figure 6.1 shows how quickly the aircraft short period dynamics return the aircraft back to its original angle of attack. The motion is best observed by watching the wingtip rotate relative to the horizon. This is a measure of the variation in angle of attack, since the flight path of the aircraft cannot significantly change within this short time-scale.

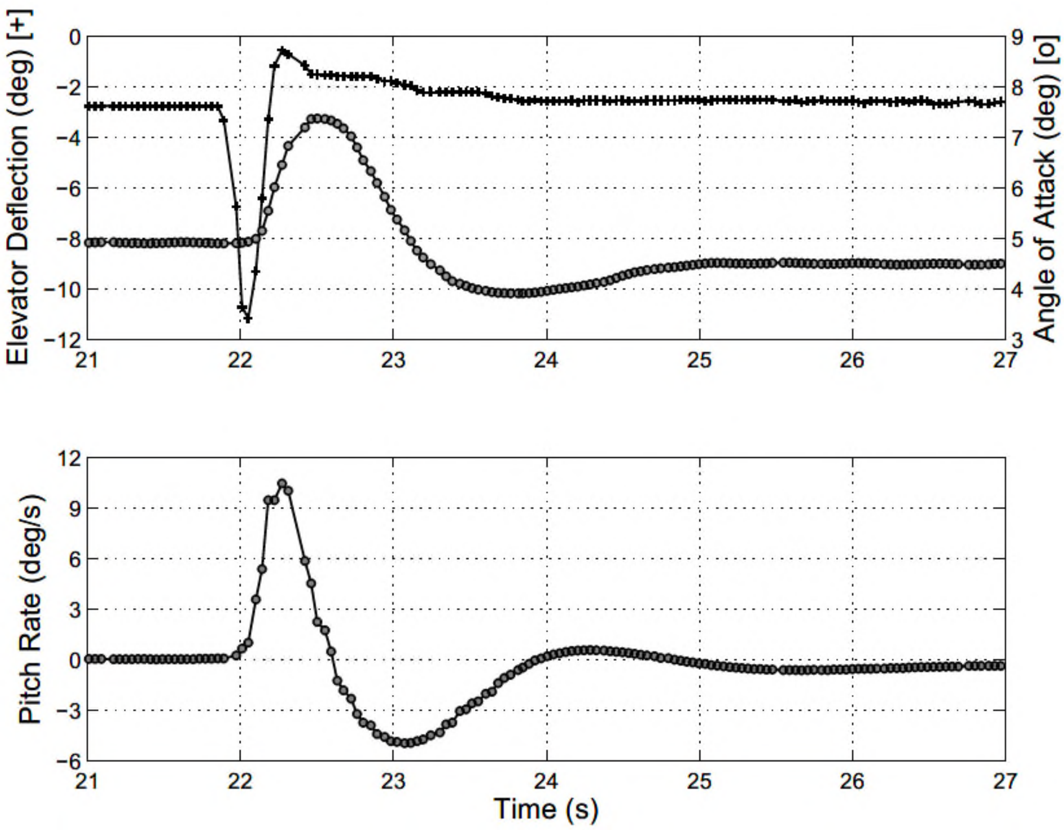


Figure 6.1: Short Period Pitch Response

Figure 6.1 shows the aircraft in a steady state condition until the elevator is moved at 21.9 seconds into the recording. By 22.4 sec the elevator is stationary again after causing the disturbance; with the aircraft now at an angle of attack increased by 2.5 deg. The subsequent response shows the short period dynamics returning the angle of attack towards its original (pre-disturbance) value.

6.2 Phugoid Oscillation

The phugoid oscillation is most commonly seen as a lightly damped low frequency oscillation in speed, which couples with height. Whenever the aircraft is disturbed from its trim speed, the mode manifests itself as a sinusoidal oscillation in which the principal variables are pitch attitude and speed change. A significant feature of the mode is that the angle of attack remains substantially constant throughout.

Thus the change in pitch attitude gives a corresponding change in the flight path angle, and hence the height excursions. The period is long, typically in the band 40 - 100 seconds, and the damping is mainly due to the rate of change of drag of the aircraft with speed change. It is possible to deduce that the period of the mode is determined almost entirely by the datum speed, and a good approximation is that the period in seconds is equal to **quarter** of the true airspeed in knots.

Since the phugoid mode has such a low frequency, it poses an undemanding task, and an average human pilot can control the mode even when it is unstable. The phugoid manifests itself as a trimming problem, which, although not regarded as hazardous when poorly damped, does contribute to an increased pilot workload.

Flight Demonstration

The aircraft is "trimmed" to maintain constant height and speed, then the phugoid mode is excited by a gentle application of the elevator to cause a change in both the pitch attitude and airspeed.

When the speed has diverged sufficiently the elevator is gently released to trim. In recovering its trimmed datum flight condition the aircraft will overshoot the trim speed and execute a phugoid oscillation exhibiting poor damping as shown in Figure 6.2. Note that application of the elevator will also excite the short period mode, but this is minimised by gentle movements of the control, and in any case any transient will have died away long before the phugoid has properly developed. The motion is best observed by assessing the pitch angle relative to the horizon while also watching your display.

Figure 6.2 shows the aircraft with the elevator pulled gently back to pitch the aircraft in a nose-up direction and reduce speed. The elevator is released at 40 seconds into the recording at which point the aircraft immediately starts to pitch in a nose-down direction. From this point on, the elevator is held fixed, and the aircraft performs the phugoid oscillation with a period of approximately 40s.

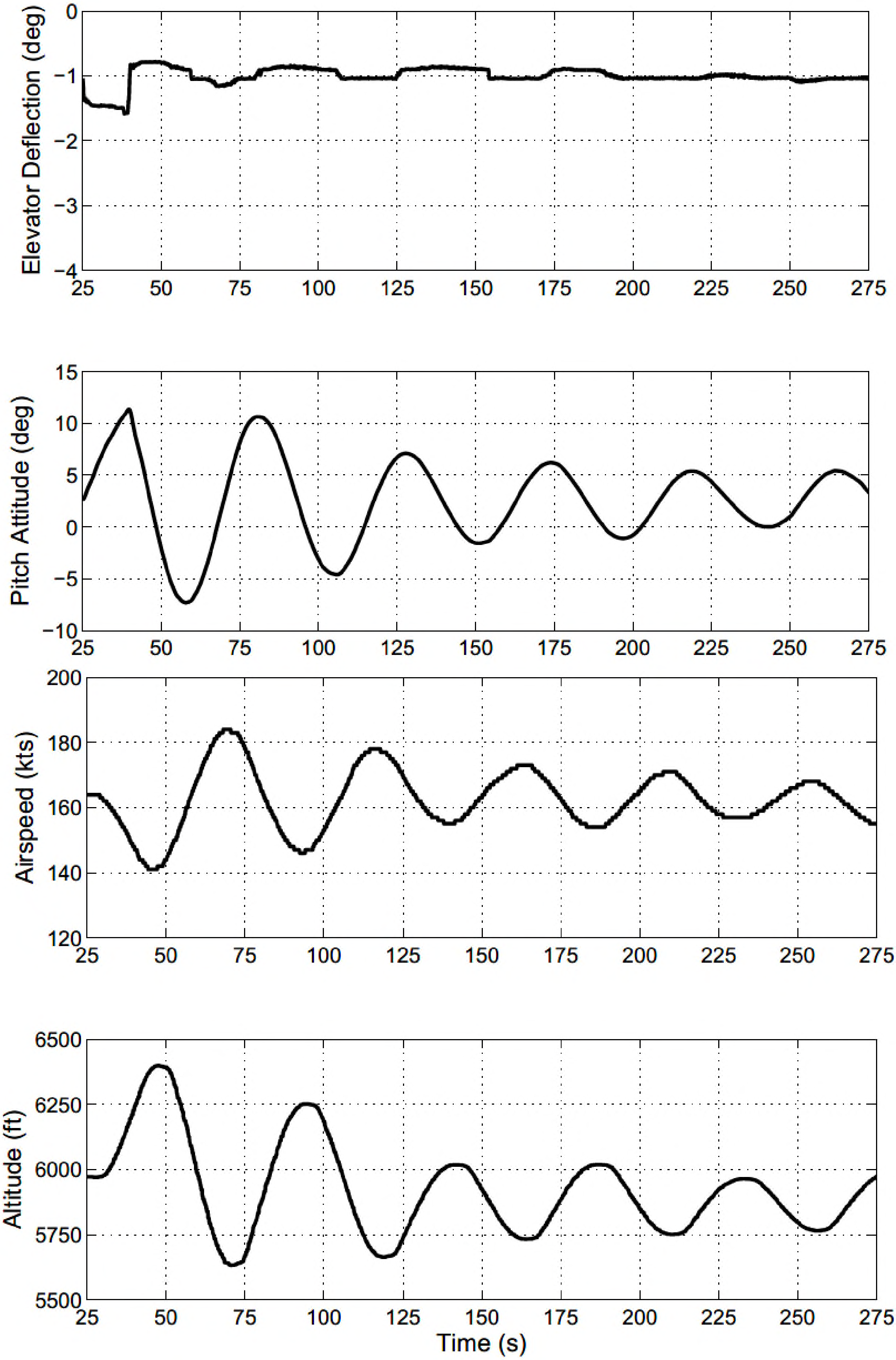


Figure 6.2: Phugoid Response

7 Lateral-Directional Dynamic Stability

7.1 Dutch Roll Mode

The Dutch roll mode is an oscillation about the aircraft yaw axis in which the principal variables are sideslip angle and yaw rate with the aircraft maintaining a straight flight path. Fundamentally, the mode is the directional equivalent of the longitudinal short period, although the damping is usually less. One characteristic of the motion is that the sinusoidal changes in sideslip cause a similar change in rolling moment (via the “dihedral effect”) and this causes the aircraft to oscillate in roll. The phase shift between cause and effect means that the forward going wing is low (and the aft going one high), so that the wing tips describe an elliptical (or possible circular) path.

The requirements on the Dutch roll mode, given in CS 25, reflect the fact that it is normally considered a nuisance rather than a handling problem:

CS 25.181(b)

“Any combined lateral-directional oscillations (‘Dutch roll’) occurring between 1.13 V_{SR} and (the) maximum allowable speed appropriate to the configuration of the aeroplane must be positively damped with controls free, and must be controllable with normal use of the primary controls without requiring exceptional pilot skill.”

Additional damping of this mode is often done electronically (using a yaw damper) rather than aerodynamically.

Flight Demonstration

The aircraft is set up in straight, wings level flight, and then the Dutch roll mode is excited by generating a disturbance in yaw via a sinusoidal displacement of the rudder at a frequency approximately equal to that of the mode. The input provides an approximately symmetrical displacement in both sideslip angle and yaw rate.

The motion is best observed by watching the motion of the wingtip relative to a fixed point on the horizon. Since the flight path remains straight, the fore and aft motion of the wingtip gives an indication of the amount of sideslip, and the up and down motion indicates the amount of roll angle that happens as a result of the oscillation. Figure 7.1 shows the aircraft in straight flight with the rudder moving left and right at the same frequency as the Dutch roll. Notice how the response increases in magnitude until the rudder is released at 34 seconds. The resulting oscillatory response indicates the damping of the Dutch roll.

7.2 Roll (Subsidence) Mode

The roll (subsidence) mode is an exponential change in roll rate about the aircraft roll axis, and is non-oscillatory. This means that when the aircraft is disturbed in roll, it will acquire a new roll rate exponentially: consequently all rolling motion (especially aileron response) has an exponential lag associated with it. The mode characteristic is almost entirely due to the viscous 'paddle' damping effect of the wing when the aircraft is disturbed in roll. It is thus always present and is always stabilising in its effect. Because of its origin, the mode is sometimes referred to as the "roll damping" mode.

This mode appears to the pilot as a lag in the roll control, and if the time constant (τ_r) is too large then the control becomes 'sluggish'. The time constant for the aircraft is small enough that the lag is almost imperceptible to the pilot. Larger aircraft will typically have a longer time constant. [Mil Spec MIL-F-8785C quotes a maximum time constant of 1.4 seconds for this mode.]

Flight Demonstration

The aircraft is set up with typically 20° of bank, and then the mode is excited by applying a small step displacement to the aileron. When the aileron is deflected the aircraft rapidly acquires a steady roll rate after a short exponential rise, the exponential being determined by the mode time constant. When the bank angle has reached approximately 30° in the opposite direction the aileron is centred to stop the rolling. A larger step displacement is then made in the opposite direction.

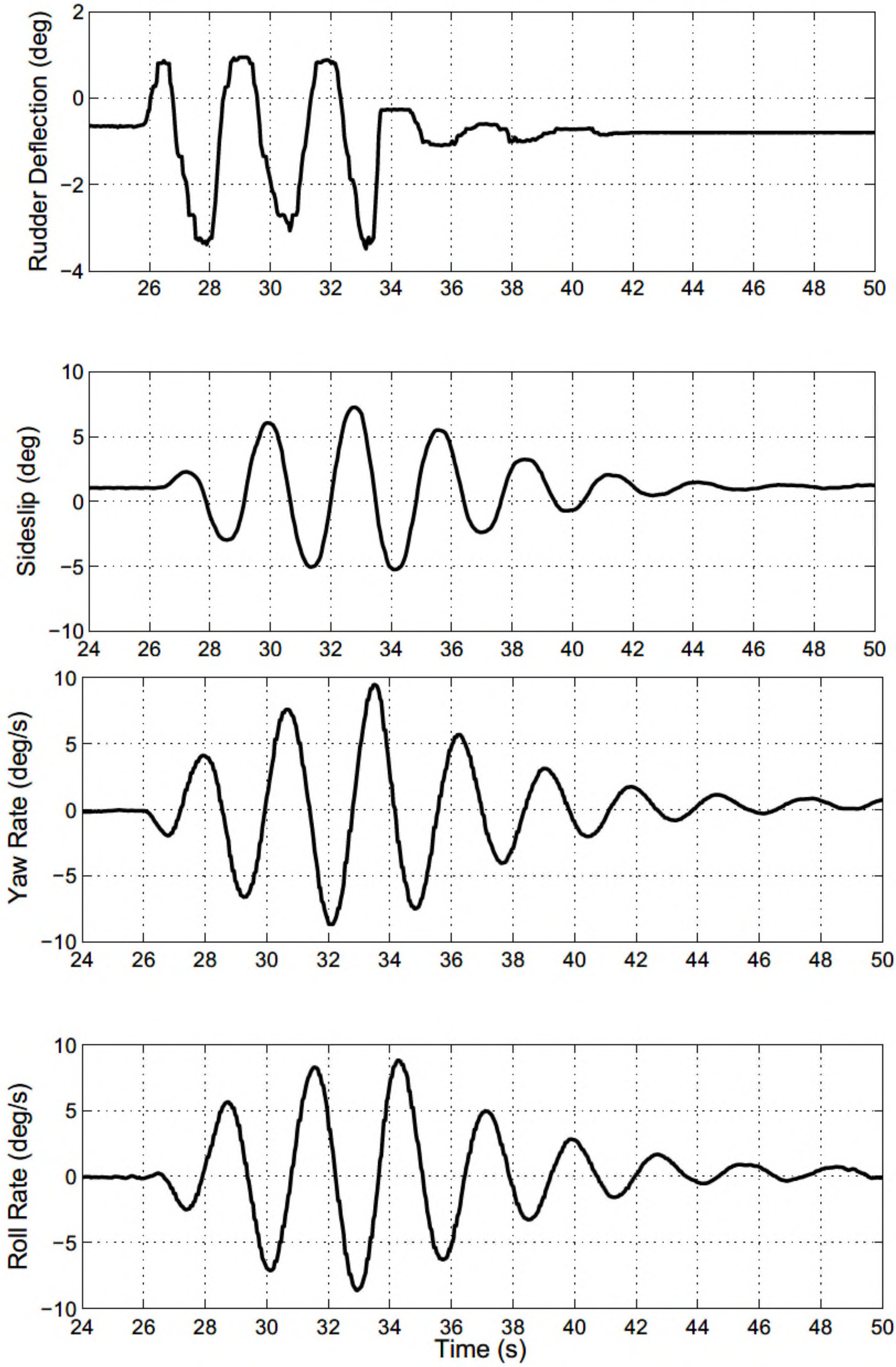


Figure 7.1 Dutch Roll Response

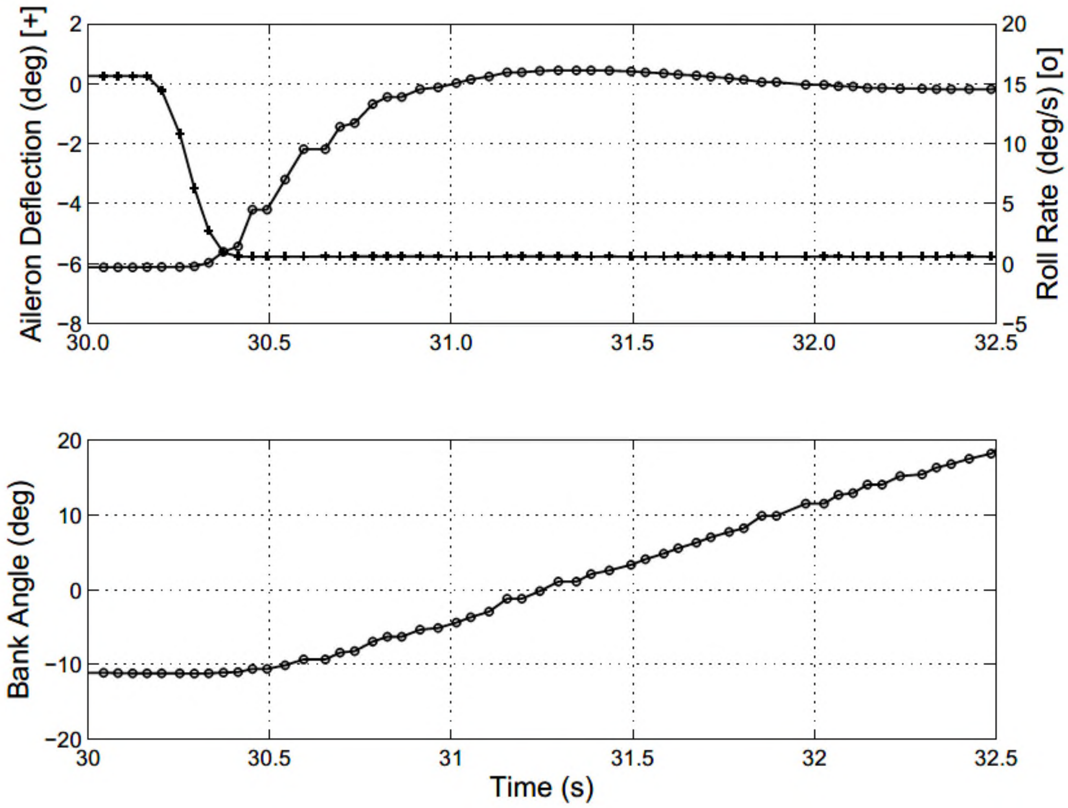


Figure 7.2 Roll Mode Response

At each aileron step displacement, the roll rate changes exponentially to a steady value. The mode is best observed by watching the wing tip for the aileron movement, and attempting to identify the (small) delay in the aircraft response. Figure 7.2 shows the aircraft at zero roll rate until the aileron step at 30.2s; at which point the aileron angle is rapidly changed to about -6.0°.

Following the aileron step, the exponential rise of the roll rate occurs between 30.4 and 31.2 seconds, and the time constant of this exponential represents the mode.

7.3 Spiral Mode

The spiral mode is non-oscillatory and manifests itself as an exponential convergence or divergence in roll attitude, which produces, when the mode is unstable, a divergent spiral descent. As the time constant of the mode is relatively large, typically 40 seconds or more, the mode is slow to develop.

Physically, when the roll attitude of an aircraft is disturbed, the lift vector will also rotate which has the potential to cause a small sideslip. If the sideslip is in the direction of the roll, any dihedral effect will produce a moment in a direction that will reduce the bank angle, but the fin will produce a moment that will yaw the aircraft in the same direction as the roll. All of these moments will be small, hence the long-time constant, and the stability of the mode depends primarily on the relative magnitudes of the dihedral effect of the wing and the “weathercock” effect of the fin. Note that it is the fin that causes the nose of the aircraft to point below the horizon, and changes an unstable rolling motion into a spiral dive.

The spiral mode, being a long period mode, does not influence short term handling significantly, and when it is stable poses no further problem. When it is unstable the aircraft slowly diverges from straight and level flight if left to its own devices. This can become hazardous when external visual cues are poor or absent, as the mode has virtually imperceptible motion cues associated with it. Consequently the pilot can become disoriented quite easily

Flight Demonstration

The aircraft is set up in a co-ordinated turn at 20° angle of bank, and the controls are released. If the spiral mode is neutral, there will be no change in bank angle. However, when the mode is unstable the aircraft will diverge exponentially, with the fin keeping the sideslip to a low level and therefore the nose of the aircraft will fall increasingly further below the horizon, resulting in the classical spiral descent if left unchecked.

Figure 7.3 shows the aircraft is initially set in a left bank at an angle of 20°, at 140 knots and at 6100 ft. The controls were released at the beginning of the plot, and the unstable spiral mode is indicated by the slowly increasing bank angle. After 55 seconds the bank angle is approaching 50°, but the progress of this increase has been influenced by some turbulence; a further indicator that the destabilizing moment is small. Although not contributing to the mode, the increase in speed and reduction in height are also shown.

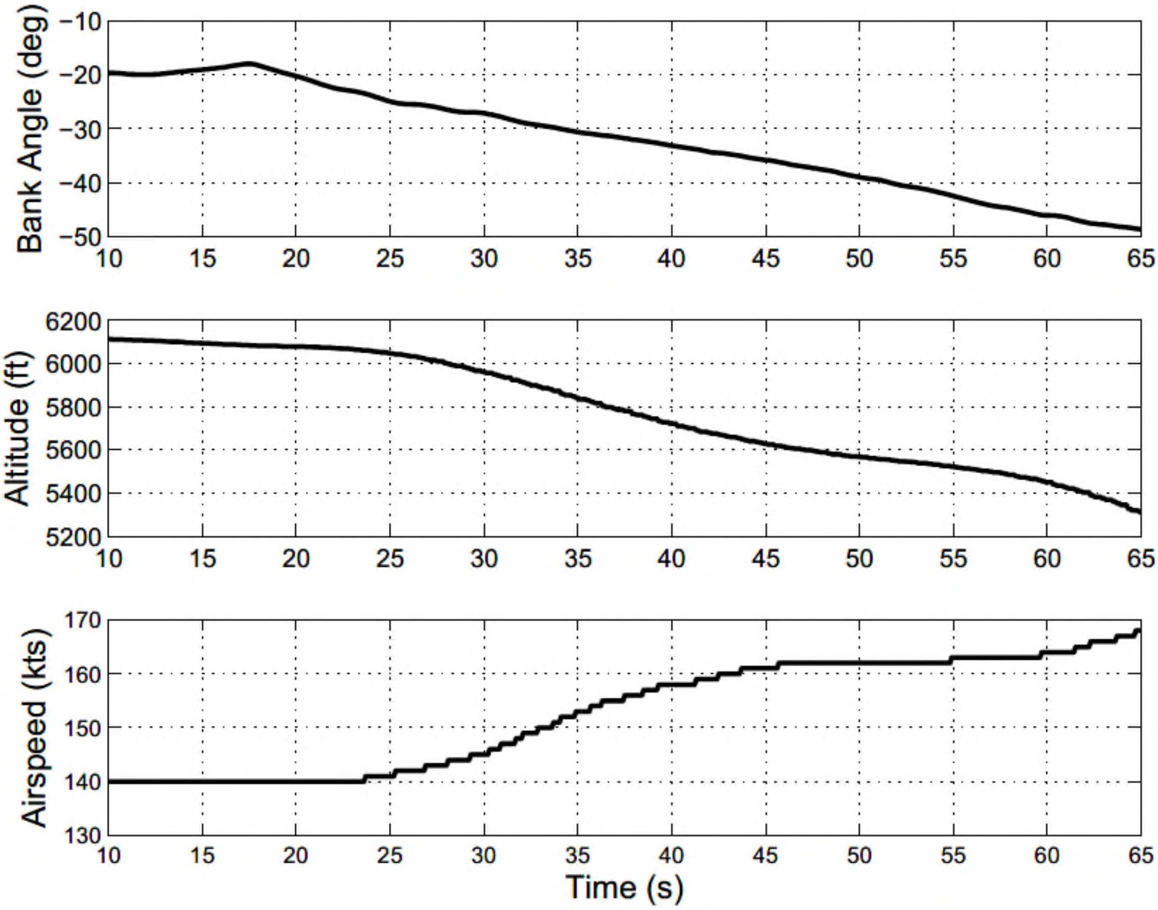


Figure 7.3 Spiral Mode Response

8. Stall Warning and Stall Protection Systems

As airspeed decreases and level flight is maintained, angle of attack increases until the pressure distribution over the aerofoil cannot be sustained and the wing stalls. The stall fixes the maximum angle of attack which determines the minimum speed boundary of the aircraft. It also determines the point at which control of flight path is lost, consequently the pilot must be made aware that the stall boundary is being approached. The applicable airworthiness requirement [**CS 25.207**] states:

"(a) *Stall warning with sufficient margin to prevent inadvertent stalling with the flaps and landing gear in any normal position must be clear and distinctive to the pilot in straight and turning flight*."

Since the inherent aerodynamic qualities of the Saab 340 do not give a clear and distinctive warning to the pilot as required by CS 25.201(d) the aircraft must be fitted with an artificial stall warning and protection system. There is a "stick shaker", which indicates that the stall angle of attack is being approached, and a "stick pusher" which lowers the nose of the aircraft to rapidly reduce the angle of attack before the stall is reached.

Transducers are fitted to the wing which sense movement of the stagnation point around the leading edge of the wing; this is related to the angle of attack. When the angle of attack reaches a pre-set figure on either wing the system alerts the pilot by an aural warning horn and by operation of a stick shaker. This operates at an incidence sufficiently below the stalling angle of attack to give the pilot an adequate margin for recovery action. If the angle of attack reaches a second pre-set value (this time sensed by both wings to prevent nuisance warnings) the stick pusher operates; this is an hydraulic ram which pushes the control column forward to lower the nose and prevent the aircraft from stalling.

Flight Demonstration

Providing good conditions, the aircraft will be trimmed in steady flight level flight at a safe altitude, well clear of cloud, and at about 1.3 times the expected stall warning speed. The power is reduced and the airspeed allowed to decay at about 1 kt/sec. Note the airspeeds at which the stall protection system sounds the warning horn, starts the stick shaker and operates the stick pusher.

Stall Warning speed ………. **Stick Push speed…………..**

Appendices

A Useful Constants

Saab 340B gross wing area (Sw)	41.8	m^2
Saab 340B standard mean chord (c)	2.08	m
Saab 340B empty mass (with pilots)	8695	kg
Acceleration due to gravity (g_0)	9.80665	m / s^2
ISA Density at sea level (ρ_0)	1.225	kg / m^3
ISA Pressure at sea level (P_0)	101,325	Pa or N / m^2
ISA Temperature at sea level (T_0)	288.15	K
To convert kts into m/s, multiply by:	0.51444	
To convert feet into metres, multiply by:	0.3048	

B Useful Relationships

B.1 Lift Coefficient

$$L = mg_0 - (T_L + T_R)\sin\theta \quad \text{and} \quad C_L = \frac{2L}{\rho_0 V_e^2 S_w} \qquad \text{(B.1)}$$

where m is the aircraft mass, T_L and T_R are the propeller thrusts, g_0, ρ_0 and Sw are defined above and Ve is the equivalent airspeed in m/s.

B.2 Drag Coefficient

$$D = (T_L + T_R)\cos\theta \quad \text{and} \quad C_D = \frac{2D}{\rho_0 V_e^2 S_w} \qquad \text{(B.2)}$$

where D is the total drag force acting on the aircraft in N: assumed to equal the horizontal component of the total thrust produced by the propellers when the aircraft is in non-accelerating level flight.

B.3 True Airspeed

$$V \equiv V_t = V_e\sqrt{\frac{\theta}{\delta}} \quad \text{Or} \quad V_t = \frac{V_e}{\sqrt{\sigma}} \qquad \text{(B.3)}$$

$\theta = T / T_0$ and δ (P / P_0), is read at the appropriate altitude from the ISA table. T_0 is defined above and σ is the relative density.

ISA values for δ (relative pressure) = $\{1 - (6.8756 \times 10^{-6}.H_p)\}^{5.2559}$

H_p (ft)	δ	H_p (ft)	δ	H_p (ft)	δ
0	1.00000	6600	0.78342	13200	0.60649
200	0.99279	6800	0.77751	13400	0.60169
400	0.98563	7000	0.77163	13600	0.59691
600	0.97851	7200	0.76579	13800	0.59217
800	0.97143	7400	0.75998	14000	0.58745
1000	0.96439	7600	0.75421	14200	0.58277
1200	0.95739	7800	0.74848	14400	0.57812
1400	0.95043	8000	0.74278	14600	0.57350
1600	0.94352	8200	0.73712	14800	0.56890
1800	0.93664	8400	0.73149	15000	0.56434
2000	0.92981	8600	0.72590	15200	0.55981
2200	0.92302	8800	0.72034	15400	0.55530
2400	0.91626	9000	0.71481	15600	0.55083
2600	0.90955	9200	0.70932	15800	0.54639
2800	0.90287	9400	0.70387	16000	0.54197
3000	0.89624	9600	0.69845	16200	0.53758
3200	0.88965	9800	0.69306	16400	0.53322
3400	0.88309	10000	0.68770	16600	0.52890
3600	0.87658	10200	0.68238	16800	0.52459
3800	0.87010	10400	0.67710	17000	0.52032
4000	0.86366	10600	0.67184	17200	0.51608
4200	0.85726	10800	0.66662	17400	0.51186
4400	0.85090	11000	0.66143	17600	0.50767
4600	0.84458	11200	0.65628	17800	0.50351
4800	0.83829	11400	0.65115	18000	0.49938
5000	0.83205	11600	0.64606	18200	0.49527
5200	0.82584	11800	0.64101	18400	0.49120
5400	0.81967	12000	0.63598	18600	0.48715
5600	0.81353	12200	0.63099	18800	0.48312
5800	0.80744	12400	0.62602	19000	0.47913
6000	0.80138	12600	0.62109	19200	0.47516
6200	0.79535	12800	0.61619	19400	0.47121
6400	0.78937	13000	0.61133	19600	0.46730

B.4 Worked Examples

Lift Coefficient

$$C_L = \frac{2mg_0}{\rho_0 V_e^2 S_w} = \frac{2\ x\ m\ x\ 9.80665}{1.225\ x\ V_e^2\ x\ 41.8} = 0.3830\frac{m}{V_e^2}$$

Now V_e equals the *equivalent* airspeed in m/s that is $0.51444V_e$ where V_e is KEAS and so:

$$C_L = 0.3830\ x\ \frac{m}{(0.5144\ x\ V_e)^2} = \ 1.447\frac{m}{V_e^2} \qquad \text{(B.4)}$$

where m is the mean aircraft mass at the time of the test, taken from the load sheet, and V_e is the aircraft speed recorded as part of the test data (KEAS).

Drag Coefficient

$$C_D = \frac{2D}{\rho_0 V_e^2 S_w} = \frac{2\ x\ T}{1.225\ x\ V_e^2\ x\ 41.8} = 0.039\frac{T}{V_e^2}$$

Likewise:

$$C_D = 0.039\ x\ \frac{T}{(0.5144\ x\ V_e)^2} = \ 0.147\frac{T}{V_e^2} \qquad \text{(B.5)}$$

where the total thrust produced by the powerplants $T = T_L + T_R$ if the thrust data is given in N. V_e is in knots.

C Safety Brief

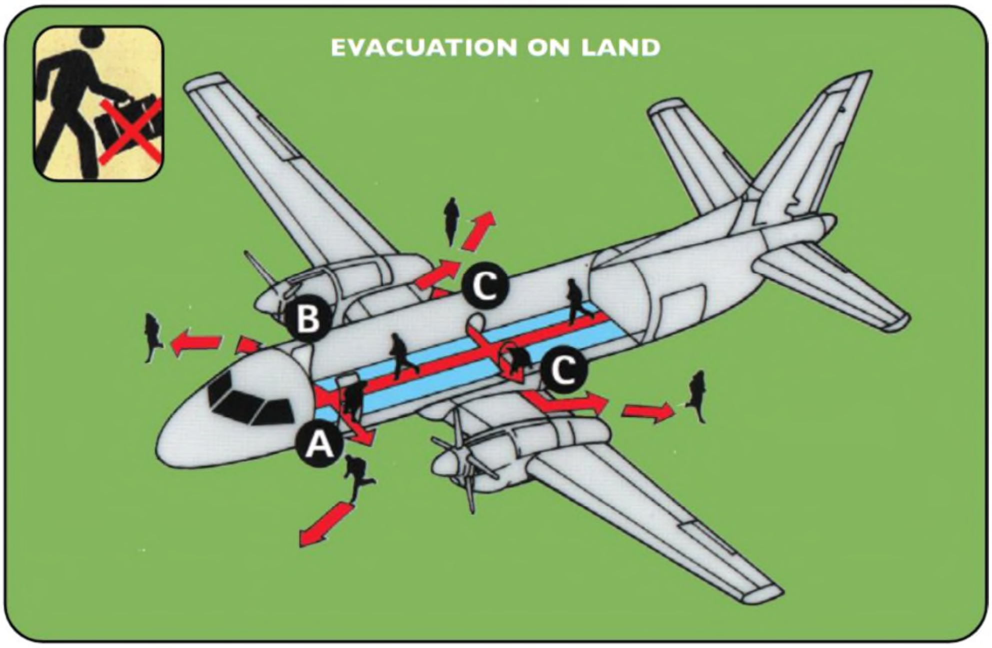
C.1 Conduct in and around the aircraft

When moving in the vicinity of the aircraft it is imperative that students note the following:

- You may be searched prior to boarding: it is your responsibility to ensure that you do not take any ‘dangerous’ articles on board.
- You will be escorted as a group to and from the aircraft, so do not wander away from your group.
- Follow the ‘escort’ - do not be tempted to cut corners as there are often quite rigid rules about where you can and cannot walk when you are ’airside’.
- Smoking is not allowed ‘airside’, and is prohibited on the aircraft.
- On entering the aircraft, please respect the seating plan for your group - we do not issue boarding cards with seat numbers, but some seats must be left empty to satisfy the CG requirements.
- Fasten your seatbelt and keep it fastened throughout the flight.
- Pay attention to the safety briefing on every flight.
- Take note of your position in relation to the emergency exits which are marked on the passenger briefing card.
- Airsickness bags are provided, if you do feel unwell make sure that a bag is readily to hand, and remove it from the aircraft after use.
- All electronic equipment can generate interference to aircraft systems, and should not normally be used on the aircraft. Mobile phones must be switched to “Flight Mode”, other electronic devices such as cameras may be acceptable, however, check with a crew member before use.

Saab340B Safety Instructions Card

Please study and leave on the aircaft



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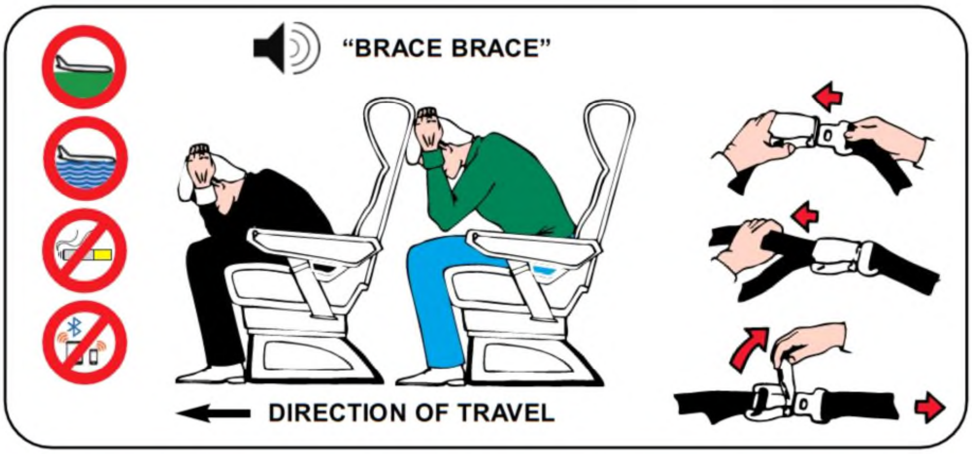

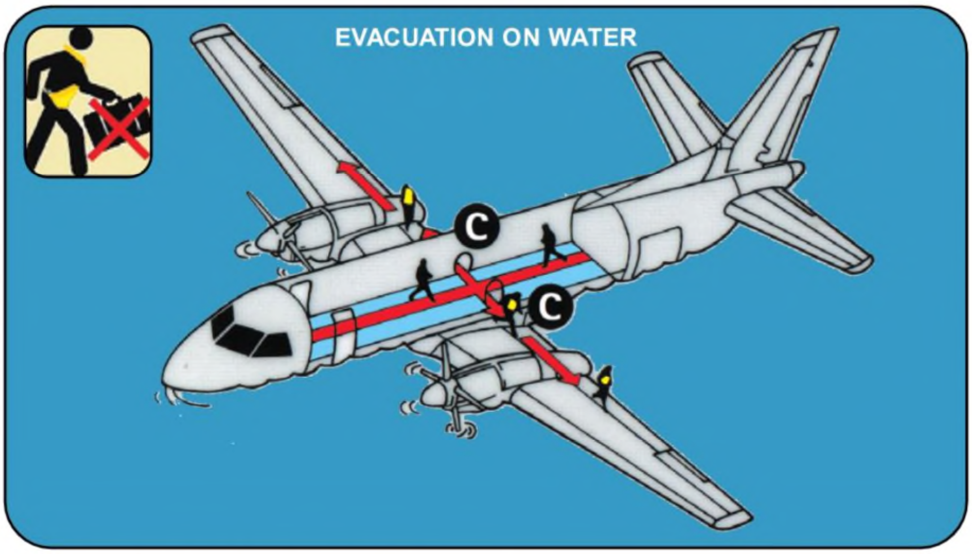

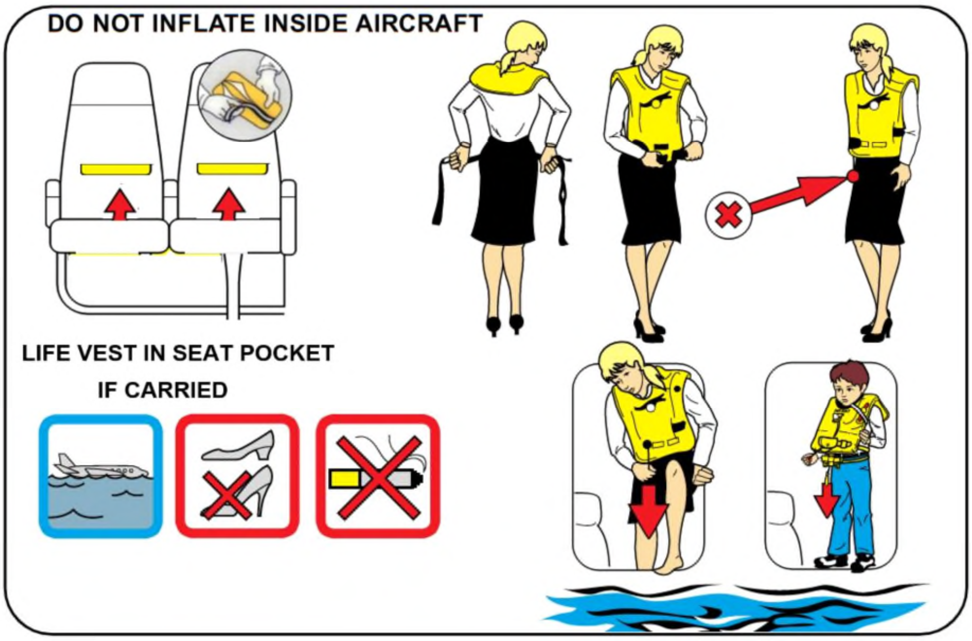

C.2 Dangerous Articles - compulsory statement

"*For safety reasons, the following articles must not be carried on board:*

- *Animals and Livestock*
- *Compressed gases, (deeply refrigerated, flammable, non-flammable and poisonous) such as butane, oxygen, liquid nitrogen, aqualung cylinders and compressed gas cylinders*
- *Corrosives, such as acids, alkalis, wet cell batteries, and items containing mercury*
- *Electronic Items, without the permission of the crew*
- *Explosives, munitions, ammunitions, including blank cartridges, fireworks, flares, hand guns and pistol caps*
- *Flammable liquids and solids, such as lighters that need inverting before ignition, lighter fuel, matches (safety matches may be carried on-board), fire-lighters, paints and thinners*
- *Radioactive materials*
- *Oxidising materials, such as bleaching powder and peroxides;*
- *Poisons and infectious substances, such as insecticides, weed killers and live virus material*
- *Other dangerous articles, such as magnetised material, offensive or irritating materials*

Medicines and toiletries may be carried in limited quantities which may be required during the journey"

NFLC operates an open and just reporting culture. If you have any concerns regards the safe operation of your flights please feel free to contact us on:

nflc.safety@cranfield.ac.uk