AP3456
The Central Flying School (CFS) Manual of Flying

Volume 2 – Aircraft Performance

CHAPTER 1 - THE EFFECT OF VARIATION IN SPECIFIC GRAVITY OF FUELS ON RANGE AND ENDURANCE

Introduction

1. Fuels in current Service use are permitted to have a wide variation in specific gravity (SG) in order to allow the oil industry to obtain fuel from many sources. However, it is emphasized that the industry makes every effort to maintain the 'normal' SG of fuel supplied and values of SG at the wider limits of the permitted range are seldom encountered. Variations in SG can have a significant effect on the range and endurance of pure jet and turboprop engines, but are normally disregarded in the case of piston engines. This chapter discusses jet engine fuels only.

2. The range and endurance per gallon of fuel depends on the number of heat units (kilojoules) per gallon of the fuel used. The number of kilojoules per kilogram of all approved fuels does not vary by more than 3% and for practical purposes may be considered a constant. Thus the output of a fuel may be considered in relationship to its SG, which depends on:

a. The basic SG at normal temperature (15 °C) of the fuel used.

b. The fuel temperature in the aircraft's tanks at the time.

3. The performance tables in Aircrew Manuals and Operating Data Manuals (ODMs) of all aircraft are based on a specified fuel of 'normal' SG, unless otherwise indicated. Aircrew Manuals also give a list of fuels approved for use in the aircraft and the reference numbers and NATO code numbers. Similarly, tank capacities, in kg of fuel, are usually shown for both AVTUR and AVTAG. Where the SG of the fuel in use varies substantially from that of the reference fuel, range corrections may be necessary.

Approved Fuels for Jet Engines

4. Information on fuels approved for in-Service use in a particular aircraft type should be obtained from the 'Release to Service' document, Aircrew Manuals or Flight Reference Cards (see also Volume 3, Chapter 19). The following are available for use in jet engines.

a. **AVTUR** - an aviation kerosene having a 'normal' SG of 0.8 at 15 °C, but a permitted range between 0.775 and 0.825, at 15 °C.

b. **AVTAG** - a wide-cut gasoline fuel having a 'normal' SG of 0.770 at 15 °C, but a permitted range between 0.751 and 0.802, at 15 °C.

c. **AVCAT** - a kerosene-type fuel similar to AVTUR but with a higher flash-point, normally used by RN carrier-borne aircraft. AVCAT has a 'normal' SG of 0.817 at 15 °C, but a permitted range between 0.788 and 0.845, at 15 °C.

d. **AVGAS** - a quality gasoline designed for piston engines. The standard RAF fuel is known as 100/130 grade and has a SG of 0.69. Use of AVGAS in jet engines has been approved in emergency only and such clearance is shown in Aircrew Manuals.

Range of Specific Gravity

5. The extreme range of SG of those fuels which could be used in jet engines varies between 0.69 (100/130 grade) and 0.845 (upper limit of AVCAT), at 15 °C. However, the range of SG of fuels normally used by RAF aircraft varies between 0.751 (lower limit of AVTAG) and 0.825 (upper limit of AVTUR), at 15 °C, but variations of temperature between +40 °C and –30 °C have the effect of increasing the extremes to 0.732 (AVTAG with basic SG of 0.751, at +40 °C) and 0.856 (AVTUR with basic SG of 0.825, at –30 °C).

Flight Planning

6. As stated earlier, every effort is made to maintain the SG of fuels at the 'normal' values, i.e. 0.8 and 0.77 for AVTUR and AVTAG respectively. Flight planning data and performance tables in Aircrew Manuals and ODMs are based on AVTUR at the 'normal' SG for temperate conditions, unless otherwise stated. To avoid ambiguity, the SG of the reference fuel is usually given where this differs from 'normal' values, eg "based on AVTUR SG 0.78". Therefore, corrections to the flight planning data are not normally required when making approximate computations of range in temperate conditions.

7. The most important aspect of the variation of SG of fuels is that of safety, ie the prime concern is the possibility of failing to achieve the range anticipated. It will be appreciated that variations in SG are most important when considering an aircraft where tanks are normally filled to capacity, ie when the volume of fuel is a fixed amount and only the SG, and thus the weight, of fuel varies. In cases where the aircraft's tanks are not filled to capacity, i.e. where fuel load is offset against payload, the weight of fuel required for a given range is virtually constant (see para 2) and variations in SG merely result in variation of the volume of fuel required; hence range corrections are unnecessary. The following paragraphs should be read in this light.

8. In temperate conditions, the use of AVTUR with flight planning data based on 'normal' AVTAG will result in better range and endurance than that deduced from the performance data; the increased performance being related directly to the variation in SG between the fuel used and the reference fuel.

9. When using AVTAG with flight planning data based on 'normal' AVTUR, in temperate conditions, in the interests of safety a reduction of 7% should be made in the range and endurance derived from such data if the SG is known to be at the low limit of AVTAG, ie 0.751. If the SG is known to be between 0.76 and 0.785, a 5% deduction should be made. The above factors are useful 'rules of thumb' and provide adequate safety margins. If the precise SG of fuel used is known, then the variations in range and endurance may be determined by equating the SG of the fuel used to that of the reference fuel, using the formula given at para 12.

10. If AVCAT has to be used (see para 4c), in temperate conditions, an increase in range and endurance will result from the use of flight planning data based on AVTAG. The same will apply in the case of AVTUR except when the AVCAT is known to have a SG of less than 0.8 ('normal' AVTUR) and is to be used with data based on 'normal' AVTUR. However, in this case the reduction in range would be about 2%.

11. If, in an emergency, it is necessary to use AVGAS 100/130 grade (see para 4d), with data based on 'normal' AVTUR, a reduction in range of about 22% should be allowed for; or 16% if using 'normal' AVTAG data. It will be appreciated that with most aircraft the emergency use of AVGAS carries other severe restrictions, particularly on the maximum height to fly, and the reduction throughout the whole range of the aircraft's performance must be considered most carefully.

Computing the Variation in Range

12. When the SG of the fuel used differs considerably from the SG of the reference fuel on which the planning data is based, a computation of the range loss or gain can be calculated from the formula:

$$R = \frac{t - r}{r} \times \frac{FD}{C}$$

where:

R	=	Range loss or gain in nautical miles
t	=	SG of the fuel used
r	=	SG of the reference fuel, ie the SG used for performance data calculations (either given specifically or 'normal' values, ie AVTUR = 0.8 or AVTAG = 0.77)
F	=	Total fuel available in kilograms weight (kg)
D	=	Distance covered on cruise in nautical miles (as found from the Aircrew Manual performance charts).
C	=	Fuel used for cruise in kilograms weight (as found from the Aircrew Manual performance charts).

An examination of this formula shows that the term $\frac{D}{C}$ gives the fuel consumption in kg/nm as defined in the Aircrew Manual. If this is then multiplied by the total fuel available for the cruise (F) then we have an expression for the range under reference conditions. The term $\frac{t-r}{r}$ is the factor by which the SG of the fuel used differs from the SG of the reference fuel.

Example:

From the ODM, an aircraft uses 470 kg of fuel (C) for a cruise distance of 750 nm (D). Consumption is therefore

$$\frac{D}{C} = \frac{750}{470} = 1.6 \text{ kg/nm.}$$

If the total fuel available for the cruise is 1,200 kg (F), then the reference range is 1,200 × 1.6 = 1,920 nm.

If the performance data was calculated using a fuel with an SG of 0.8 (r), but the SG of the actual fuel to be used is 0.78 (t), then

$$\frac{t-r}{r} = \frac{0.78 - 0.8}{0.8} = -0.025$$

and R = –0.025 × 1,920 = – 48 nm

This means that using a fuel with an SG of 0.78 instead of 0.8 results in a reduction in the range achievable of 48 nm. In general, if (t – r) is negative, ie if the fuel used has a lower SG than the reference fuel, then there will be a reduction in range. Conversely, using a fuel with a higher SG than the reference fuel will give an increase in range.

13. **Fuel Required**. To determine the quantity of fuel of known SG needed to achieve a given range, multiply the quantity of the reference fuel required by $\frac{r}{t}$

Example:

From the ODM, an aircraft uses 470 kg of fuel with an SG of 0.8 for a cruise distance of 750 nm. If the fuel to be used has an SG of 0.78, then the amount of fuel needed will be:

$$470 \times \frac{0.8}{0.78} = 482 \text{ kg}$$

14. **Contents Gauges**. Fuel contents gauges are normally calibrated in kilograms, but actually measure the amount of fuel by volume, i.e. litres. Therefore, the gauge will only show the correct weight of fuel in the tanks when it is calibrated to the same SG as the fuel in use. In some aircraft, such as the VC10, fuel gauges are compensated for SG.

15. Fuel Gauge Readings. In order to calculate the fuel gauge reading when the fuel in the tanks has a different SG to the fuel for which the gauges are calibrated, the following formula may be used:

$$\text{Fuel gauge reading (kg)} = \text{Fuel in tanks (kg)} \times \frac{g}{t}$$

where:

g = SG of fuel for which the gauge is calibrated
t = SG of fuel in tanks

Example:

Weight of fuel in tanks	= 8,000 kg
SG of fuel for which the gauges are calibrated	= 0.78
SG of fuel in tanks	= 0.8
Fuel indicated on gauges	$= 8{,}000 \times \frac{0.78}{0.8}$
	= 7,800 kg

Although this situation may be confusing in that the actual amount of fuel is greater than that indicated, at least the error is on the side of safety. The more dangerous situation arises when the SG of the fuel in the tanks is less than the SG of the fuel used to calibrate the gauges. In this case, the gauge would indicate that there is more fuel in the tanks than there actually is.

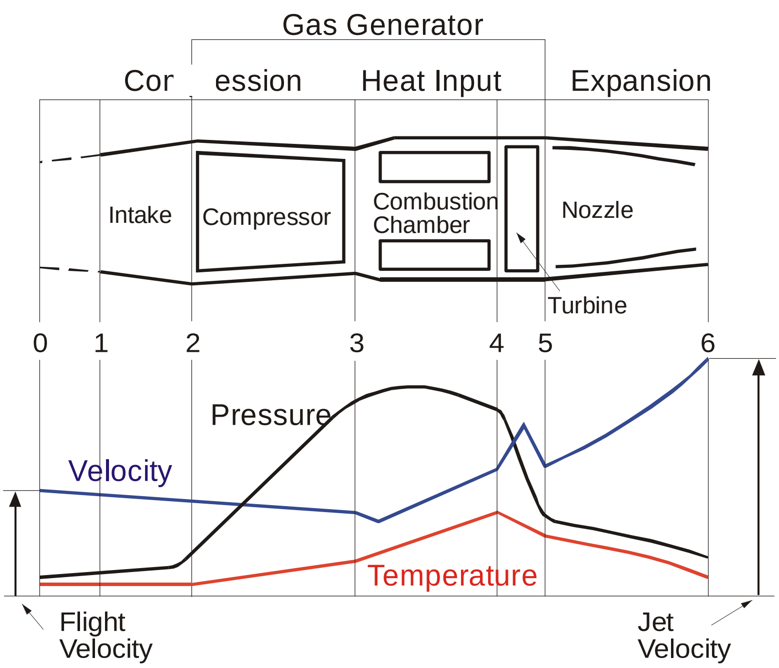
16. **Fuel Weight**. On aircraft which normally operate with full tanks, the actual weight of fuel carried can be calculated from a knowledge of the capacity of the tanks (ie their volume) and the SG of the fuel used. For example, fuel with an SG of 0.8 has a density of 800 kg/m^3. If the capacity of the tanks is 2 cubic metres, then the total weight of fuel is 1,600 kg. The reading indicated on the gauges with full tanks can then be calculated from the formula in para 15.

CHAPTER 2 - JET RANGE

Introduction

1. The turbojet engine consists of five distinct components; air intake, compressor, combustion chamber, turbine and exhaust nozzle, see Fig 1. The core of the engine, comprising compressor, combustion chamber and turbine may be termed the gas generator. Because of external compression, air enters the intake (0-1) at a lower velocity than the flight speed, and is then further slowed (1-2) before the compressor causes a substantial rise in both pressure and temperature (2-3). In the combustion chamber heat energy of fuel is added to the airflow, providing a very high turbine entry temperature (3-4). To keep the gas temperature within acceptable limits, and to provide the large mass flow needed to achieve a high thrust, much more air is used than is required for combustion. The turbine (4-5) extracts energy from the gas flow in order to drive the compressor, resulting in a drop in pressure and temperature at the turbine. The remaining energy from the gas generator is expanded in the nozzle and exhausted as a high velocity jet (5-6).

2-2 Fig 1 Layout of a Typical Turbojet



2. The thrust produced by a turbojet engine can be expressed thus:

Thrust = mass flow of air × (jet pipe speed – intake speed).

ENGINE CONSIDERATIONS

Thrust

3. From the expression for thrust it follows that any factor which affects the mass flow, the jet pipe speed or the intake speed must affect the output of the engine. The variable factors are engine rpm, temperature, altitude and true airspeed:

a. **RPM**. With an increase in rpm the mass flow and acceleration of the flow will increase and this will produce an increase in thrust. Component efficiencies are such that thrust outputs required in flight are obtained using between 50% and 95% of maximum rpm.

b. **Temperature**. As temperature decreases air density increases and this results in an increase in air mass flow and therefore an increase in thrust.

c. **Altitude**. As altitude is increased, at a constant rpm, there is a decrease in density and therefore a decrease in mass flow which leads to a reduction in thrust. Decreasing temperature slightly offsets this.

d. **Speed**. An increase in TAS gives an increase in intake speed. Because jet pipe speed is limited by the maximum permissible jet pipe temperature, the acceleration given to the airflow by the engine is decreased. However, as speed increases, the action of the compressor is supplemented by ram effect which increases the mass flow. These two effects, one trying to increase, the other trying to decrease thrust tend to nullify one another and the thrust of a jet engine is not significantly affected by forward speed.

Specific Fuel Consumption

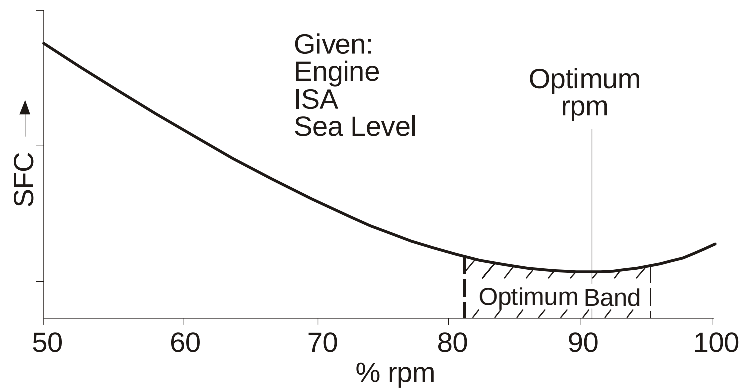
4. Specific fuel consumption (SFC) is the ratio between thrust and gross fuel consumption (GFC):

$$\text{SFC} = \frac{\text{GFC}}{\text{Thrust}}$$

5. SFC is measured in units of fuel per hour per unit of thrust and is therefore affected by the same variables that determine the output of the engine:

a. **RPM**. Turbojet engines are designed to operate at a low SFC in an optimum rpm band around 90% of maximum. The optimum rpm is sometimes known as the design rpm. Engines using an axial flow compressor have a fairly wide band of rpm giving almost optimum SFC, this is illustrated in Fig 2.

2-2 Fig 2 SFC and rpm



b. **Temperature**. The temperature rise through an engine is a measure of thermal efficiency. The upper temperature is limited by maximum permitted jet pipe temperature but a decrease in intake temperature will improve thermal efficiency and SFC will be reduced.

c. **Altitude**. At low altitudes the required amount of thrust for cruising flight is achieved at relatively low rpm where SFC is poor. The higher air temperatures also reduce the engine's thermal efficiency. At higher altitudes rpm have to be increased to obtain the required thrust and eventually the rpm will enter the optimum band, and at the same time the lower ambient temperature will increase the thermal efficiency of the engine. At very high altitude the rpm may need to be increased to beyond the optimum and the rarefied air may cause inefficient combustion. It can be seen that although altitude has practically no direct effect on SFC the secondary effects of altitude require careful consideration.

d. **Speed**. Speed has an insignificant effect upon SFC compared with the increase in rpm required to achieve it.

Summary

6. In summary, the effects of variations in engine rpm, temperature, altitude and speed are:

a. **Thrust**

(1) Increases with rpm

(2) Increases with decreasing temperature

(3) Decreases with increasing altitude

b. **SFC**

(1) Reduces with rpm up to the optimum band

(2) Reduces with decreasing temperature

c. **Speed**. Over the cruising speed range aircraft speed does not significantly affect thrust or SFC.

AIRFRAME CONSIDERATIONS

Efficiency

7. The efficient operation of an aircraft requires that it be flown in a manner which achieves the delivery of the maximum weight (payload) over the maximum range using minimum fuel.

$$\text{Efficiency} = \frac{\text{Work Out}}{\text{Work In}} = \frac{\text{Drag} \times \text{Distance}}{\text{GFC}}$$

$$= \frac{\text{Thrust} \times \text{TAS}}{\text{GFC}}$$

Specific Air Range

8. The distance travelled per unit of fuel is the specific air range (SAR).

$$\text{SAR} = \frac{\text{Distance Travelled}}{\text{Fuel Used}} = \frac{\text{TAS}}{\text{GFC}}$$

but GFC = Thrust $\times$ SFC therefore $\text{SAR} = \frac{\text{TAS}}{\text{Thrust} \times \text{SFC}}$

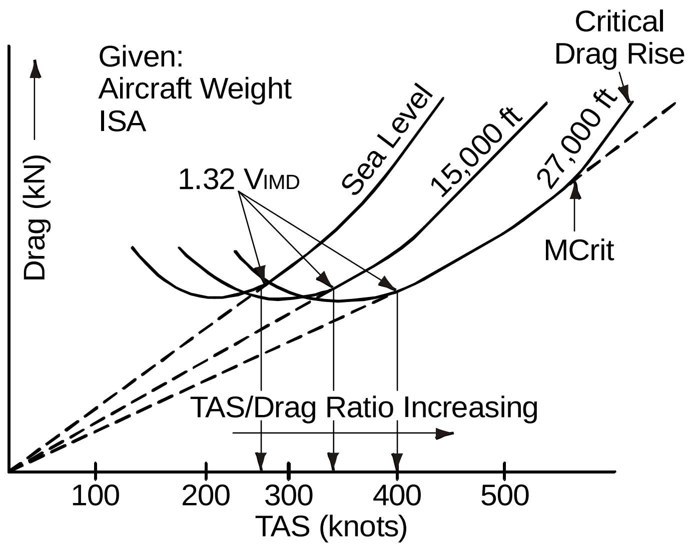
In level unaccelerated flight, thrust is equal to drag so: $\text{SAR} = \frac{\text{TAS}}{\text{Drag}} \times \frac{1}{\text{SFC}}$

TAS/Drag Ratio

9. The TAS/Drag ratio depends upon the aircraft's speed, altitude and weight. Temperature is also a factor, but this will be considered when airframe and engine considerations are brought together.

10. **Altitude**. At a constant IAS the TAS increases with altitude and therefore the TAS/Drag ratio also increases as shown in Fig 3. The maximum TAS/Drag ratio occurs at the tangent to the appropriate altitude curve drawn from the origin of the graph. This speed is 1.32 $\times$ the minimum drag speed (V_{IMD}) for each curve and therefore represents a particular IAS and angle of attack. As altitude is increased compressibility becomes significant as 1.32 V_{IMD} approaches the critical Mach number (M_{CRIT}) for the angle of attack which this speed represents. At M_{CRIT} the drag rise changes the slope of the TAS/Drag curve and the maximum TAS/Drag ratio will no longer be achieved at 1.32 V_{IMD}. If the M_{CRIT} is maintained, allowing the IAS to decrease, the TAS/Drag ratio will continue to increase but at a reduced rate until it reaches its optimum value.

2-2 Fig 3 $\frac{\textbf{TAS}}{\textbf{Drag}}$ Ratio at Various Altitudes

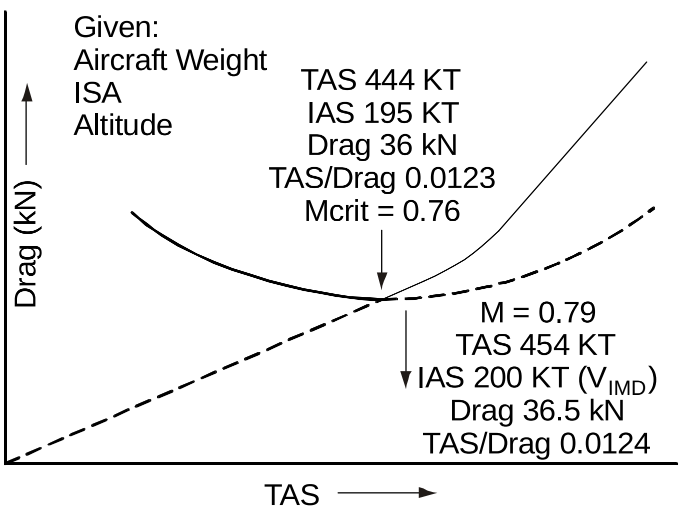


11. **Optimum Altitude**. The optimum altitude is reached when V_{IMD} coincides with M_{CRIT}. In this situation, the TAS is limited by compressibility and the drag has been reduced to a minimum. It should be recognized that as the IAS is allowed to fall towards V_{IMD} the aircraft's angle of attack will gradually increase from its 1.32 value. The increased angle of attack results in a reduction in M_{CRIT}; however, the Mach number appropriate to 1.32 V_{IMD} should be maintained to achieve the optimum TAS/Drag ratio. This can be seen in Fig 4 where a reduction in M_{CRIT} gives a TAS/Drag ratio which is slightly less than maximum. At this Mach number the TAS will fall as altitude is increased.

12. **Optimum Mach Number**. In practice the optimum Mach number may vary slightly with altitude and is in fact found by flight testing. Nevertheless the pattern of near constant Mach number and reducing IAS to achieve optimum TAS/Drag ratio with increasing altitude holds good.

13. **Reducing Weight**. As weight reduces, V_{IMD} and drag reduce. The method used to maintain the optimum TAS/Drag ratio is to climb the aircraft at a constant Mach number and angle of attack at a fixed throttle setting. As the aircraft gently climbs the thrust reduces at the right rate to balance the drag which is reducing because at a constant angle of attack the IAS is reducing.

2-2 Fig 4 Optimum TAS/Drag Ratio



Summary

14. Turbojet airframe efficiency can be summarized as follows:

a. The aircraft should be flown for the maximum value of TAS/Drag.

b. At low altitudes the maximum TAS/Drag ratio is achieved at 1.32 V_{IMD}.

c. Above the altitude where 1.32 V_{IMD} is equivalent to M_{CRIT} maximum TAS/Drag ratio is achieved at an optimum Mach number determined by flight testing.

d. TAS/Drag ratio improves with altitude to its maximum value, which is obtained at a pressure altitude where V_{IMD} is equivalent to a particular Mach number which is slightly in excess of M_{CRIT}

e. As weight reduces the aircraft should be flown in a cruise climb at a constant Mach number and angle of attack.

ENGINE AND AIRFRAME COMBINATION

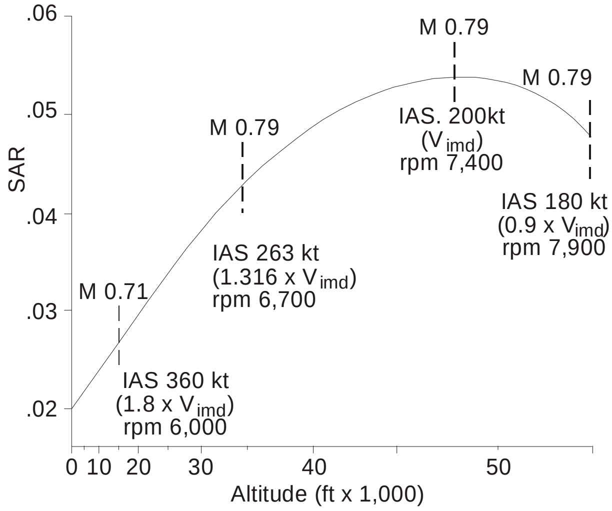
Altitude

15. **RPM and Thermal Efficiency**. At a constant IAS the TAS/Drag ratio increases with altitude. For best SAR at low altitude, airframe considerations require that the aircraft should be flown at 1.32 V_{IMD} but, at this speed only low rpm are required and SFC is poor. There is also a penalty to be paid in engine thermal efficiency because of the high ambient air temperatures. As altitude is increased thrust reduces and rpm need to be increased towards the optimum band where SFC improves; the thermal efficiency also increases as the temperature falls. Thus, in general SAR improves with increasing altitude.

16. **Optimum Altitude**. The increase in TAS/Drag ratio as altitude is increased is halted by the formation of shockwaves. In addition, as altitude is increased thrust begins to fall, and eventually the rpm have to be increased beyond the optimum value, resulting in reduced efficiency. An altitude will be reached at which, due to compressibility and engine limitations, the SAR starts to fall. For maximum range a turbojet engine aircraft should therefore be flown at a particular Mach number which is governed by compressibility. Since at the same time the airframe factor of SAR requires the aircraft to be flown at the IAS for minimum drag, the optimum altitude is the pressure altitude at which these conditions coincide.

17. **Optimum Range Speed.** Assuming ISA temperatures, particular values of rpm are required to provide the necessary thrust which, by design, should be contained within the optimum band. The rpm required to maintain optimum range speed will also increase with altitude. Actual values of rpm and Mach number can only be obtained by flight testing, but Fig 5 provides an example of the speed and rpm relationship for a maximum SAR performance curve.

2-2 Fig 5 Example of Speed and Rpm Relationships for Maximum SAR



The Cruise Climb

18. **Effect of Reducing Weight**. V_{IMD} is proportional to the square root of the aircraft weight. Thus, to ensure maximum performance as weight changes, altitude must change, i.e. the $\frac{\text{weight}}{\text{relative pressure}}$ ratio must remain constant. As weight falls the reducing V_{IMD} is equivalent to the Mach number at an increasing altitude (falling relative pressure). Hence the cruise-climb technique for jet aircraft.

19. **Engine Rpm.** As long as the temperature remains constant the rpm set at one weight will ensure the correct rate of climb for the cruise-climb. If the temperature falls the thrust increases and rpm should be reduced. The ratio of rpm (N) to the square root of the relative temperature (t) should be constant, ie $\frac{(\mathrm{N})}{\sqrt{\mathrm{t}}}$ is constant.

20. **Effect of Changing Temperature**

a. **Altitude**. The altitude is governed by the IAS/Mach number relationship which is not affected by temperature. However, in very hot temperatures the engine may be unable to provide the rpm demanded by $\frac{(\mathrm{N})}{\sqrt{\mathrm{t}}}$. In this event a lower altitude may have to be flown at reduced efficiency.

b. **Thrust**. At low temperatures the rpm must be reduced and at high temperatures they must be increased. The rpm should remain in the optimum band to ensure that the SFC does not adversely affect SAR.

c. **SAR**. An increase in temperature will lead to an increase in TAS but there will be a similar increase in GFC through SFC increasing because of poorer thermal efficiency, so that SAR remains unchanged.

Effect of Wind

21. It is important to note that whenever range flying is mentioned in Aircrew Manuals it invariably refers to still-air range. A head wind will have a detrimental effect on SAR and, conversely, a tail wind will increase SAR . It is therefore sound practice to examine the winds at various altitudes while flight planning to determine the best height to fly.

22. The basic requirements for maximum range in a turbojet aircraft have been shown to be minimum SFC and maximum TAS/Drag ratio. Broadly speaking, to fulfil these requirements and achieve the best still-air range jet aircraft normally fly as high and fast as possible, the range speed being governed by that at which compressibility effects start to degrade the TAS/Drag ratio. If a strong head wind component is encountered in any jet aircraft the best solution lies in seeking a flight level where the effect of the head wind is minimized, bearing in mind the other factors affecting range. In high performance aircraft, increasing speed into a head wind is unlikely to be of benefit since the TAS/Drag ratio will be adversely affected by compressibility effects. In low performance aircraft increasing speed may be worthwhile since compressibility effects will be of no consequence.

CHAPTER 3 - JET ENDURANCE

Introduction

1. Flying for endurance implies flying in conditions which realize the minimum fuel flow and so the longest possible time in the air on the fuel available. The need for maximum endurance arises less frequently than that for maximum range, but circumstances can arise which require aircraft to loiter or stand-off for varying periods of time during which fuel must be conserved. Whereas range flying is more closely concerned with specific fuel consumption (SFC) and air nautical miles per kilogram of fuel, endurance flying is concerned with the gross fuel consumption (GFC), i.e. the weight of fuel consumed per hour.

Principles

2. Broadly, since fuel flow is proportional to thrust, fuel flow is least when thrust is least; therefore maximum level flight endurance is obtained when the aircraft is flying at the Indicated Airspeed (IAS) for minimum drag (V_{IMD}), because in level flight thrust is equal to drag.

3. Maximum endurance is obtained at an altitude which is governed by engine considerations. Although for a given set of conditions the IAS for minimum drag remains virtually constant at all altitudes, the engine efficiency varies with altitude and is lowest at the lowest altitudes where rpm must be severely reduced to provide the low thrust required.

4. To obtain the required amount of thrust most economically, the engine must be run at maximum continuous rpm. Therefore maximum endurance is obtained by flying at such an altitude that, with the engine(s) running at or near optimum cruising rpm, just enough thrust is provided to realize the speed for minimum drag, or M_{CRIT}, whichever is the lower. Above the optimum altitude little, if any, additional benefit is obtained, and in some cases there may be a slight reduction because burner efficiency decreases at or about the highest altitude at which level flight is possible at V_{IMD}**.** In general, optimum endurance is obtained by remaining between 20,000 ft and the tropopause at the recommended IAS and appropriate rpm. The greater the power/weight ratio of the aircraft, the greater will be the optimum height. With aircraft having high power/weight ratios, maximum endurance is obtained at the tropopause.

5. Altitude should only be changed to that for maximum endurance if the aircraft is above or near the endurance ceiling, otherwise if the aircraft is climbed from a much lower altitude, a considerably higher fuel flow will be required on the climb thereby reducing overall endurance.

6. On engines having variable swirl vanes, the consumption increases markedly if the rpm are so low that the swirl vanes are closed. If the altitude is low enough to cause the swirl vanes to close at the rpm required for V_{IMD}**,** the aircraft should either be climbed to the lowest altitude at which V_{IMD} can be obtained with the swirl vanes open, or the rpm increased to the point at which the vanes open, accepting the higher IAS.

Effect of Weight

7. Drag and thrust at the optimum IAS are functions of the all-up weight; the lower the weight the lower the thrust and fuel flow. Endurance varies inversely as the weight and not as the square root of the weight as in range flying because in pure endurance flying the TAS has no importance.

Effect of Temperature

8. In general, the lower the ambient air temperature, the higher the endurance, due to increased thermal efficiency, and vice versa. However, the effect is not marked unless the temperature differs considerably from the standard temperature for the altitude. In any case, the captain can do nothing but accept the difference, since any set of circumstances requiring flying for endurance usually ties the aircraft to a particular area and height band.

Twin and Multi-engine Aircraft

9. When flying for endurance in twin or multi-engine aircraft at medium and low altitude, endurance can be improved by shutting down one or more engines. In this way, the live engine(s) can be run at rpm closer to the optimum for the thrust required to fly at V_{IMD}, thus improving GFC. Provided that the correct number of engines are used for the height, altitude has virtually no effect on the endurance achieved.

Use of the Fuel Flowmeter

10. The fuel flowmeter is a useful aid when flying for endurance. If the endurance speed is unknown, the throttle(s) should be set at the rpm which give the lowest indicated rate of fuel flow in level flight for the particular altitude.

Reporting Procedure - Aircraft Endurance

11. Whenever reporting aircraft endurance, the time for which the aircraft can remain airborne should be given. It is confusing and dangerous to report endurance in terms of amount of fuel remaining because of the possibility of misinterpretation.

Conclusions

12. Maximum endurance is achieved by flying at an altitude where optimum rpm give the minimum drag speed. It will rarely pay to climb to a higher altitude unless the commencing altitude is very low; in any event, the instruction or need to fly for endurance may preclude this. At the lower altitudes, maximum endurance may be obtained either by flaming-out engines to use optimum rpm on the remainder, or by using near-optimum rpm to give V_{IMD}. It should be remembered that:

a. The importance of flying at V_{IMD} outweighs engine considerations, always assuming that an engine, or engines, will be stopped in the low level case.

b. At lower altitudes, there will be a slight decrease in endurance due to the higher ambient temperature reducing engine thermal efficiency.

CHAPTER 4 - JET CRUISE CONTROL

Introduction

1. Previous chapters of this section have set out the pattern of turbojet aircraft performance. This chapter presents the most simple, but at the same time, most efficient means of flying these aircraft for maximum range in various roles. The most commonly used procedures are studied under various headings, each one being more efficient than its predecessor.

Low Altitude

2. At low level, the efficiency of the engine (rpm within the optimum band) outweighs the efficiency of the airframe (1.32 × V_{IMD}). Normally, a high IAS, approximately 2.0 × V_{IMD}, is flown using rpm which are reduced as weight decreases. Range is about 30% to 40% of maximum.

Medium Altitude

3. At medium altitude, a reducing IAS (1.32 × V_{IMD}) is flown using reducing rpm contained within the lower half of the optimum band. Range is about 50% to 70% of maximum.

High Altitude

4. **Constant Altitude**. To take maximum advantage of decreasing weight, an optimum Mach number (set by compressibility) is initially flown at an altitude where near maximum cruising rpm are required. As weight reduces, so the speed and rpm are reduced to maintain a constant angle of attack. The advantage to the $\frac{\text{TAS}}{\text{drag}}$ ratio is that although the TAS is reduced, because the angle of attack is being maintained at a constant, drag is reducing as the square of the speed, thus resulting in an overall increase in the $\frac{\text{TAS}}{\text{drag}}$ ratio. The required decrease in rpm will have little effect on the $\frac{1}{\text{SFC}}$ ratio, as the engine will still be operating within the optimum rpm range. However, on long flights, this method does lead to problems in flight planning because of the changing speed, and so an alternative and more practical method is to maintain the originally selected Mach number by slightly reducing rpm as weight decreases. An improvement in the $\frac{\text{TAS}}{\text{drag}}$ ratio is thus obtained as TAS is now constant, and drag is reducing as weight decreases. Range is about 80% to 90% of maximum.

5. **Stepped Climb**. An optimum Mach number is flown at an altitude approximately 1,000 ft above the ideal starting altitude set by the optimum $\frac{\text{weight}}{\text{relative pressure}}$ value (see Volume 2, Chapter 2, Para 18). Level cruise is flown by reducing rpm until the aircraft is 3,000 ft below the ideal cruise climb altitude. A climb is then made to the next semi-circular height 1,000 ft above the ideal altitude, and so on. Range is about 95 % of maximum.

6. **Cruise Climb**. An optimum Mach number is flown starting at an altitude determined by the optimum $\frac{\text{weight}}{\text{relative pressure}}$ using an rpm setting determined by the optimum $\frac{\text{N}}{\sqrt{\text{t}}}$ value. Range is

maximum. However, in very hot conditions, a lower $\frac{\text{weight}}{\text{relative pressure}}$ value (lower altitude) may have to be accepted with a lower $\frac{N}{\sqrt{t}}$ value (lower rpm) to avoid exceeding maximum cruising rpm.

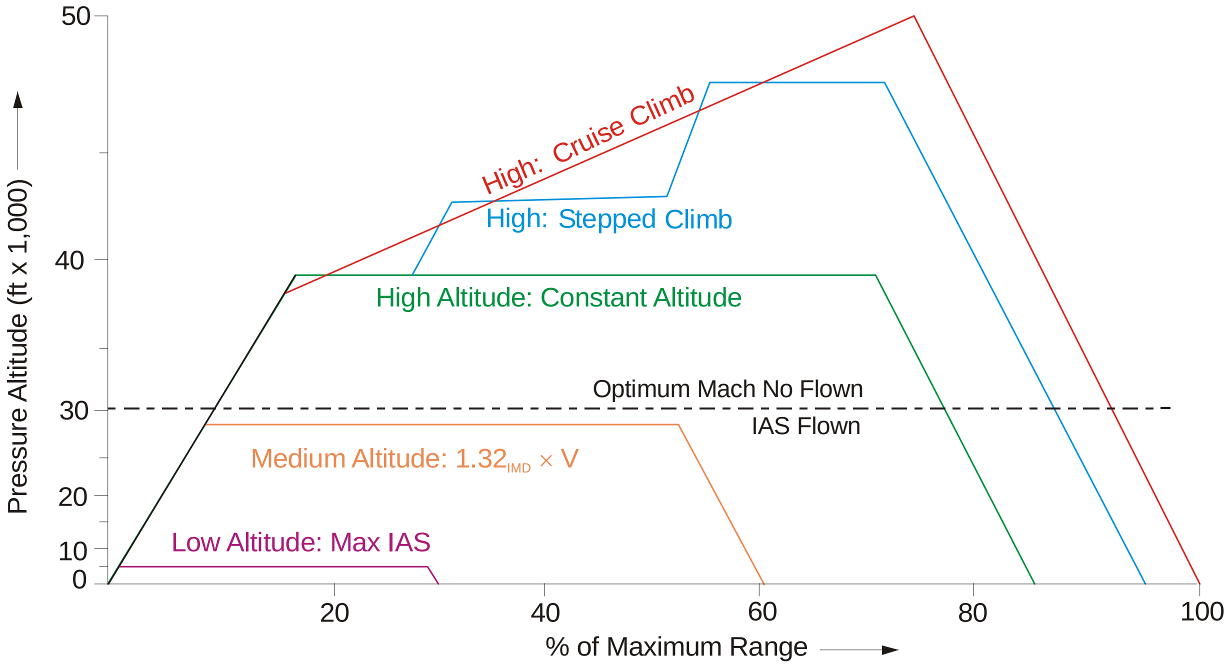
Fast Cruise

7. Turbojet aircraft should never be flown at slower speeds than those outlined above except when endurance is required. Frequently, faster speeds are flown in the interests of time economy at the expense of fuel economy. The performance penalties are obviously progressive.

Summary

8. In Fig 1, the profile of each type of cruise procedure is shown with an indication of the range achieved.

2-4 Fig 1 Profiles of Turbojet Aircraft Control Procedures



9. The tables below relate to a typical four-jet aircraft and indicate the advantage in SAR (range) given by a cruise climb as opposed to a level cruise. For this aircraft, the cruise climb should be started at flight level (FL) 420. Using the cruise climb technique, ie rpm constant within the optimum band, at a constant indicated Mach number (IMN) and climbing as weight is reduced, it can be seen that the SAR improves from 0.0267 to 0.0325, whilst weight is reduced by 10,000 kg and the aircraft is climbed through 4,000 ft. In the level cruise case, the cruise is started 1,000 ft higher at FL 430 and SAR is slightly worse than in the cruise climb case. At an all-up weight (AUW) of 51,000 kg the SAR, height and rpm are all equal for both techniques. However, by the time the AUW has reduced to 44,000 kg, the SAR in the case of the level cruise is 1.85% worse than in the climbing case, and the disparity would continue to increase as weight is reduced. If air traffic regulations preclude the use of the cruise climb technique, the aircraft using the level cruise should now be climbed through 4,000 ft, ie to the next appropriate semi-circular height and the cruise continued.

CRUISE CLIMB

Wt	FL	SAR (anmpkg)	rpm
54,000	420	0.0267	7,050
51,000	430	0.0281	7,050
44,000	460	0.0325	7,050

LEVEL CRUISE

Wt	FL	SAR (anmpkg)	Rpm
54,000	430	0.0266	7,130
51,000	430	0.0281	7,050
44,000	430	0.0319	6,700

10. It is apparent from this study of turbojet engine aircraft performance that, by virtue of their engines, these aircraft operate best at a high altitude in a cruise climb, because of the considerable reduction in weight which occurs during flight. Their speed is limited by compressibility to a particular Mach number whose value depends upon design.

CLIMB, DESCENT AND EMERGENCIES

Climb

11. Since flying at high altitudes is all-important for a high SAR, it is essential that the initial cruising altitude is reached as quickly as possible. This results in the use of the highest possible rpm and therefore a high GFC, but it is the most economical method of reaching the required altitude. Thus, the climb is made at maximum permitted climbing (intermediate) rpm and at the correct climbing speeds. If there is a time limit on the use of intermediate power, and the time for the climb exceeds this, then a quicker climb is achieved by using maximum continuous rpm initially, and intermediate (climbing) rpm, for the full time period allowed, during the latter period of the climb.

Descent

12. The most economical method of descending is to start before the destination is reached, at a rate that allows the aircraft to arrive over the destination at the required altitude. In this way altitude is most efficiently converted into distance.

13. There are two practical methods of descending: either a cruise descent with the aircraft clean and the throttles closed as far as possible, bearing in mind the need to maintain pressurization and aircraft services; or a rapid descent, using maximum drag devices and a higher speed, with throttles set as for the cruise descent. The fuel used in the rapid descent is less than in the cruise descent, but since the distance covered is also less a true comparison between the two methods can only be made by taking into account the extra fuel used at altitude before starting the rapid descent. In practice, the method of descent used will vary with aircraft type and role and is normally governed by techniques recommended in Aircrew Manuals and procedures laid down by the operating authority.

Emergencies

14. **Engine Failure**. If one or more engines fail when flying for range at high altitude the reduced thrust will cause the aircraft to lose height and range will be reduced. The loss in range depends on the amount of height lost; the height loss being governed by the reserve of thrust (rpm) available from the live engines. Maximum range will be achieved by flying the aircraft at the best range speed and highest rpm available (maximum continuous). The technique for range flying after engine failure is to assume the correct range speed for the condition (available in the ODM) and set maximum continuous rpm. The aircraft will then drift down until the increase in thrust enables height to be stabilized. For maximum range, the aircraft should then be cruise climbed from this new height, maintaining the IMN achieved at the stabilized height. When engine failure occurs, the range will be less than is possible with all engines operating. For example, by using the correct technique, the loss of remaining range for a typical four-jet aircraft is approximately 6% for loss of one engine and 20 % for loss of two engines.

15. **Cabin Pressure Failure**. If cabin pressure is lost and the aircraft is forced to descend rapidly, the cruise should be resumed at a height limited by the type of oxygen equipment carried. Best range will then be achieved by adopting a cruise at a constant altitude and the recommended range speed for the height. As AUW is reduced, the range speed will also reduce and this can be achieved by either throttling back, or, if possible, flaming-out an engine. If, for operational purposes or in the interests of navigation, the speed is kept constant, the resulting loss in range is not great. If the recommended landing reserve of fuel is available, a constant speed can be justified in most cases.

CHAPTER 5 - TURBOPROP RANGE AND ENDURANCE

Introduction

1. Turboprop engines combine the efficiency of the propeller with the power output of the gas turbine to gain an advantage in power/weight ratio over jet engines at speeds up to about 350 kt. The Range, Endurance, and other performance aspects of turboprop aircraft can be examined against the same engine and airframe factors that were considered for jets in Volume 2, Chapter 1.

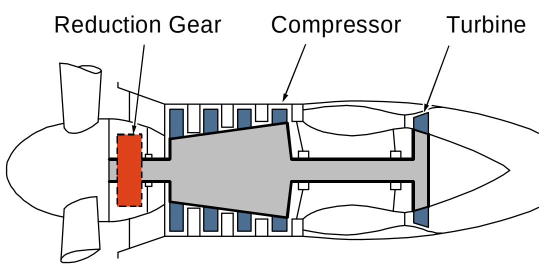
ENGINE CONSIDERATIONS

General

2. The turboprop engine consists of a gas turbine engine driving a propeller. There are two main types of these engines depending on the arrangement of the compressors and propellers in relation to the turbine:

a. **Direct Drive Turbine**. In the direct drive turbine engine the compressor and the propeller are driven on a common shaft by the turbine, the power being transmitted to the propeller through a reduction gear (Fig 1).

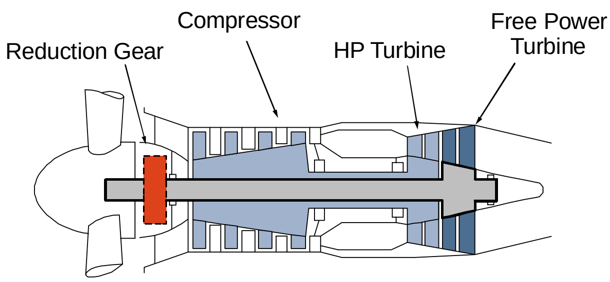
2-5 Fig 1 Direct Drive Turboprop Engine



b. **Free Power Turbine**. In the free power turbine engine, the compressor is connected to the high pressure turbine. The free power turbine is coupled independently to the propeller via the reduction gear (Fig 2a).

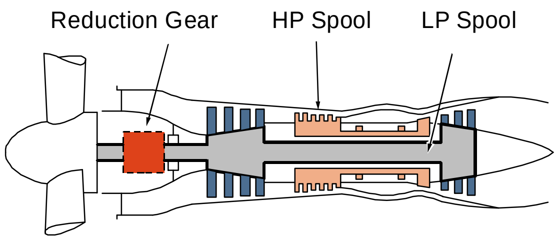
2-5 Fig 2 Free Power Turbine Engine/ Compound Engine

Fig 2a Free Power Turbine Engine



c. **Compound Engine**. The compound engine is a two-spool engine, with the propeller driven by the low-pressure spool (Fig 2b).

Fig 2b Compound Engine



Power Utilization

3. The energy produced by burning fuel in the engine is used in two ways. Over 90% of the energy is extracted by the turbines to drive the compressor(s) and the propeller; the remaining energy appears in the jet exhaust as thrust. The following figures show the utilization and amount of power output in a typical turboprop engine:

Turbine produces	7300 kW
Compressor uses	4100 kW
Loss through reduction gear and ancillary equipment	75 kW
Total power available for propulsion	3125 kW
Jet thrust	230 kW

Factors Affecting Power Output

4. **RPM**. For practical purposes, the power output from a turboprop engine varies directly as the rpm. In cruising flight however, the engine is run as follows:

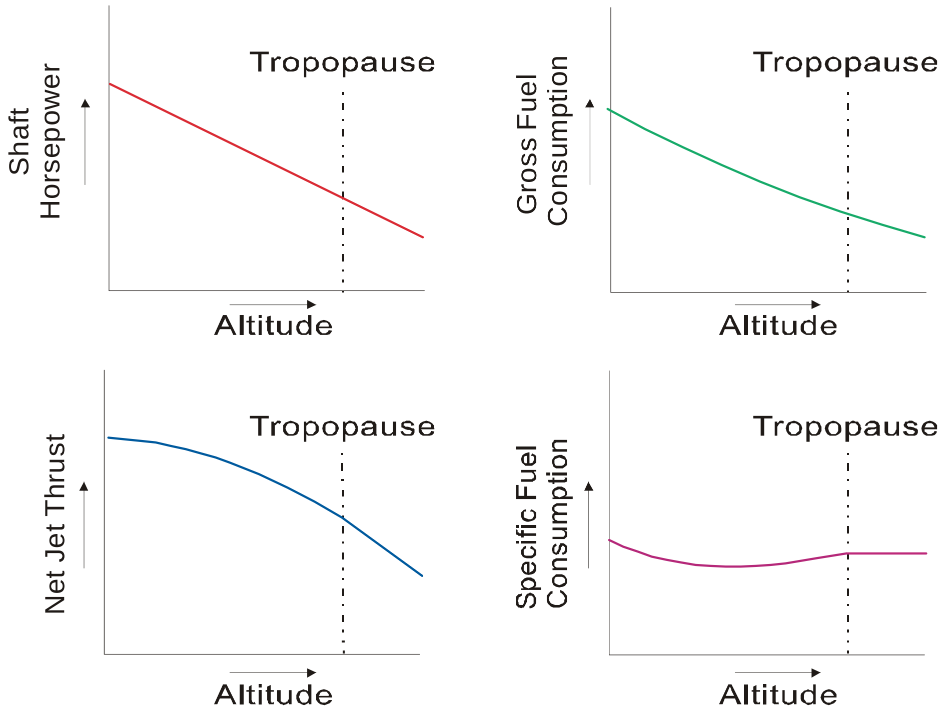
a. **Direct Drive Turbine.** The rpm are maintained at the maximum continuous setting (MCrpm).

b. **Free Power Turbine**. The low pressure system is kept at MCrpm but the high pressure side may be throttled if required; however, this reduces efficiency and therefore increases the specific fuel consumption (SFC).

5. **Temperature**. An increase in intake temperature will reduce power output because the mass flow through the engine will be reduced due to reduced air density.

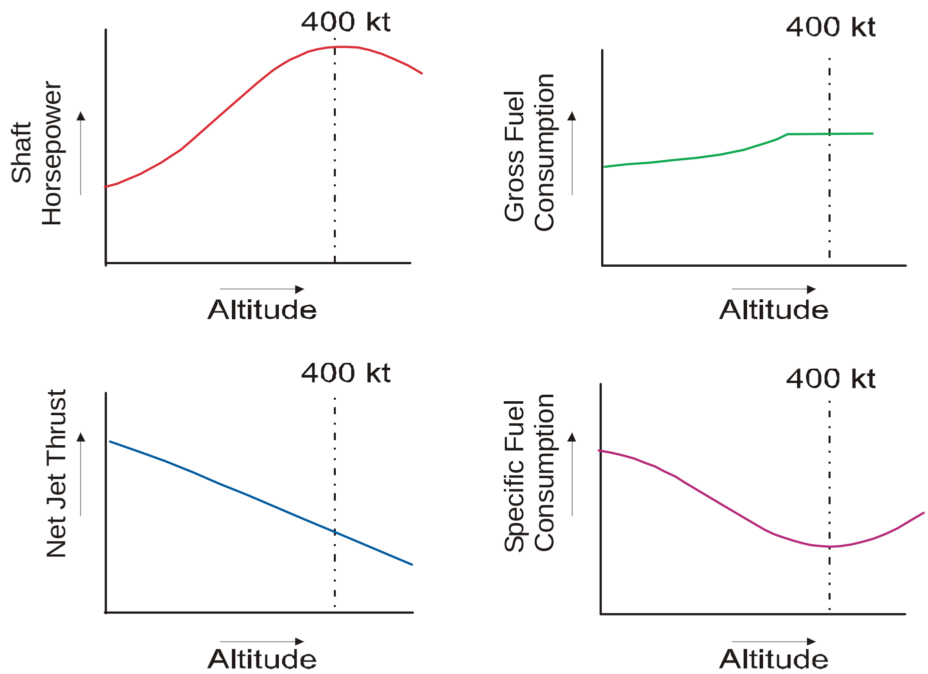
6. **Altitude**. At a constant TAS, power output falls with increase in altitude due to the decrease in air density, but the rate of reduction of power is mitigated by the fall in air temperature, up to the Tropopause above which it reduces more sharply (see Fig 3 which shows the generalized effects of altitude on turboprop performance).

2-5 Fig 3 Effect of Altitude on Turboprop Performance



7. **Forward Speed**. As forward speed increases, ram effect supplements the pressure at the intake, thus augmenting the action of the compressor and thereby increasing power output. However, propeller efficiency begins to fall at about 300 kt TAS and the gain due to the ram effect is progressively degraded until, at about 450 kt TAS, reducing propeller efficiency completely offsets the increasing ram effect. Generally, the effects of propeller efficiency and ram pressure can be ignored in the cruising range of current turboprop aircraft (see Fig 4 which shows the generalized effects of speed on turboprop performance).

2-5 Fig 4 Effect of Speed on Turboprop Performance



Specific Fuel Consumption (SFC)

8. The gross fuel consumption (GFC) of an engine, measured in quantity per unit time, gives no indication of the operating efficiency except in a very general way. This would be analogous with comparing miles per gallon achieved on one car journey with another, along a possibly quite dissimilar route. Comparisons of fuel consumption between vehicles with different capacity engines would be meaningless. If, however, the fuel used is related to power produced, then a means exists of assessing how efficiently the fuel is used and comparisons can be made between different power plants. The measurement of fuel used/hr/kW is called the specific fuel consumption. The term, fuel used/hr, is the GFC, therefore SFC = GFC/kW. The factors affecting power output also affect the SFC.

9. **RPM** Turboprop engines are designed to operate most efficiently at one particular rpm setting, known as maximum continuous rpm (MCrpm). Therefore, for the best (minimum) value of SFC the engine should be operated at MCrpm at all times.

10. **Temperature**. SFC improves in cold air because of the increased thermal efficiency of the engine.

11. **Altitude.** Altitude has no direct effect on SFC. As altitude increases, power decreases with reducing air density but GFC is reduced automatically by the fuel control unit (FCU) within the engine to keep the fuel/air ratio constant. However, SFC does improve because of lowering temperatures with increasing altitude up to the point where propeller efficiency begins to fall due to reducing air density (see Fig 3).

12. **Forward Speed**. Shaft power increases with speed because of the improved mass flow through the engine. The GFC will be higher, with shaft power the dominant factor, up to the point where the FCU power-limiter prevents any further fuel flow, to avoid over-stressing. The shaft power will continue to rise due to the ram-ratio improvement. However, the increased shaft power is converted to thrust by a propeller which loses efficiency at high forward speed, thereby losing thrust. In comparison with the pure jet engine, which shows a straightforward increase in SFC with increasing speed, the turboprop would, in general, show an improving (lower) SFC at increasing speed up to the point where propeller losses bring about a rapid deterioration (see Fig 4).

Summary

13. In order to achieve maximum efficiency (low SFC), the turboprop engine should be operated:

a. At MCrpm.

b. At high altitude (better thermal efficiency in colder air).

c. At a TAS between 300 kt and 400 kt, so that advantage may be taken of ram effect while the propeller efficiency is still high.

AIRFRAME CONSIDERATIONS

Flying for Range

14. An aircraft is flown for range in such a way that a maximum weight of payload is delivered over a maximum distance using minimum fuel. This aim can be expressed as:

$$\text{Efficiency} = \frac{\text{work out}}{\text{work in}} = \frac{\text{weight} \times \text{distance}}{\text{fuel used}}$$

15. Specific air range (SAR) is defined as air distance travelled per unit quantity of fuel, i.e. nautical miles per kilogram of fuel. Thus:

$$\text{SAR} = \frac{\text{distance travelled}}{\text{fuel used}}$$

16. If the above expression is divided, top and bottom, by time, then:

$$\frac{\text{distance travelled}}{\text{time}} \times \frac{\text{time}}{\text{fuel used}} = \frac{\text{TAS}}{\text{GFC}}$$

17. It was shown in para 8 that the efficiency of a turboprop engine is the ratio of GFC to power output, i.e.:

$$\text{SFC} = \frac{\text{GFC}}{\text{Power}} \quad \therefore \text{GFC} = \text{power} \times \text{SFC}$$

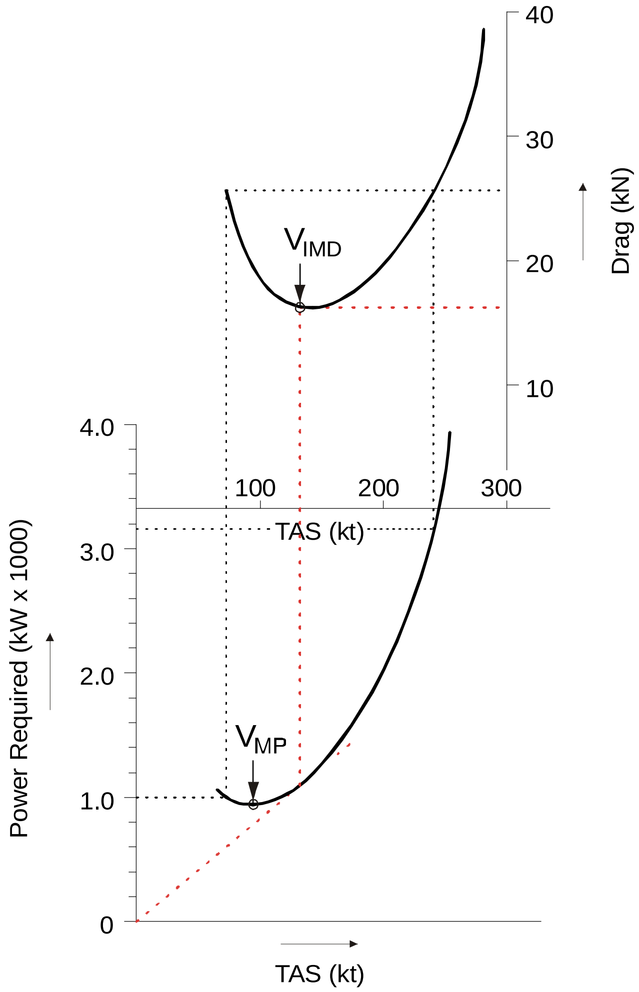
Substituting 'power × SFC' for GFC in the equation for SAR at para 16:

$$\text{SAR} = \frac{\text{TAS}}{\text{power}} \times \frac{1}{\text{SFC}}$$

18. Therefore, at a particular weight, in still air, a turboprop aircraft should be flown for a maximum product of airframe efficiency, TAS/power and engine efficiency, 1/SFC to achieve maximum SAR.

19. Fig 5 shows how a graph of TAS against power required has been evolved from a TAS/drag curve by multiplying each value of drag by the appropriate TAS and converting to kilowatts. Note that the lowest point on the TAS/power curve, known as the minimum power speed (V_{MP}), is slower than the indicated minimum drag speed (V_{IMD}). Also that the maximum value of the TAS/power ratio is found at a point where the tangent from the origin touches the curve. This speed coincides with V_{IMD} for the following reason.

2-5 Fig 5 Derivation of TAS / Power



Given: Aircraft Weight; ISA; Sea Level

Power required is a product of drag and TAS, thus:-

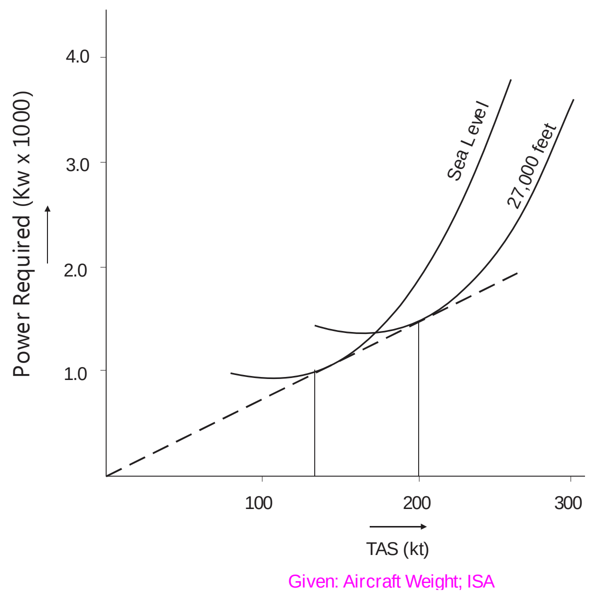
$$\text{Power} = \text{drag} \times \text{TAS}$$

$$\frac{\text{TAS}}{\text{power}} = \frac{1}{\text{drag}}$$

i.e., maximum TAS/power ratio occurs at the minimum drag or maximum 1/drag speed. Note that Fig 5 is drawn for sea level conditions where TAS = EAS.

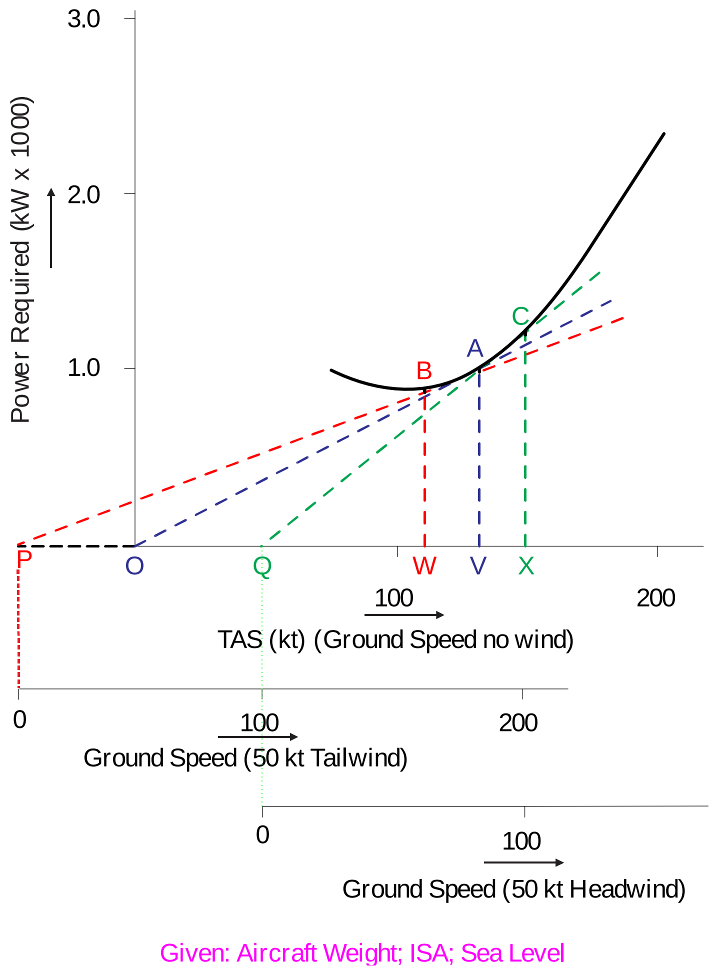
20. **Effect of Altitude**. An aircraft flying at V_{IMD} experiences a constant drag at any altitude, because V_{IMD} is an indicated airspeed. This ignores compressibility which is unlikely to affect the airframe performance. However, as altitude increases, the TAS for a given EAS increases, but the power required also increases by the amount (power required = drag × TAS). Thus the ratio TAS/power is unaffected. Fig 6 shows how power required increases with altitude, whilst airframe efficiency remains unchanged.

2-5 Fig 6 Effect of Altitude on TAS/Power Ratio



21. **Effect of Wind Velocity**. Range should properly be an expression of ground distance divided by fuel used. thus in place of TAS/power, ground speed/power should be studied. In Fig 7 a normal TAS/power curve has been given 3 horizontal axes.

2-5 Fig 7 Effect of Wind Component



a. In still air, a tangent from the origin O indicates a minimum drag speed V and the specific range is OV/AV.

b. With a 50 kt tail wind, the $\frac{\text{ground speed}}{\text{power}}$ ratio at this speed V would be $\frac{PV}{AV}$. To obtain a maximum $\frac{\text{ground speed}}{\text{power}}$ ratio in these conditions a slightly slower speed W is flown, where PB is the tangent from P to the curve, and $\frac{PW}{BW}$ represents the maximum possible efficiency.

c. Similarly, with a 50 kt headwind, the $\frac{\text{ground speed}}{\text{power}}$ ratio is maximum at a slightly higher speed X, where QC is the tangent from the false origin Q to the curve, and $\frac{QX}{CX}$ represents the maximum efficiency.

22. **Air Speed Adjustments**. The effects of head and tail winds on range speed are small and depend upon their numerical relationship to the TAS. Unless a headwind exceeds 25% of TAS, or a tailwind 33% of TAS, air speed adjustments need not be made.

23. **Recommended Range Speed (RRS).** In practice, aircraft are flown for range at a speed slightly faster than V_{IMD} for two reasons:

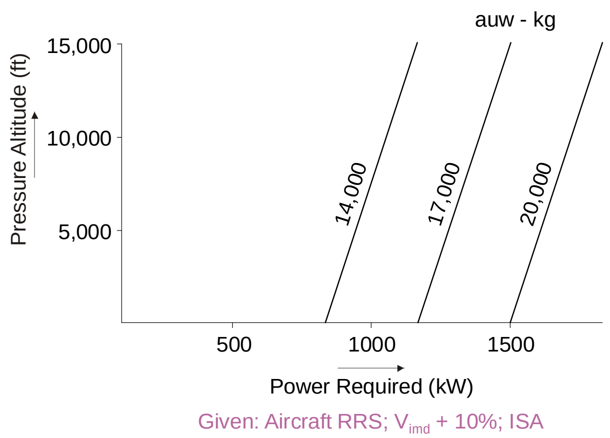
a. The variation of $\frac{\text{TAS}}{\text{power}}$ at speeds near V_{IMD} is negligible and a slightly higher speed will allow a faster flight without undue loss of range.

b. When flying at V_{IMD}, any turbulence or manoeuvres will cause a loss of lift and height. This height can only be regained by the application of more power and consequent fuel wastage. A higher speed ensures a margin for such events.

24. Turboprop aircraft are, therefore, normally flown at the recommended range speed which is some 10% to 20% higher than V_{IMD}.

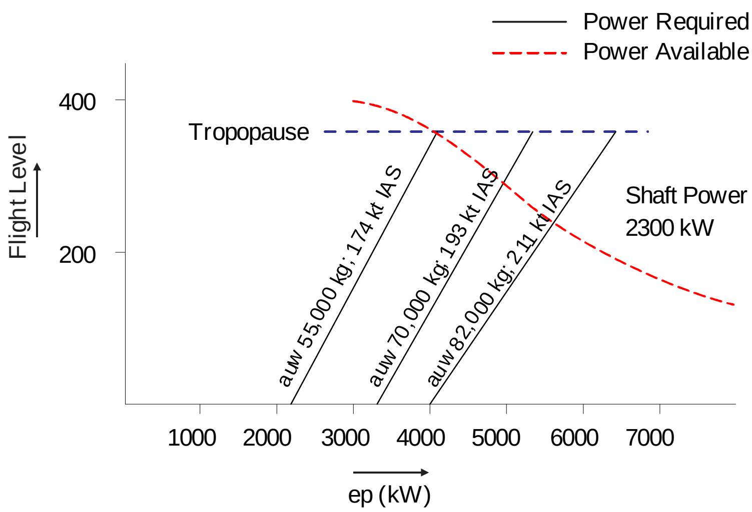
25. **Effect of Weight**. As V_{IMD} is proportional to the square root of the all-up weight, so also is RRS. The power required at RRS depends upon the drag and the TAS. Drag depends upon the lift/drag ratio at the angle of attack represented by this airframe speed, and the weight. As the TAS depends upon the value of RRS (an indicated airspeed), altitude and temperature, it is possible to predict the power required by an airframe flown at RRS for a whole spectrum of weights and altitudes (Fig 8).

2-5 Fig 8 Power Required at Various Altitudes and Weights



26. **Altitude to Fly**. Because the TAS/power ratio achieved at RRS is independent of altitude, the altitude to fly is determined by engine considerations. At sea level more power will be developed by the engines at the MCrpm setting than is required, but this reduces with height. Therefore, at some particular altitude, power output will equal the power required to drive the airframe at the RRS for that altitude, and this is the altitude to start the cruise for range. Fig 8 showed the power required to propel an airframe at RRS at various weights against altitude. Fig 9 gives an example of the weight/power relationship.

2-5 Fig 9 Derivation of Flight Level for Maximum Range



Given: Aircraft Engines; MCrpm; ISA

Endurance

27. For maximum endurance, minimum fuel flow is required and this is achieved when the product of power required and SFC is a minimum. The airframe demands least power at minimum power speed, plus about 10 kt for control, at as low an altitude as possible. The engines are most efficient at MCrpm at a high altitude (cold temperature) and a fairly high TAS (around 300 kt). Due to these conflicting requirements a compromise is achieved at a medium altitude where an airframe speed in the region of V_{IMD} is maintained by throttling the engines. The actual altitude chosen is not unduly critical as there are almost equally good reasons for flying higher (increased engine efficiency) or lower (less power required).

SUMMARY

Turboprop Range and Endurance

28. Maximum range for a turboprop aircraft is achieved in the following manner:

a. Maximum range is achieved in a cruise climb using MCrpm and an IAS slightly above V_{IMD}.

b. In this cruise climb, TAS varies around a mean value; SAR improves as both drag and SFC fall.

c. Temperatures colder than ISA cause:

(1) The aircraft to fly higher.

(2) A slight increase in TAS.

(3) An increase in SAR.

29. Maximum endurance is achieved at a medium altitude where the engines are throttled to produce an IAS of about V_{IMD}.

CHAPTER 6 - TURBOPROP CRUISE CONTROL

Introduction

1. Due to operational and Air Traffic Control (ATC) requirements, turboprop engine aircraft are more commonly flown lower and faster than the heights and speeds that would be determined solely by range and endurance considerations. Cruise control techniques can achieve time economy at reasonable expense of fuel economy and, under certain conditions of wind velocity, the range may be little affected. A stepped cruise may be flown to approximate the advantages of the cruise climb within any limitations imposed by ATC.

Low Speed Cruises

2. **Cruise Climb**. It was seen in the previous chapter that maximum ranges will be achieved in turboprop aircraft when it is cruise climbed near V_{IMD} using MCrpm. The initial height to start the cruise will depend on weight and temperature. Ideally, the Indicated Airspeed (IAS) should be reduced continuously as weight reduces, but in practice a half-hourly adjustment to the IAS is sufficient. This cruise is referred to in Operating Data Manuals (ODMs) as the constant C_L cruise.

3. **Stepped Cruise**. When ATC procedures do not allow a cruise climb, a stepped cruise will achieve near maximum range. A level cruise is started at the nearest quadrantal height below the ideal stabilizing altitude, using MCrpm. The speed is allowed to build up until the weight has reduced to a point where it is possible to accept the next higher quadrantal or semi-circular height where the IAS will be at, or near, RRS. The procedure is repeated as often as required.

High Speed Cruises

4. When it is not essential to obtain the maximum range for the fuel carried, turboprop aircraft may be cruised at speeds in excess of the best range speed; range, of course, will be less than maximum. However, when flying a high speed cruise, a certain IAS, called V_{NO} (normal operating speed), or indicated Mach number, called M_{NO} (normal operating Mach number), should not be exceeded, since these indicated speeds are imposed by airframe structural limitations.

5. **Cruise Climb**. In the cruise climb at V_{NO} the aircraft is climbed to its stabilizing altitude (which is lower than for the maximum range low speed cruise) with MCrpm set on the engines and the IAS kept at V_{NO}. As weight reduces, the aircraft will gradually climb, the TAS will increase and so will the indicated Mach number (IMN). When M_{NO} is reached, the aircraft is then flown at this IMN. During the ensuing climb, IAS will reduce and, up to the tropopause, so will TAS.

6. **Stepped Cruise**. Where ATC rules prevent a cruise climb, quadrantal heights will have to be flown. When V_{NO} has been reached in the level cruise, the engines would have to be throttled (which is undesirable) to prevent V_{NO} being exceeded, or the aircraft climbed, using MCrpm. In the stepped cruise technique, a level cruise at the nearest quadrantal above the V_{NO} stabilizing altitude is flown using MCrpm. When the speed reaches V_{NO}, the aircraft is climbed to the nearest quadrantal or semi-circular height where the IAS will be less than V_{NO}. Steps are made each time the speed reaches V_{NO} or M_{NO}.

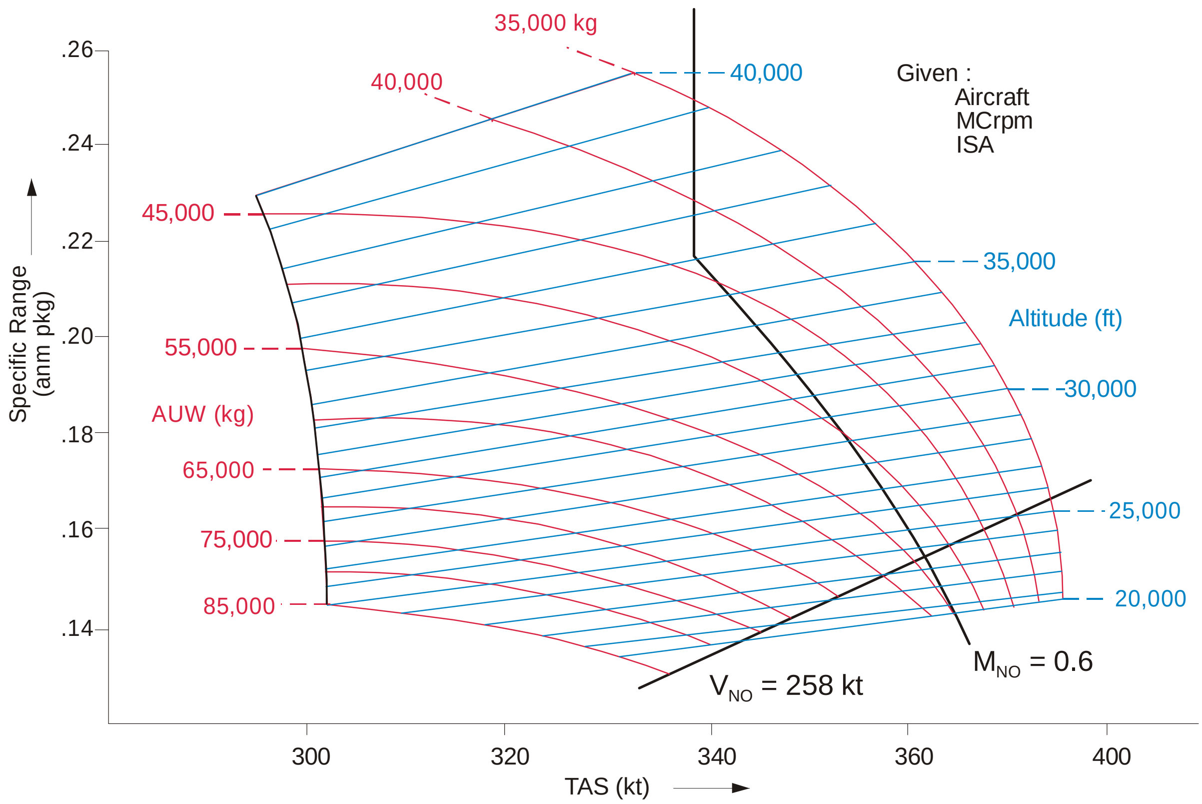
Wind Shear

7. Turboprop performance is considerably affected when there is a wind shear with height and to obtain maximum economy it is better to fly the aircraft lower than optimum altitude in a high speed cruise. Usually, unless the shear is greater than 3 kt per 1000 ft, it is ignored. Where a shear in excess of this is experienced, information in the ODM for the aircraft gives the optimum altitudes to fly.

Cruise Grid

8. Cruise grids similar to that in Fig 1 are sometimes provided in ODMs. These grids are produced from flight trials. It will be noted that it is for a particular temperature condition and a whole set of graphs would be produced for temperature variations above and below ISA conditions. The diagram shows variation in SAR against TAS for changes in weight and associated altitudes. From such diagrams the range performance of a turboprop aircraft can be determined for a large number of cruise conditions.

2-6 Fig 1 Cruise Grid



Climb and Descent

9. **Climb**. The turboprop aircraft, unlike the pure jet aircraft, is not climbed at the maximum rate possible to achieve the best range. The aircraft is climbed at a speed which is a compromise between those which give the best rate of climb at the best range, and is one which gives good fuel ratio. The correct climb speed for a particular aircraft type can be obtained from the appropriate ODM.

10. **Descent**. The principle for the turboprop descent is similar to that for the pure jet, i.e. the cruising altitude is maintained for as long as possible followed by a rapid descent with the engines at minimum power, aiming to arrive over the destination airfield at a height from which a circuit and landing can be made. In practice, the rate of descent may be limited by such considerations as ATC restrictions, passenger comfort and use of sufficient engine rpm to maintain aircraft services.

CHAPTER 7 - PISTON RANGE AND ENDURANCE

FLYING FOR RANGE

Introduction

1. To cover the greatest air distance on the fuel available, the airframe, engine and propeller of a piston engine aircraft must all operate at their most efficient settings. Since an aircraft is rarely flown to its limit of range, the term 'flying for range' is normally used to indicate maximum fuel economy over any flight stage. Although financial economy is desirable, the main requirement is that of maximum fuel reserve at the end of a flight stage for use during possible holds or diversions.

2. For every aircraft there is an Indicated Airspeed (IAS) at which the total drag, and the power required from the engine to counteract this drag, is least. The propeller converts power into thrust (equal to total drag) but always with some conversion losses which are at a minimum when the propeller is operating at its best setting. When these requirements are satisfied, the maximum number of air nautical miles per kg of fuel is obtained.

3. For reasons outlined later in this chapter, the pilot may be unable to achieve this ideal condition, and may not, therefore, be able to obtain maximum fuel economy. However, guidance is given in the Aircrew Manual or Operating Data Manual (ODM) on handling each type of aircraft. Graphs or tables are supplied from which the best speed, height and engine settings for the particular sortie can be determined.

Derivation of Range Speed

4. The greatest fuel economy is not obtained by flying as slowly as possible, since high power is needed to fly at very low air speeds owing to the large amount of lift dependent drag. The level flight speed produced by using minimum power is also unsuitable since the resulting low speed is uneconomical in terms of distance travelled per kg of fuel used.

5. It will be shown that maximum range in level flight at a certain all-up-weight (AUW) is obtained at the IAS at which the total drag is least (provided that the correct engine and propeller settings are used). On a given flight, assuming that the fuel consumption depends on the power setting to obtain the desired IAS, and the time during which this setting is used, then for a given IAS:

$$\text{Fuel used per hour} \propto \text{Drag} \times \text{TAS}$$

Therefore, for a given distance (time) total fuel used can be expressed as:

$$\text{Power} \times \text{Time} \propto \text{Drag} \times \text{TAS} \times \text{Time}$$

$$\therefore \quad \text{Power} \times \text{Time} \propto \text{Drag} \times \text{Distance}$$

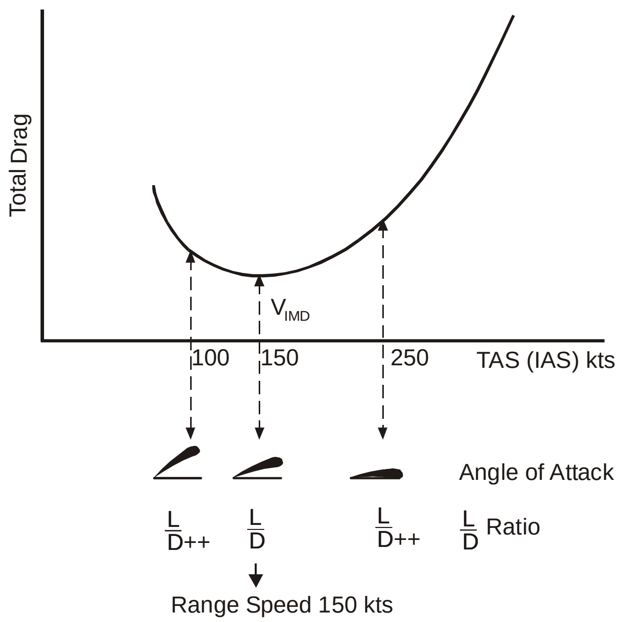
Thus, the less the drag the less the fuel used for a given distance and, also, the greater the distance flown on a fixed quantity of fuel. Since the total drag depends on the TAS there is only one speed at which minimum drag is realized in level flight - this is the speed for maximum range.

Airframe Considerations

6. **Variation of Total Drag**. The total drag for a particular aircraft configuration varies with air speed and angle of attack. These are interrelated as shown in Fig 1 (sea level total drag curve for a hypothetical aircraft, plotted against TAS which, in this case, is also IAS). Minimum total drag is

obtained at an IAS of 150 kt which is, therefore, the optimum range speed for this aircraft. Below this speed the lift dependent drag associated with the high angle of attack increases the total drag; at a higher speed zero lift drag rises steeply, despite the smaller angle of attack.

2-7 Fig 1 Derivation of Range Speed



7. $\frac{\text{Lift}}{\text{Drag}}$ **Ratio.** Assuming that the AUW remains constant, the lift in level flight follows suit. Although the lift remains constant, the manner in which it is obtained changes with speed. At low speeds the angle of attack is increased, drag rises and produces a poor value of $\frac{\text{Lift}}{\text{Drag}}\left(\frac{\text{L}}{\text{D}}\right)$, a ratio which gives a direct indication of the aerodynamic efficiency of the airframe. The ratio also deteriorates at high speeds and, in consequence, maximum range is obtained at the angle of attack giving the best ratio of $\frac{\text{L}}{\text{D}}$. In flight, the pilot is unlikely to know the $\frac{\text{L}}{\text{D}}$ ratio but, by flying at the recommended IAS for best $\frac{\text{L}}{\text{D}}$ ratio the ideal flight condition is achieved. This is V_{IMD}.

8. **Recommended Range Speed**. In practice, a piston-engine aircraft is flown for range at a speed slightly faster than V_{IMD} for two reasons:

a. Because of the shallow slope of the total drag curve in the region of minimum drag, the variation in $\frac{\text{TAS}}{\text{Power}}$ (inversely $\propto$ drag, see para 5) at speeds close to V_{IMD} is negligible and a slightly higher speed will allow a faster flight with no appreciable loss of range.

b. When flying at V_{IMD}, any turbulence or manoeuvre will result in loss of lift and height. This loss can only be regained by the application of more power and consequent fuel wastage. A slightly higher speed ensures a margin for such events.

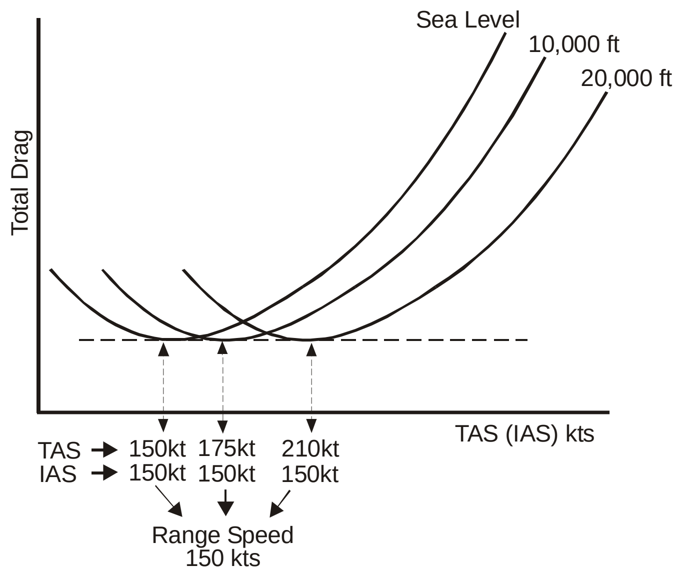
Piston-engine aircraft are, therefore, normally flown at a recommended range speed which is some 10% higher than V_{IMD}.

9. **Effect of Altitude on Drag**. By plotting the total drag against TAS for various altitudes, curves similar to those in Fig 2 are obtained. For the comparatively low performance of piston-engine aircraft (ie neglecting compressibility drag) it can be accepted that for a given IAS, the drag remains the same at all altitudes, although the TAS corresponding to this IAS increases progressively. As far as the airframe is concerned, the recommended range IAS holds good at any altitude and, while the higher TAS results in more air miles for a given time at high altitude than at sea level, this advantage is not gained without cost. The power required for a given IAS increases with altitude. From para 5, it can be deduced that:

$$\text{Power Required} \propto \text{Total Drag} \times \text{TAS}$$

Accordingly, since the drag at the equivalent IAS remains constant with altitude, any increase in TAS can only be obtained by increasing the power output.

2-7 Fig 2 Variation of Total Drag with Altitude



10. **Effect of Wind.** For any aircraft flying at the recommended range speed, the greater range for the fuel available will be achieved by flying downwind. However, it is rare that the wind is blowing consistently in the direction required. During an 'out and home' flight in given wind conditions, the total fuel used is greater than that used on the same flight in still air. This is due to the greater number of air miles flown. This is explained in Table 1 by reference to the 'out and home' flight of a hypothetical aircraft with the following operating data:

TAS at Range IAS		130 kt
Flight Distance		2 × 100 nm
Fuel Consumption (Range Speed)		150 kg/hr
Flight Conditions:	a.	Still Air
	b.	50 kt Headwind
	c.	50 kt Tailwind

Table 1 Effect of Wind

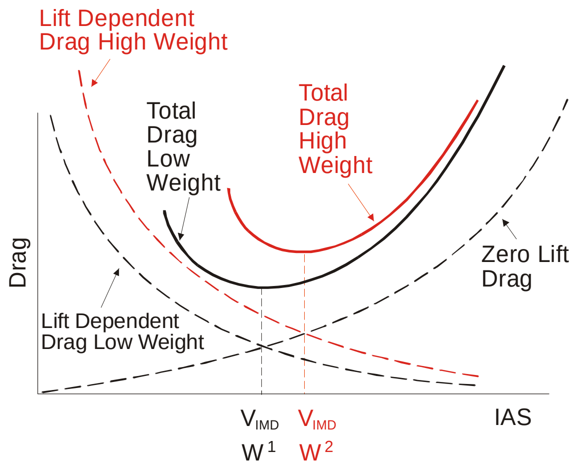
a. Still Air (out and home)	
Flight Distance	200 nm total
Ground Speed	130 kt
Flight Time	92 mins (1.54 hrs)
Fuel Used	231 kg
Air Miles Covered	200 nm (130 × 1.54)
b. Into 50 kt Headwind (Outbound)	
Flight Distance	100 nm
Ground Speed	80 kt
Flight Time	75 mins (1.25 hrs)
Fuel Used	187.5 kg
Air Miles Covered	162.5 nm (130 × 1.25)
c. In 50 kt Tailwind (Inbound)	
Flight Distance	100 nm
Ground Speed	180 kt
Flight Time	33 mins (0.55 hrs)
Fuel Used	82.5 kg
Air Miles Covered	71.5 nm (130 × 0.55)

Thus, 234 air miles will be flown on the round trip in windy weather against 200 air miles in still conditions with the attendant increase in fuel consumption. In this simple example the added distance translates into the consumption of an additional 39 kg of fuel.

11. **Practical Adjustments for Wind.** Nevertheless, the difference due to wind is quite small. Unless a headwind consistently exceeds 25% of the TAS or a tailwind 33% of TAS, no adjustment to range speed should be made. If the Aircrew Manual gives no advice to the contrary and range is critical, it may be beneficial to increase the recommended range speed by 5% in headwinds above 50 kt and reduce the speed by 5% in tailwinds over 50 kt. Any greater increase in speed also increases drag and the fuel consumption disproportionately. Any greater decrease brings manoeuvrability difficulties which are described in para 8b.

12. **Effect of Changes in All-up Weight.** An increase in AUW requires a corresponding increase in C_L to maintain level flight. This imposes a similar change in the lift dependent drag. Since this component of total drag varies with C_L^2 (or weight^2) (see Volume 1, Chapter 5), a change in AUW will alter the point where the lift dependent and zero lift drag curves cross and so change V_{IMD}. Thus, an increase in AUW reduces the range and, more importantly, any reduction in AUW will improve range.

2-7 Fig 3 Effect of Weight on Drag Curves



Summary of Airframe Considerations

13. Airframe considerations can be summarized as follows:

a. Maximum range is obtained at the IAS for minimum drag.

b. Total drag at a given IAS is constant at all altitudes.

c. The TAS corresponding to the range IAS increases with altitude.

d. The power required for a given IAS increases with altitude.

e. The IAS is the pilot's guide to the attitude giving the best angle of attack and $\frac{L}{D}$ ratio.

Engine Considerations

14. **Power**. The power output of a piston engine is normally measured in kilowatts (kW) although references will still be seen to the earlier term, brake-horse-power (BHP) (1 BHP = 746 watts). The power delivered depends upon the weight of fuel consumed. This weight is largely dependent on:

a. The pressure of fuel/air mixture fed to the engine, known as manifold air pressure (MAP).

b. The rate at which this mixture flows into the engine. This rate is governed by the crankshaft speed (RPM).

15. **Full Throttle Height**. With increasing altitude the power available falls steadily. At the same time, power required has been rising steadily with altitude owing to the factors outlined in para 9. Eventually, an altitude is reached at which the power available is just equal to the power required to give range IAS. For the set engine and speed conditions, this is called the full-throttle height and produces the best range.

16. **Fuel Economy**. Fuel consumption may be measured in 2 ways:

a. **Gross Fuel Consumption (GFC) (kg fuel used per hour)**. The GFC varies with the power output and, therefore, for a fixed-pitch propeller, with RPM. The case of the constant-speed propeller is more involved, since the power output depends on both throttle and propeller pitch setting. In general, for a given throttle setting, the power output is proportional to the RPM. A certain power output may be obtained, for instance, by using:

(1) High RPM, low MAP and, hence, a small throttle opening.

(2) Medium RPM and MAP giving a larger throttle opening.

(3) Low RPM, high MAP and full throttle.

The best fuel economy would result from the last case owing to the reduced number of engine working strokes per minute, associated with the higher volumetric efficiency gained from a high MAP. Other mechanical considerations, such as a possibly improved propeller efficiency at low RPM, add to this effect.

b. **Specific Fuel Consumption (SFC) (kg fuel used per kW of engine power per hour)**. If the power output of an engine were always constant then its efficiency could be measured purely by observing its GFC. However, the power output of an aircraft varies considerably as, for example, when the weight, and hence the drag, reduce in flight. So, another term, which relates the power output to the fuel used, has to be found. The term for this ratio is specific fuel consumption (SFC) and, for a piston engine, is given by:

$$\text{SFC} = \frac{\text{GFC}}{\text{Power}}$$

SFC is best at a minimum and is dependent upon the following factors:

(1) **MAP and RPM.** As mentioned in para 16a, there is a choice of MAP and RPM settings which can provide the same power output. High MAP and low RPM give minimum SFC because:

(a) Friction and inertia losses are low.

(b) Engine driven services absorb minimum power.

(c) Volumetric efficiency is better.

(2) **Altitude**. As altitude increases up to full throttle height, the SFC falls (improves) for three reasons:

(a) Progressively less throttling of the engine is needed.

(b) Progressively less useful power is expended in exhausting the spent mixture against the ambient air pressure.

(c) The colder air contributes to a greater rise in temperature within the engine, which is a measure of thermal efficiency.

(3) **Temperature.** In cold air, at a particular altitude, the RPM may be reduced to provide the power required because:

(a) The power available from the engine is greater.

(b) The power required to propel the airframe in cold air is less.

The colder the temperature, therefore, the lower will be the SFC because the power absorbed (wasted) at the lower temperature is reduced and the thermal efficiency is increased.

(4) **Other Factors.** The use of carburettor or induction warm air (decreased air density) and/or selection of induction air filtration (because the change in ram effect decreases MAP) will both increase (worsen) SFC.

Summary of Engine Considerations

17. A piston aero-engine is able to produce the most efficient power output required in flight under the following conditions:

a. Not more than maximum weak mixture MAP is set.

b. Minimum RPM used.

c. Aircraft at full throttle height.

d. Carburettor intake or induction air to 'COLD' and air filter 'OUT'.

Airframe/Engine Combination

18. **Specific Air Range.** Specific Air Range (SAR) is defined as air distance travelled per unit quantity of fuel, ie air nm per kg of fuel. Thus:

$$\text{SAR} = \frac{\text{distance travelled}}{\text{fuel used}}$$

Clearly, the greater the SAR the greater the range. Now, if the above expression is divided top and bottom by 'time', then:

$$\text{SAR} = \frac{\text{distance travelled}}{\text{time}} \times \frac{\text{time}}{\text{fuel used}} = \frac{\text{TAS}}{\text{GFC}}.$$

From para 16b , GFC = Power × SFC, so:

$$\text{SAR} = \frac{\text{TAS}}{\text{Power}} \times \frac{1}{\text{SFC}}$$

Therefore, for SAR to be greatest (best range), at a particular weight, in still air, a piston-engine aircraft should be flown for a maximum product of airframe efficiency $\left(\frac{\text{TAS}}{\text{Power}}\right)$ and engine efficiency $\left(\frac{1}{\text{SFC}}\right)$.

19. **Design Features**. By calculating the total drag at the minimum drag speed the designer can determine the power (measured in kW) required to obtain this speed. A propeller is then selected which absorbs least power from the engine in producing the power required but, even under ideal conditions, the propeller efficiency rarely exceeds about 80%. Next, the designer selects an engine which gives:

a. Sufficient power for the take-off and the flight performance required.

b. The required power for range speed, when using full throttle and minimum RPM.

c. The lowest SFC when using range power.

The airframe and engine are carefully matched to provide the best combination possible. Nevertheless, a compromise may have to be made on certain aircraft with specific roles.

20. Best Altitude for Maximum Range. The two factors concerned with the determination of best altitude to obtain maximum range are:

a. The power required at range IAS.

b. The power available, at various altitudes, using full throttle and minimum RPM.

Since, with increasing altitude the power available falls, an altitude is found at which the power available is just equal to the power required to give range IAS, as explained in para 15 - this is the full throttle height for the power required to give the recommended range speed using minimum RPM and weak mixture and is called the Range Height.

21. **Effect on Range of Varying the Cruising Altitude**.

a. **Flight above Recommended Altitude.** As altitude increases above the maximum weak mixture MAP full throttle height for maximum range, the power required can only be achieved by increasing RPM. In this case, the SFC will increase (deteriorate) because more useful power is absorbed within the engine.

b. **Flight below Recommended Altitude**. Descending from the recommended altitude, the power available increases. Power required, on the other hand, falls steadily resulting in a growing power surplus. The pilot may accept the higher speed obtained (together with the increased drag and deteriorating $\frac{\text{Lift}}{\text{Drag}}$ ratio) or close the throttle to maintain the recommended range IAS. The first is much more acceptable since, not only is a higher cruising speed obtained, but the loss in range experienced is far less than that which would result from running the engine for a longer period at a severely throttled, and therefore uneconomical, setting.

Procedure to Obtain Maximum Fuel Economy

22. **Summarized Procedure.** The procedure for setting a piston-engine aircraft to fly for maximum range by ensuring maximum fuel economy may be summarized as follows:

a. Climb to the recommended altitude as described in the Aircrew Manual.

b. Set the throttle fully open.

c. Reduce the RPM until the recommended range IAS is obtained.

d. If necessary, adjust the speed for prevailing wind conditions.

e. Use weak mixture whenever possible.

f. Use carburettor or induction cold air with filtration off.

g. Reduce drag to a minimum by trimming the aircraft correctly after ensuring that flaps, undercarriage or spoilers etc are retracted.

FLYING FOR ENDURANCE

Introduction

23. To keep an aircraft airborne for as long as possible on the fuel available, the IAS should be that at which the engine consumes least fuel. Endurance (measured in hours) is obtained by dividing the fuel available (in kilograms) by the gross fuel consumption (GFC) (in kilograms per hour). Since the GFC is assumed to be proportional to the power output, maximum fuel economy is obtained by using the level flight speed corresponding to the lowest power output of the engine.

24. This speed is determined by the aerodynamic characteristics of the airframe, the SFC of the engine at the power output required, the propeller efficiency and the AUW. In practice, the speed may be modified by handling considerations.

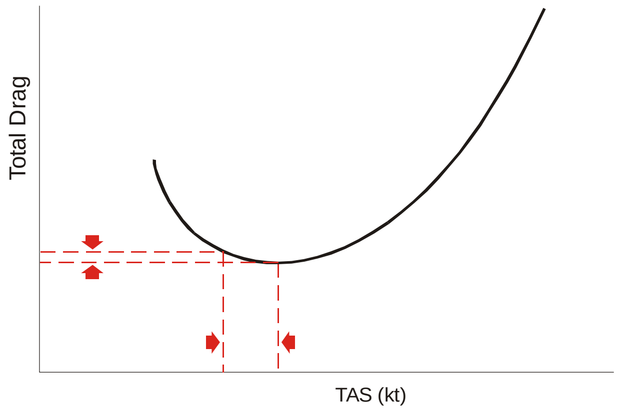
25. The actual endurance, being a function of AUW, depends on the altitude, the outside air temperature and the external condition of the aircraft itself. Wind has no effect on endurance.

Derivation of Endurance Speed

26. By gradually reducing the power during level flight, a speed and power is eventually reached at which the aircraft can just maintain a height at which there is no margin of control for handling purposes. If the total drag and TAS are known, this minimum power may easily be calculated since power = total drag × TAS. Assuming the fuel consumption in kilograms per hour to be proportional to the power output, the speed for minimum power gives maximum endurance.

27. It is interesting to note that at this low speed the drag is greater than when flying at range speed (which is higher), while the power output is less. This is because the power depends on two factors (drag and airspeed) which alter when power is reduced, but not in proportion. It can be seen from Fig 4 that an appreciable reduction can be made in the speed for minimum drag (ie the ideal range speed) without incurring any large increase in drag. The overall effect is a reduction in the product of the two factors, although drag has increased and speed decreased.

2-7 Fig 4 Variation of Total Drag with Speed

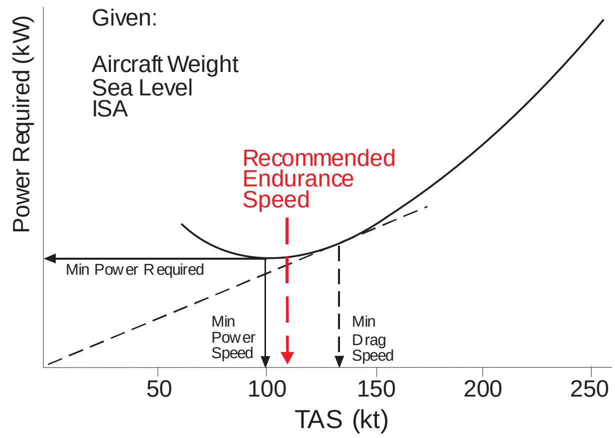


28. Although the theory is sound, flight at the speed for minimum power is less than practicable since all ability to manoeuvre disappears and engine handling problems arise. For these reasons a margin of speed is added to improve manoeuvrability, resulting in a slightly higher recommended endurance speed which should be maintained within plus or minus five knots. Endurance speed may be defined as:

> The level flight speed produced by the use of minimum power, plus a margin for control and manoeuvre.

29. From the power required curve for a typical aircraft shown at Fig 5, the best endurance speed would appear to be 100 kt. However, the recommended endurance speed would be some 10 kt above this speed.

2-7 Fig 5 Derivation of Endurance Speed



30. It should be recognized that the recommended IAS for endurance is considerably higher than the minimum possible flight IAS. There is a large rate of drag increase at the lower speeds as shown in Fig 4. To overcome this drag, flight at very low airspeeds entails the use of power much higher than that required for maximum endurance. Indeed, level flight is possible on some aircraft at speeds well below the power-off stalling speed.

Factors Affecting Endurance

31 **Variation in AUW.** Since speed changes are obtained by varying the power, it follows that less power is required for the reduced IAS associated with lower weight and that fuel consumption will be proportional to the weight of the aircraft.. As a rule of thumb, a 10% reduction in AUW increases endurance by about 7%.

32. **Effect of Altitude**. In level flight, maximum endurance decreases with increasing altitude. Assuming the fuel used per hour to be proportional to the power used, this can be expressed in the following manner:

$$\text{Power Used} = \text{total drag} \times \text{TAS}$$

For a given IAS the drag is the same at all altitudes but the corresponding TAS increases with height. It follows that the power required to maintain endurance speed increases with altitude, as does the fuel consumption, and endurance is therefore reduced. Although maximum endurance is obtained at sea level, flight at this height is impractical particularly because safety height and other airspace rules have to be observed. Nevertheless, the lowest practical altitude possible is recommended for endurance.

33. **Effect of Additional Drag.** Since the recommended speed must be maintained to provide adequate control and manoeuvrability, more power must be used to counteract extra drag arising from the carriage of external stores. Endurance will be improved by any measures taken to ensure that the aircraft is aerodynamically clean; however, if such an increase in drag must be accepted, the best IAS is obtainable only at a higher power setting, thus reducing endurance.

34. **Effect of Temperature Variation**. At high atmospheric temperatures the air density, the weight of the induction charge and, therefore, the power output, are all reduced. To maintain a given IAS, an increase in MAP and/or RPM is necessary and there may be a considerable loss of endurance.

Technique for Endurance Flying

35. The technique for endurance flying is as follows:

a. Fly at the lowest, safe, practicable altitude.

b. Adjust the throttle and RPM to give the recommended endurance speed for the AUW and maintain this speed within 5 knots.

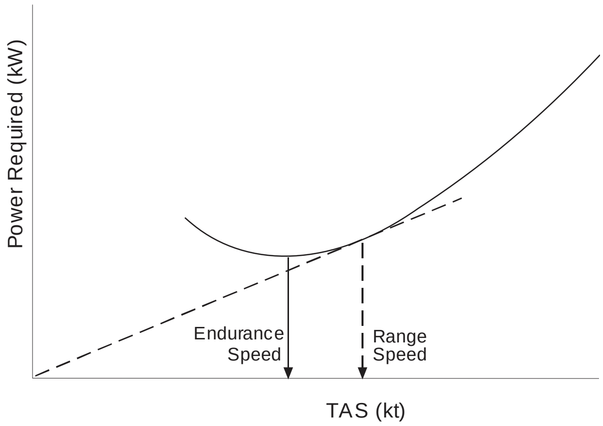
c. Use weak mixture, carburettor or injector cold air and air filter out of operation.

d. Trim the aircraft correctly and reduce all possible drag such as that caused by flaps, stores etc.

Comparison of Range and Endurance Speeds

36. Since endurance implies minimum power and range implies minimum drag, comparison of the two recommended speed is not easy. However, both speeds may be identified by referring to Fig 6 which plots power required against TAS for a given aircraft. The endurance speed, ie that requiring minimum power, is found at the base of the power curve. The range speed is that giving the highest ratio of $\frac{\text{TAS}}{\text{Power}}$ and may be found by drawing a tangent from the origin of the graph to the power curve. The range speed occurs at the point of tangency.

2-7 Fig 6 Relationship between Range and Endurance Speeds



37. It should be remembered that the recommended speeds for an aircraft are unlikely to match the ideal because they take account of individual aircraft characteristics, manoeuvrability requirements, altitude considerations and the like. That said, in general, the endurance speed is about 80% of the range speed.

SUMMARY

Piston Engine Range and Endurance Flying

38. Table 2 gives a summary of the factors affecting range and endurance in piston-engine aircraft.

Table 2 Summary of Range and Endurance Flying - Piston-engine Aircraft

RANGE			ENDURANCE	
EFFECT OF	***ON RANGE SPEED***	***ON RANGE OBTAINED***	***ENDURANCE SPEED***	***ENDURANCE OBTAINED***
ALTITUDE	No effect.	Maximum at Recommended Range Height. Reduced above and below.	No effect.	Maximum at Sea Level.
WIND	Increase 5% in headwinds and decrease 5% in tailwinds over 50 kt.	Reduced economy on round flight into and downwind compared with round flight in still air.	No effect.	No effect.
ADDITIONAL DRAG	Reduced in cases of large drag increment.	Reduced in proportion to drag increase.	Speed must be maintained, even if extra power is necessary.	Reduced in proportion to additional drag.
ATMOSPHERIC TEMPERATURE	No effect.	Range reduced in increased outside temperatures but offset by increased TAS owing to reduction in density.	No effect.	Endurance reduced in increased outside temperatures.
VARIATION IN ALL-UP WEIGHT	The lower the AUW, the lower the Range Speed.	Range increased as AUW is reduced.	The lower the AUW, the lower the Endurance Speed.	Endurance increased as AUW is reduced.

CHAPTER 8 - DENSITY ALTITUDE EFFECTS

Air Density

1. **Density Altitude**. The International Standard Atmosphere (ISA) (see Volume 1, Chapter 1) includes in its definition a mean sea level air density of 1.225 kg/m^3. Since temperature as well as pressure affects air density, and therefore aircraft performance, a suitable method of measuring air density is required. Although engineers express air density in terms of kg/m^3, for aircraft operations it is more convenient to express it as density altitude. Density altitude is defined as that height (above or below mean sea level) in the standard atmosphere to which the actual density at any particular point corresponds. For standard conditions of temperature and pressure, density altitude is the same as pressure altitude.

2. **Calculation of Density Altitude.** Density altitude can be calculated by several methods which include:

a. Using the density altitude graph (Fig 1).

b. Using a simple computer which is normally provided by aircraft manufacturers for specific aircraft types.

c. Using the simple formula:

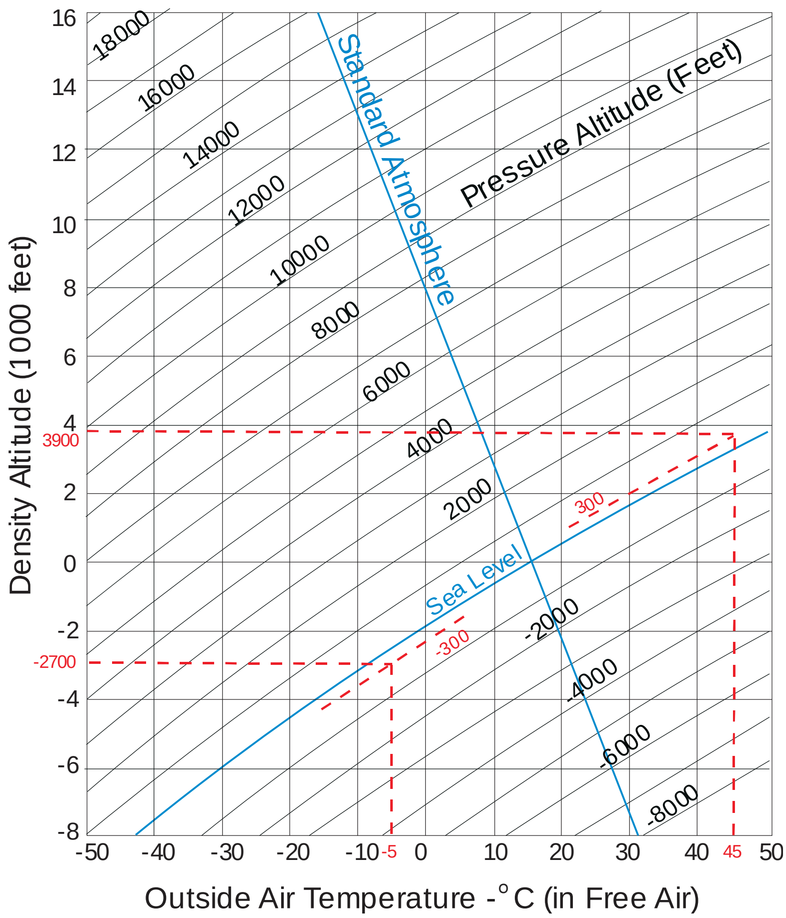
Density Altitude = Pressure altitude $\pm$ (120t),

where t is the difference between local air temperature at pressure altitude and the standard temperature for the same pressure altitude. If the air temperature is higher than standard, then (120t) is added to pressure altitude, if it is lower, subtracted.

Variations in Surface Density

3. **Low Density**. When atmospheric pressure is low or air temperature is high (or a combination of both), the density of air is reduced and the density altitude at sea level becomes high. For example, in the Persian Gulf on a day when surface temperature is 45 ºC and sea level pressure 1003 mb, pressure altitude at the surface is +300 ft (10 mb $\times$ 30 ft per mb (see Volume 1, Chapter 1)). Entering the density altitude graph (Fig 1) at +300 ft pressure altitude and correcting for 45 ºC OAT, gives a density altitude of 3,900 ft at the surface.

2-8 Fig 1 Density Altitude Graph



4. **High Density**. When low air temperatures are combined with high atmospheric pressure, the density of air increases; this gives rise to low density altitude at sea level. For example, on a winter's day in Scotland where the surface temperature is –5 ºC and sea level pressure is 1023 mb, then pressure altitude at the surface will be -300 ft. Using the chart at Fig 1 shows that this pressure altitude, corrected for OAT of –5 ºC, gives a density altitude of –2,700 ft at the surface.

Effects of Density Altitude on Performance

5. **Low Density Altitude**. The effects of low density altitudes on aircraft handling and performance are entirely advantageous. At –3,000 ft density altitude, less collective pitch will be required than at sea level in the standard atmosphere and less power will be required to drive the rotor. Because of the increased performance, a careful check must be kept on payload to ensure maximum AUW is not exceeded.

6. **High Density Altitude**. The effects of high density altitudes on aircraft handling and performance must be fully appreciated by pilots. More collective pitch will be required at high density altitudes than at sea level in the standard atmosphere, and more power will be required to overcome the extra rotor drag resulting from the increase in pitch. The engine power may well be limited under high temperature conditions so that payload may have to be reduced to maintain an adequate performance margin.

CHAPTER 9 - INTRODUCTION TO SCHEDULED PERFORMANCE

Introduction

1. Scheduled performance is the calculated performance of an aircraft that, when used for planning purposes, delivers an acceptable level of safety. It comprises the certified performance data scheduled in the Aircraft Flight Manual (or equivalent) and the regulations that govern its use. The principle conditions of these regulations are:

a. Aircraft are required to be certified to ensure that they can proceed safely from their departure airfield to a destination airfield carrying sufficient fuel reserves to divert to a suitable alternate airfield should a landing at destination not be possible.

b. Certain categories of aircraft are required to be capable of sustaining flight following an engine failure such that a return to the point of departure, diversion to a suitable take-off alternate or continued flight to the nominated destination is possible.

While military aircraft are not subject to civil regulations (see Para 7b), it has been directed that wherever possible, performance criteria that at least match civil regulations, should be applied to UK military aircraft (see Para 7c). Each aircraft type is certified to a particular regulation in force at the time, and this regulation remains the basis for scheduled performance throughout the life of the type (see Para 22). Thus the MOD has aircraft with scheduled performance criteria based on different regulations. The following paragraphs describe the regulations and how they are applied.

Regulators

2. **International Civil Aviation Organization (ICAO).** On 7 December 1944, the majority of the world's nations became signatories to the "Chicago Convention", the aim of which was to assure the safe, orderly and economic development of air transport. ICAO sets out, in the terms of the convention, the rules, regulations and requirements to which each signatory must adhere.

3. **Civil Aviation Authority (CAA).** The Civil Aviation Act 1982 is the UK’s means of discharging its ICAO responsibilities. This duty is undertaken by the CAA and the Department for Transport (DfT).

4. **Joint Aviation Authority (JAA).** The JAA was an associated body representing the civil aviation regulatory authorities of a number of European States who had agreed to co-operate in developing and implementing common safety regulatory standards and procedures. It was not strictly a regulatory body, regulation being achieved through the member authorities (CAA for UK), but regulations were issued in the form of Joint Aviation Requirements (JARs).

5. **European Aviation Safety Agency (EASA).** In Sep 2003, EASA took over responsibility for the airworthiness and environmental certification of all aeronautical products, parts and appliances designed, manufactured, maintained or used by persons under the regulatory oversight of European Union (EU) Member States. All aircraft type certificates are therefore issued by EASA and are valid throughout the EU. The establishment of EASA created a Europe wide regulatory authority which has absorbed most functions of the JAA. EASA devolves the administration of the certification process to national aviation authorities. In the UK these authorities are the DfT and the CAA, who have historically fulfilled this role.

6. **Military Aviation Authority (MAA).** The responsibility for the certification of new military air systems rests with the MAA which has full oversight of all Defence aviation activity (see Para 7c).

Regulations

7. **Airworthiness Regulations.** Aircraft may be put into one of two groups; EASA regulated aircraft and nationally regulated aircraft. Thus, both EASA and the CAA publish regulations for the certification and operation of aircraft. Regulations for UK military aircraft are also issued by the MAA.

a. EASA publishes regulations through documents known as EU-Ops which are further explained in Certification Specifications (CSs). The EASA regulations do not apply to military aircraft unless the aircraft is of a type for which a design standard has been adopted by EASA. JARs (see Para 4) were issued before the establishment of EASA and have been transformed into EU-Ops.

b. The CAA publishes regulations in the CAP 393 Air Navigation: The Order and Regulations, generally referred to as the Air Navigation Order (ANO). The ANO is law and provides legal definitions. It is comprised of Articles, which are individual points of law and Schedules which are implementation lists for Articles. Article 252 of the ANO states that, with some exceptions, the ANO does not apply to military aircraft. The ANO defines the meaning of military aircraft in Article 255 as:

i. The naval, military or air force aircraft of any country.

ii. Any aircraft being constructed for the naval, military or air force of any country under a contract entered into by the Secretary of State.

iii. Any aircraft for which there is in force a certificate issued by the Secretary of State that the aircraft is to be treated for the purposes of this Order as a military aircraft.

c. UK Military Flying is regulated by the MAA. MAA01, The Military Aviation Authority Regulatory Policy, states that:

> The authority to operate and regulate registered UK military aircraft is vested in the Secretary of State (SofS). Notwithstanding the fact that the majority of provisions of the ANO do not apply to military aircraft, the Crown could be liable in common law if it were to operate its aircraft negligently, and cause injury or damage to property. SofS' instruction to Defence is that where it can rely on exemptions or derogations from either domestic or international law, it is to introduce standards and management arrangements that produce outcomes that are, so far as is reasonably practicable, at least as good as those required by legislation.

Thus military aircraft are provided with scheduled performance data, and for normal operations adopt the equivalent civilian regulations. Where for reasons of military necessity these regulations are relaxed, the authorization to do so is held at an appropriate level.

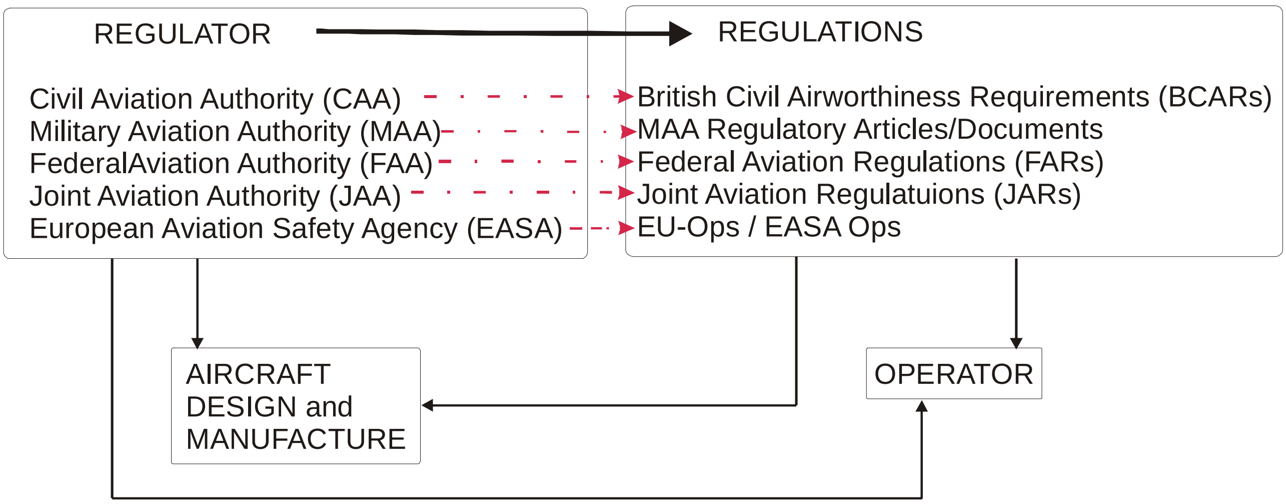
8. The CAA also issues:

a. British Civil Airworthiness Requirements (BCARs)(CAP 553) which are not law in themselves, but do provide information on how to comply with UK requirements, which are law. BCARs are split into two volumes, one (Section A) relating to aircraft types where the CAA has primary type responsibility, and the other (Section B) relates to aircraft types where EASA has primary type responsibility. BCARs detail the process that an aircraft has to go through to receive a Certificate of Airworthiness from the CAA. The process involves a series of flight tests and the determination of a performance group (see Para 17). Where EASA has primary responsibility for type certification, a performance category for an aircraft is also determined.

b. CAP 747, Mandatory Requirements for Airworthiness is the means by which airworthiness requirements made mandatory by the CAA are notified, and gives further information on EASA aircraft and non-EASA aircraft.

9. Fig 1 diagramatically summarizes the regulatory process. The appropriate Regulator determines the Regulations which the aircraft designer and manufacturer follow to receive certification. The operator adheres to the Regulations and Directives of the Regulator, with some exceptions in the case of military aircraft (see Paras 11 to 13).

2-9 Fig 1 The Regulatory Process



Performance Planning

10. Before a flight, aircrew have to make performance calculations to ensure that their flight is achievable within the regulations that their aircraft has been certified for. Operating Data Manuals (ODM) (see Volume 2, Chapter 16) or their electronic equivalent use Performance Graphs and Charts to allow a maximum take off weight to be calculated for any given flight. These graphs and charts are derived in the following way:

a. **Expected Performance**. Wind tunnel tests and computer predictions are used to check if an aircraft meets its design specification in terms of its expected performance.

b. **Measured Performance.** Measured performance is the average performance of an aeroplane or group of aeroplanes being tested by an acceptable method in specified conditions.

c. **Gross Performance**. The gross performance is such that the performance of any aeroplane of the type, measured at any time, is at least as likely to exceed Gross Performance as not.

d. **Net Performance.** This represents the Gross Performance factored by the amount considered necessary to allow for various contingencies. These contingencies take into account:

i. Unavoidable variations in piloting technique (i.e. temporary below average performance).

ii. An aircraft performing below the fleet average.

iii. Unexpected need to manoeuvre from the planned flight profile (e.g. to avoid birds).

e. **Safety Factor.** The difference between Net and Gross performance (the safety factor) may vary from 10% to 40% depending on the stage of flight and the statistical chance of an accident; 1 in 10,000,000 is considered to be an acceptable risk. It is a legal requirement that all performance planning should be conducted to net performance considerations with the exception of Military Operating Standards (MOS) data (Para 13).

11. **Normal Operating Standard (NOS)**. When military flights operate within the bounds of Net Performance as laid down by the relevant authority they are considered to be operating to NOS. The aircraft ODM contains data relating to NOS.

12. **Reduced Operating Standard (ROS).** Operators may in certain circumstances permit a flight that requires a reduction in the safety factor that net performance applies. The aircraft ODM may contain data relating to ROS.

13. **Military Operating Standard (MOS)**. There may be instances when maintaining NOS or ROS safety measures means that a military task is not possible. In certain circumstances, and only for operational reasons, Senior Commanders may authorise specific tasks to be conducted using a higher lever of risk than that for the NOS. In the extreme case, the highest level of risk will be where the aircraft is authorized to operate to Gross Performance limits alone, with no additional safety factorisation. Depending upon circumstances, ROS may be appropriate instead of MOS; here the level of acceptable risk would be factored to lie somewhere between Net and Gross values. The aircraft ODM may contain data relating to MOS.

14. **Performance Data.** Performance data, presented mainly in graphical form, is derived from a statistical analysis of test flights made to determine Gross Performance for each of the four stages of flight under varying conditions of weight, temperature, airfield altitude, runway slope and wind. Analysis is made for one or two power unit inoperative cases. Gross Performance is then suitably factored to allow for contingencies which cannot be accounted for operationally.

15. **The Performance Plan.** The performance plan is part of any flight plan and is concerned solely with the safety of the aeroplane. It determines that the aircraft can:

   a. Take off safely within the runway length available and with enough distance remaining to abandon a take off if required.

   b. Climb, after take off, to a safe height missing any obstacles by a prescribed horizontal or vertical margin.

   c. Carry enough fuel to transit to the destination with sufficient fuel reserves to divert to a suitable alternate airfield should a landing at destination not be possible. For long distance flights the fuel load may need to be increased by an additional amount, known as contingency fuel, to cater for unforeseen events such as weather avoidance.

   d. Descend and land at the destination or alternate aerodrome coming to a complete halt within the runway distance available.

16. All of these factors will determine the maximum All Up Weight (AUW) at take off.

Performance Groups

17. It is the duty of the licensing authority (EASA or CAA) to ensure that no aeroplane is granted a Certificate of Airworthiness (C of A) until it has been shown that it can satisfy the conditions laid down in the relevant regulations.

Demonstration of performance capability allows the certification of an aeroplane type into one of several Performance Groups. Whilst military aircraft are not subject to the civil regulations, it is intended that wherever possible, military aircraft should be operated to at least the same performance standards as civil aircraft (see Para 7c).

18. Notwithstanding the above statement, situations of such urgency may arise that, in order to utilize medium sized and large military aeroplanes to their maximum potential, reduced safety margins are both necessary and acceptable (see Paras 11 to 13). Combat aeroplanes are not operated to any scheduled performance requirements because to do so would reduce their operational efficiency. An element of risk is acceptable in the operation of these aircraft and it is an MAA responsibility to define and apply appropriate safety margins.

19. The EASA performance groups are described in the following sub-paragraphs and the performance requirements that each aircraft are required to meet are detailed in the relevant EASA CS document. CS 25 details the requirements for large aircraft which are classified into either Performance Group A or C. CS 23 details the requirements for normal, utility, aerobatic and commuter category aircraft which are classified into Performance Group B.

a. **Performance Group A.** Multi-engine aeroplanes powered by turbo propeller engines with a maximum approved passenger seating configuration of more than 9 or a maximum take-of mass exceeding 5,700 kg (12,500 lb), and all multi-engine turbojet powered aeroplanes. Aircraft in this class can sustain an engine failure in any phase of flight between the commencement of the take off run and the end of the landing run without diminishing safety below an acceptable level. A forced landing should not be necessary because of performance considerations.

b. **Performance Group B.** Propeller driven aeroplanes with a maximum approved passenger seating configuration of 9 or less and a maximum take-off mass of 5,700 kg (12,500 lb) or less. Any twin engined aircraft in this class that cannot attain the minimum climb standards as specified in CS23 (see para 20) shall be treated as a single engined aeroplane.

c. **Performance Group C.** Aeroplanes powered by reciprocating engines with a maximum approved passenger seating configuration of more than nine or a maximum take-off mass exceeding 5,700 kg (12,500 lb). Aircraft in this class are able to operate from contaminated surfaces and are able to suffer an engine failure in any phase of flight without endangering the aeroplane.

d. **Unclassified Aircraft.** Where full compliance with the requirements of the appropriate regulations cannot be shown due to specific design characteristics (e.g. supersonic aeroplanes or seaplanes), the operator shall apply approved performance standards that ensure a level of safety equivalent to that of the appropriate regulations. The manner in which a particular aircraft type is to be flown, the purpose for which it may be used and the absolute maximum TOW are specified in the Flight manual and ODM. These are issued as part of the Certificate of Airworthiness.

20. The CAA also publishes the requirements for aircraft performance groups in Schedule 1 of Regulation 8 in the ANO. The ANO gives data for performance groups A, B, C, D, E, F, X and Z. Groups A and B are combined and have the same requirements. All aircraft of relevance to this chapter fit into either group A or C.

a. **Performance Group A.** Aeroplanes with a maximum certificated take off weight exceeding 5,700 kg (12,500 lb) and with a performance level such that at whatever time a power unit fails, a forced landing should not be necessary.

b. **Performance Group C**. Aeroplanes with a maximum certificated take off weight not exceeding 5,700 kg (12,500 lb) and with a performance level such that a forced landing should not be necessary if an engine fails after take off and initial climb.

Performance Planning Considerations

21. It is important to note that certification regulations are subject to periodic revision in the light of experience and improved techniques, but it is not always possible to modify aeroplane performance capabilities to comply with such amendments. When dealing with a specific aeroplane type, therefore, reference must be made to the ODM to ascertain the date of the regulations to which the aeroplane is certificated. For example, the Royal Air Force operates the King Air B200 Classic and King Air B200 GT aircraft, which are basically the same aircraft type with different flight deck instruments. They also have different engines, but with the same power output. The King Air B200 Classic operates to CAA regulations and is placed in Performance Group C. The King Air B200 GT, which was introduced into service some four years after the King Air B200 Classic, operates to EASA regulations and is placed in Performance Group B. Once an aircraft has been certified, it cannot be re-certified. As a result, the RAF King Air aircraft types operate to different performance group requirements. It is still mandatory, however, to comply with the latest Air Navigation Regulations.

22. For the purpose of performance planning a flight is divided into four stages:

a. **Take-off.** The take-off stage is from the commencement of the take-off run and includes initial climb up to a screen height of 35 ft (this can be 50 ft for aircraft not certified to Perf A).

b. **Take-off Net Flight Path.** The take-off net flight path extends from a height of 35 ft (or 50 ft) to a height of 1,500 ft above the aerodrome.

c. **En Route.** The en route stage extends from a height of 1,500 ft above the departure aerodrome to a height of 1,500 ft above the destination or alternate aerodrome or, in extreme adverse circumstances, any suitable aerodrome.

d. **Landing**. The landing stage extends from a height of 1,500 ft at the destination, or alternate aerodrome, to the point where the aeroplane comes to a stop after completing the landing run.

23. Aeroplane operators deduce the most restrictive take-off weight from the net performance data after consideration of:

a. Structural limitation (C of A limit) for take-off and landing.

b. Weight, altitude and temperature (WAT) limit for take-off and landing.

c. Departure aerodrome criteria (runway lengths, slopes etc).

d. Take-off net flight path.

e. En route.

f. Landing aerodrome criteria at destination or alternate aerodrome.

Table 1

Performance Categories of Multi-engined Medium and Large Military Aircraft Operated by UK Forces.

Aircraft Type	Regulator	Regulations	Performance Category	Notes
Atlas A-400M	EASA	EU Ops	A	
Air Seeker RJ	FAA	Boeing Mil Spec	MIL-M-007700	Military performance spec similar to Perf A
Avenger T1	MAA	MARs	A	Aircraft originally certified to FAA 14 CFR part 23
Avro RJ100	CAA	BCARs	A	
BAe146-100 CCMk2	CAA	BCARs	A	
BAe146-200 QC Mk3	CAA	BCARs	A	
C17	FAA	Boeing Mil Spec	MDC S001/2	Military performance spec similar to Perf A
Defender BN2T	CAA	BCARs	B	
Hercules C130-J (Mk 4/5)	MAA	MARs	A	MAA require all aircraft to be certified to relevant DefStan. ROS / MOS outside of Perf A
HS 125-700 CC Mk3	CAA	BCARs	A	
Islander AL1	CAA	BCARs	C	
Islander CC2	CAA	BCARs	C	
KC-30 Voyager	EASA	EU Ops	A	
King Air B200	CAA	BCARs	C	
King Air B200-GT	EASA	EU Ops	B	
Sentinel R1	FAA	FARs	A	FAR 25
Sentry AEW Mk1	FAA	Boeing Mil Spec	MIL-M-7700B	Military performance spec similar to Perf A
Shadow R1	MAA	MARs	A	Aircraft originally certified to FAA 14 CFR part 23
Tristar C Mk2	CAA	BCARS	A	Subsidiary performance derived from FAA and CAA data
Tristar K Mk1 / KC Mk1	CAA	BCARS	A	Subsidiary performance derived from FAA and CAA data

CHAPTER 10 - DEFINITION OF TERMS

Introduction

1. Performance planning necessitates the introduction of certain terms, which enable the aeroplane performance to be related to such factors as aerodrome dimensions, maximum landing and take-off weights etc. The definitions given below cover the terms used in this section on scheduled performance.

Field Lengths

2. **Take-off Run Available (TORA)**. TORA is the length of runway which is declared by the State to be available and suitable for the ground run of an aeroplane taking off. This, in most cases, corresponds to the physical length of the runway pavement (ICAO).

3. **Stopway**. Stopway is a defined rectangular area on the ground, at the end of a runway, in the direction of take-off, designated and prepared by the Competent Authority as a suitable area in which an aeroplane can be stopped in the case of an interrupted take-off (ICAO). The width is the same as the runway.

4. **Accelerate Stop Distance Available (ASDA)**. ASDA is the length of the take-off run available plus the length of stopway available (if stopway is provided) (ICAO). Older ODMs may use the term Emergency Distance Available (EMDA).

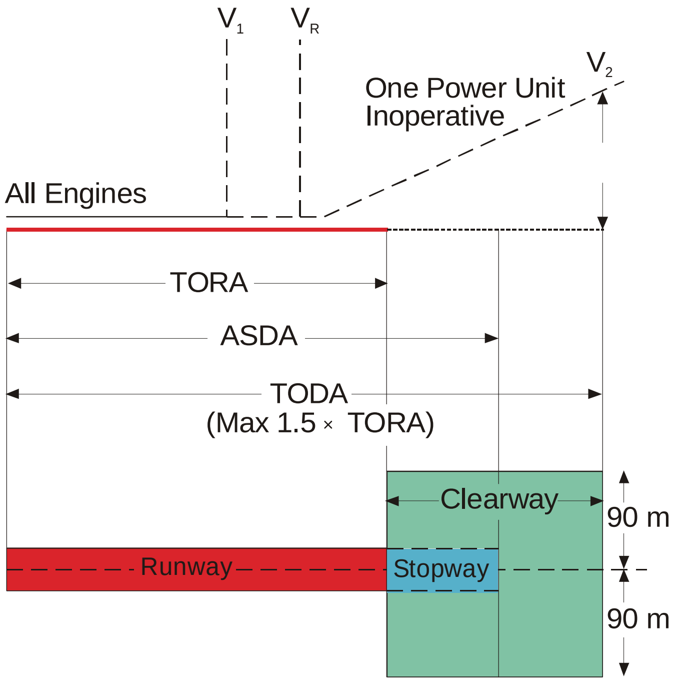
5. **Clearway**. Clearway is a defined rectangular area, 300 ft or 90 metres either side of the centreline, on the ground or water at the end of the runway, in the direction of take-off and under control of the Competent Authority, selected or prepared as a suitable area over which an aeroplane may make a portion of its initial climb to a specified height (ICAO).

6. **Take-off Distance Available (TODA).** TODA is the length of the take-off run available plus the length of the clearway available (if clearway is provided) (ICAO). TODA is not to exceed 1.5 × TORA.

7. **Balanced Field Length.** When the ASDA is equal to the TODA, this is known as a 'balanced field'. Some early ODMs have simplified graphs to solve the take-off problem when a balanced field exists. If these graphs are used to solve an unbalanced field (by reducing TODA to equal ASDA) then a weight penalty will be incurred.

8. **Unbalanced Field Length**. When the ASDA and TODA are different lengths, this is known as an 'unbalanced field'. By taking into account the varying amounts of stopway and clearway, the highest possible take-off weight for field length considerations will be obtained.

2-10 Fig 1 Take-off Performance Terminology



Speeds

9. **Indicated Air Speed (IAS).** IAS is the reading on the pitot-static airspeed indicator installed in the aeroplane, corrected only for instrument error.

10. **Calibrated Air Speed (CAS)**. CAS is IAS corrected for pressure error. The pressure error correction (PEC) can be obtained from the ODM for the type.

11. **Equivalent Air Speed (EAS)**. EAS is CAS corrected for compressibility.

12. **True Air Speed (TAS)**. TAS is the true airspeed of the aircraft relative to the undisturbed air. It is obtained by correcting the EAS for density.

13. **Decision Speed (V1)**. V_1 is a speed above which, in the event of a power unit failure, take-off must be continued, and below which take-off must be abandoned. V_1 depends on weight and aerodrome dimensions and is the most important product of performance planning. Its calculation before every flight ensures that, in the event of a power unit failure, the pilot's decision to abandon or continue take-off is completely objective. (Note: V_1 must not be less than V_{MCG} (see para 21), not greater than V_R and not greater than V_{MBE} (see para 26)).

14. **Rotation Speed (VR)**. V_R is a speed used in the determination of take-off performance at which the pilot initiates a change in the attitude of the aeroplane with the intention of leaving the ground. It is normally a function of aeroplane weight and flap setting, but it can vary with pressure altitude and temperature in some aeroplanes.

15. **Unstick Speed (VUS).** V_{US} is the speed that the main wheels leave the ground if the aircraft is rotated about its lateral axis at V_R. This speed is a function of flap setting and aircraft weight and is sometimes known in civilian aviation circles as Lift-off Speed (V_{LOF}).

16. **Take-off Safety Speed (V2)**. V_2 is a speed used in the determination of take-off performance. It is a legal requirement that at least this speed should be attained on one power unit inoperative performance by the time the aeroplane has reached a height of 35 ft. It is a function of weight, temperature and flap setting.

17. **All Engines Screen Speed (V3)**. V_3 is the speed at which the aeroplane is assumed to pass through the screen height with all engines operating on take-off.

18. **Steady Initial Climb Speed (V4)**. V_4 is a speed, with all engines operating, used in the scheduled take-off climb technique. It should be attained by a height of 400 ft.

19. **Target Threshold Speed (VAT).** V_{AT} is the speed at which the pilot should aim to cross the runway threshold for landing in relatively favourable conditions. The speeds at the threshold are:

a. V_{AT0} - all engines operating.

b. V_{AT1} - a critical engine inoperative.

20. **Maximum Threshold Speed (VTMAX).** V_{TMAX} is the speed at the threshold above which the risk of exceeding the scheduled landing field length is unacceptably high; normally assumed to be V_{AT_0} + 15 knots.

21. **Power Failure Speed Ratio (V1 /VR).** The power failure speed ratio for a given aeroplane weight and aerodrome dimensions is defined as the ratio V_1/V_R. The ratio is introduced into performance planning for convenience, as graphical presentation of aeroplane performance data is simplified when presented in terms of V_1/V_R rather than in terms of V_1 alone. V_R depends on the aeroplane weight and flap setting (some also consider pressure altitude and temperature). It is a simple matter to calculate V_1 once the aeroplane weight and V_1/V_R ratio has been determined.

22. **Ground Minimum Control Speed (VMCG)**. V_{MCG} is the minimum speed on the ground at which it is possible to suffer a critical power unit failure on take-off and maintain control of the aeroplane within defined limits.

23. **Air Minimum Control Speed (VMCA)**. V_{MCA} is the minimum control speed in the air in a take-off configuration at which it is possible to suffer a critical power unit failure and maintain control of the aeroplane within defined limits.

24. **Minimum Measured Unstick Speed (VMU)**. V_{MU} is the minimum demonstrated unstick speed; the minimum speed at which it is possible to leave the ground, all power units operating and climb without undue hazard.

25. **Minimum Speed in the Stall (VMS)**. V_{MS} is the minimum speed achieved in the stall manoeuvre:

a. V_{MS1} - with the aeroplane in the configuration appropriate to the case under consideration (EAS).

b. V_{MS0} - with the wing flaps in the landing setting (EAS).

26. **The Minimum Control Speed, Approach and Landing (VMCL)**. V_{MCL} is the minimum control speed in the air in an approach or landing configuration; the minimum speed at which it is possible, with one power unit inoperative to maintain control of the aeroplane within defined limits while applying maximum variations of power.

27. **Maximum Brake Energy Speed (VMBE).** V_{MBE} is the maximum speed on the ground from which a stop can be accomplished within the energy capabilities of the brakes.

Temperature

28. **International Standard Atmosphere (ISA)**. ISA is an atmosphere defined as follows:

The temperature at sea level is 15 °C; the pressure at sea level is 1013.2 mb (29.92 in Hg); the temperature gradient from sea level to an altitude (36,090 ft) at which the temperature becomes –56.5 °C is 1.98 °C per 1,000 ft.

29. **Outside Air or Static Temperature (OAT or SAT)**. OAT (or SAT) is the free air static (ambient) temperature.

30. **Indicated Air or Total Air Temperature (IAT or TAT).** IAT (or TAT) is the static air temperature plus adiabatic compression rise as indicated on the Total Air Temperature Indicator.

31. **Declared Temperature**. The appropriate average monthly temperature plus half the associated standard deviation is called the 'declared temperature'.

Altitude and Height

32. **Pressure Altitude.** Pressure Altitude is the expression of atmospheric pressure in terms of altitude, according to the inter-relation of these factors in the ISA. It is obtained by setting the sub-scale of an accurate pressure type altimeter to 1013.2mb.

33. **Height**. Height is the true vertical clearance distance between the lowest part of the aeroplane in an unbanked attitude with landing gear extended and the relevant datum.

34. **Gross Height**. Gross Height is the true height attained at any point on the take-off flight path using gross climb performance.

35. **Net Height.** The net height is the true height attained at any point on the take-off flight path using net climb performance.

36. **Screen Height.** Screen Height is the height of an imaginary screen which the aeroplane would just clear when taking off or landing, in an unbanked attitude and with the landing gear extended.

Performance

37. **Measured Performance.** Measured Performance is the average performance of an aeroplane or group of aeroplanes being tested by an acceptable method in the specified conditions.

38. **Gross Performance.** Gross performance is the average performance which a fleet of aircraft should achieve if satisfactorily maintained and flown in accordance with the techniques described in the Aircrew Manual.

39. **Net Performance**. Net performance represents the gross performance diminished by the amount considered necessary to allow for various contingencies, eg variations in piloting technique, temporary below average aircraft performance etc. It is expected that the net performance will be achieved in operation, provided the aircraft is flown in accordance with the recommended techniques.

40. **Gradient and Slope.** For deriving and applying Flight Manual information, gradient is the tangent of the angle of climb expressed as a percentage. The term 'slope' is used in place of 'gradient' when referring to aerodrome surfaces and obstacle profile. The ratio, in the same units and expressed as a percentage of:

$$\frac{\text{Change in Height}}{\text{Horizontal Distance Travelled}}$$

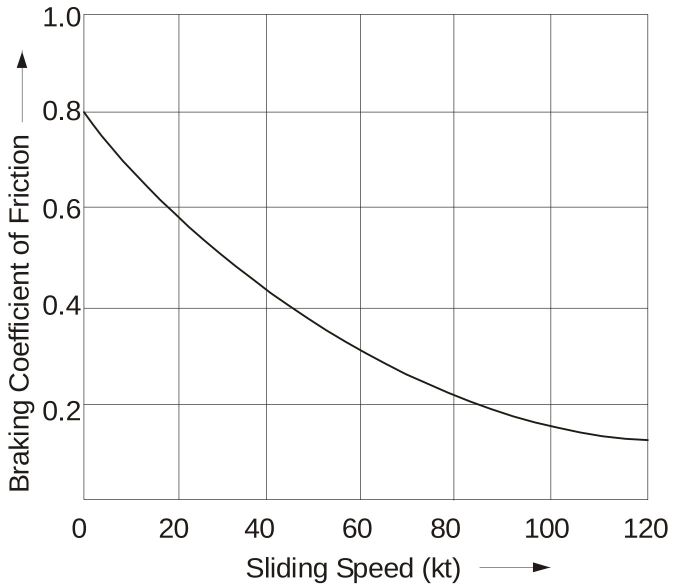
Aerodrome Surface Characteristics

41. **Braking Coefficient of Friction.** Braking Coefficient of Friction is the tangential force applied by an aerodrome surface (expressed as a proportion of the normal force) upon an appropriately loaded

smooth tyred aeroplane main wheel when it is being propelled across the surface, in a direction parallel to the plane of the wheel, when the speed of slip over the area of contact with the aerodrome surface is close to the speed of the aeroplane across that surface.

42. **Reference Wet Hard Surface.** A Reference Wet Hard Surface is a hard surface for which the relationship between aeroplane ground speed and the braking coefficient of friction is as given in Fig 2.

2-10 Fig 2 Reference Wet Hard Surface



Weights

43. **Aircraft Prepared for Service Weight (APS Weight)**. APS Weight describes the weight of a fully equipped operational aeroplane - but empty i.e. without crew, fuel or payload.

44. **Maximum Zero Fuel Weight (Max ZFW)**. Max ZFW is the weight of the aeroplane above which all the weight must consist of fuel. This limitation is always determined by the structural loading airworthiness requirements; unlike the other weight limitations, it is not associated with any handling or performance qualities.

45. **Take-off Weight (TOW)**. TOW is the gross weight including everything and everyone carried in or on it at the commencement of the take-off run.

46. **Landing Weight.** Landing Weight is the weight of the aeroplane at the estimated time of landing allowing for the weight of the fuel and oil expected to be used on the flight to the aerodrome at which it is intended to land or alternate aerodrome, as the case may be.

47. **Weight, Altitude and Temperature Limit (WAT)**. The highest weight at which the relevant airworthiness climb minima are met is termed WAT. The expressions 'altitude' and 'temperature' refer to the assumed pressure altitude and assumed atmosphere temperature for the take-off or landing surface.

Miscellaneous

48. **Critical Power Unit.** The Critical Power Unit is the power unit, failure of which gives the most adverse effect on the aeroplane characteristics relative to the case immediately under consideration.

49. **Power Unit Failure Point**. For determination of take-off performance, the power-unit failure point is that point at which a sudden complete failure of a power unit is assumed to occur.

50. **Decision Point**. For the determination of take-off performance, the decision point is the latest point at which, as a result of power unit failure, the pilot is assumed to decide to discontinue a take-off.

51. **Power Unit Restarting Altitude***.* Power Unit Restarting Altitude describes an altitude up to which it has been demonstrated to be possible safely and reliably to restart a power unit in flight.

52. **Engine Pressure Ratio (EPR)***.* EPR is a ratio of pressures, usually the maximum cycle pressure (compressor delivery pressure) to air intake pressure or ambient pressure (depending on specific applications) which is an important turbine engine parameter and can be displayed to the pilot.

53. **Bleed Air**. Air which has been compressed in the main engine compressor and utilized for cabin pressurization and the driving of various services is termed 'Bleed Air'.

Notes:

1. By distinguishing between power unit failure point and decision point, account is taken of the delay which occurs before a power unit failure can be detected.

2. An airspeed indicator reading is commonly used as a decision point criterion.

CHAPTER 11 - TAKE-OFF PERFORMANCE

Introduction

1. In the planning for take-off it is assumed that, although the aeroplane has all power units operating at the start point, one power unit will fail after commencement of the take-off run and before take-off is complete. Analysis of the take-off plan is therefore conducted on 'all power units operating' net performance up to the time of supposed power unit failure and thereafter on 'one power unit inoperative' net performance. However, it is possible in certain instances that a limiting factor in calculation of maximum take-off weight within the regulations can be imposed when all power units continue to operate throughout take-off. In the event of an actual power unit failure during take-off, the captain must decide whether to abandon or continue take-off. Performance planning ensures, amongst other considerations, that this decision is always completely objective.

Take-off Planning Considerations

2. Maximum permissible take-off weight for any flight will be the least weight obtained after considering:

   a. C of A limit.

   b. WAT limit for take-off.

   c. Field length requirements.

   d. Take-off net flight path.

   e. En route terrain clearance.

   f. WAT limit for landing.

   g. Landing distance requirements.

3. **C of A Limit.** The C of A specifies the manner in which the aeroplane may be flown and purpose for which it may be used. It also specifies maximum structural take-off weight for the type of aeroplane. This weight, which is normally listed in the aeroplane ODM under 'Limitations', is absolute and must not be exceeded.

4. **WAT Limit.** Air Navigation (General) Regulation 7(1) requires "that weight does not exceed the maximum take-off weight specified for the altitude and the air temperature at the aerodrome at which the take-off is to be made". This is the weight, altitude and temperature (WAT) limit, which is designed to ensure compliance with positive gradients of climb from take-off to a height of 1,500 ft above the take-off surface, as specified in British Civil Airworthiness Requirements (BCARs), with one power unit inoperative and all power units operating. The WAT limit in the ODM is normally presented as a single graph which is based on the most limiting of the climb requirements for the aeroplane type and is, in most cases, the one engine inoperative second segment climb requirement. This limit is readily obtained entering with arguments of airfield altitude and temperature, then extracting maximum weight for these conditions. Individual WAT limit graphs are published to account for different variables (such as varying flap positions, and cases where engine power output can be augmented). The WAT limit graphs take no account of field length.

5. **Field Length Requirements***.* Air Navigation (General) Regulation 7(2) requires that: "the take-off run, take-off distance and the accelerate stop distance respectively required for take-off, specified as being appropriate to:

   a. The weight of the aeroplane at the commencement of take-off run.

b. The altitude at the aerodrome.

c. The air temperature at the aerodrome.

d. The condition of the surface of the runway from which the take-off will be made.

e. The slope of the surface of the aerodrome in the direction of take-off over the TORA, the TODA and the ASDA, respectively, and

f. Not more than 50% of the reported wind component opposite to the direction of take-off or not less than 150% of the reported wind component in the direction of take-off.

do not exceed the take-off run, the take-off distance and the accelerate stop distance, respectively, at the aerodrome at which the take-off is to be made".

6. **Decision Speeds**. In addition, the regulations require that the decision point (normally decision speed, V_1) at which the pilot is assumed to decide to continue or discontinue the take-off in the event of a power unit failure, must be common to all 3 distance requirements and satisfy each individually. Some ODMs contain additional decision speed information for use in wet runway conditions. The use of the 'wet' decision speed will enhance the ability to stop on a wet surface and is, on average, 5 kt to10 kt less than the 'dry' decision speed usable with the same regulated take-off weight using dry accelerate stop distance. However, in such conditions there will be a short risk period of up to 4 seconds, following V_1 WET, during which, should a power unit failure occur on take-off, the aeroplane will not achieve the screen height of 35 ft at the end of TODR. The minimum height which will be reached will be 15 ft, but in most cases will be more.

7. **Procedures**. In determining field length requirement each critical aspect of TORA, ASDA and TODA must be considered, regardless of whatever method of presentation (D and X or D and R graphs etc) is used.

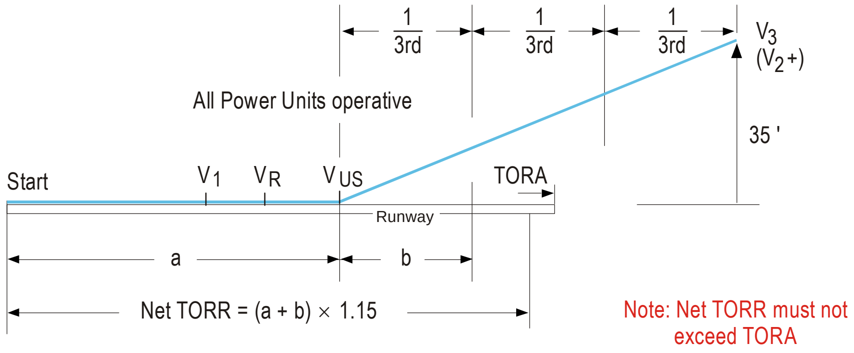
Take-off Field Length Requirements

8. The regulations specify distinct minimum requirements for take-off field length, some of which may be more limiting than others. The take-off weight and V_1 speed must be such that the most severe of these requirements is covered. The requirements under BCARs are described within this section. The JAR regulations are similar in concept, but vary slightly in detail. The aircraft ODM will take account of the appropriate requirements.

9. **Take-off Run Required (TORR)**. The TORR shall be the greatest of:

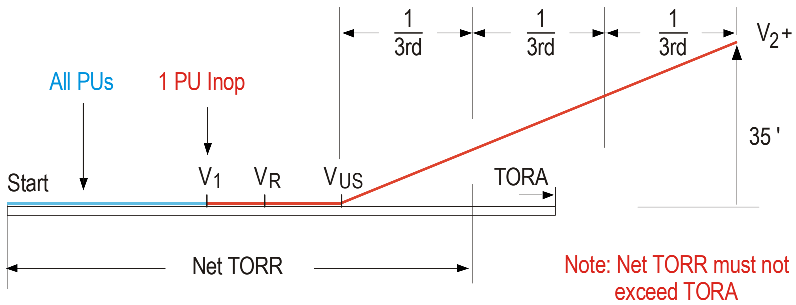
a. **All Power Units Operating.** This is 1.15 × the sum of the gross distance from the starting point to the point where the aeroplane becomes airborne (V_{US}) and 1/3 of the gross distance between V_{US} and the point at which it attains a screen height of 35 ft (see Fig 1).

2-11 Fig 1 TORR - All Power Units Operating



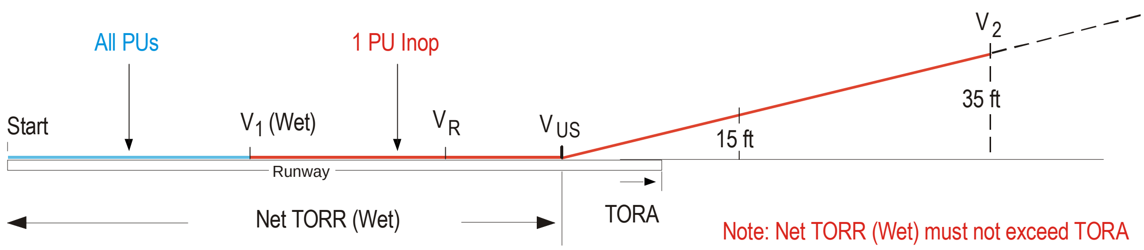
b. **One Power Unit Inoperative**. This is the gross distance from the starting point to the point where the aeroplane becomes airborne (V_{US}) plus 1/3 the gross distance between V_{US} and the point at which it attains a screen height of 35 ft. The power unit fail point shall be such that failure would be recognized at the decision point (V_1) appropriate to a dry runway (see Fig 2).

2-11 Fig 2 TORR - One Power Unit Inoperative



c. **One Power Unit Inoperative - Wet Runway**. This is the gross distance from the starting point to the point where the aeroplane becomes airborne (V_{US}) to effect a transition to climbing flight to attain a screen height of 15 ft at the end of TODA, in a manner consistent with the achievement of a speed not less than V_2 at 35 ft. The power unit failure point shall be such that failure would be recognized at the decision point (V_1) appropriate to a wet runway (see Fig 3).

2-11 Fig 3 TORR - One Power Unit Inoperative (Wet Runway)

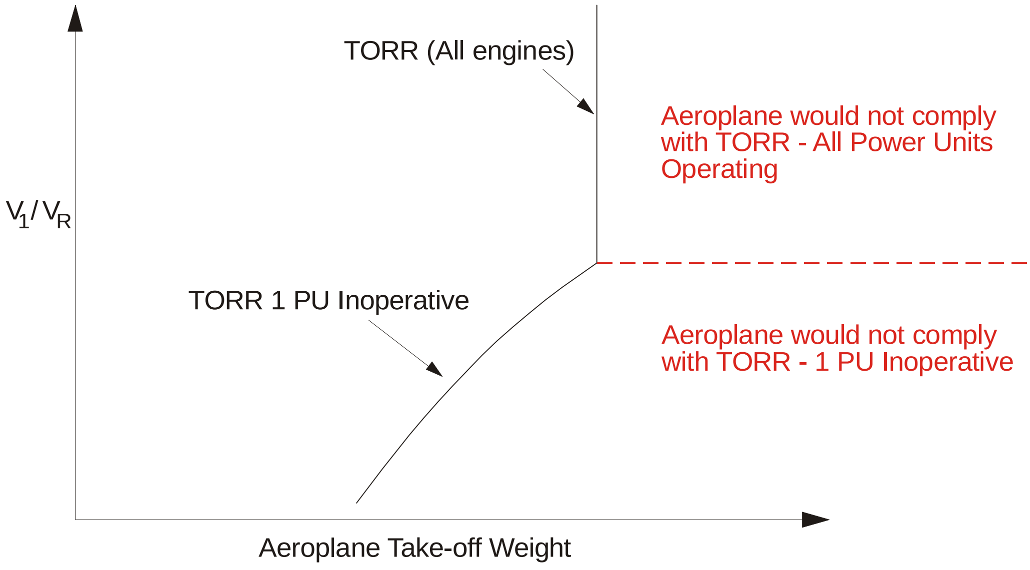


10. **TORR Compliance**. In order to comply with Air Navigation (General) Regulation 7(2) therefore, TORR should be the greatest of sub-paras 9a, b or c and must not exceed TORA.

11. **TORR Analysis**. In take-off run analysis there are 2 variables: V_1 and aircraft weight. If the aircraft is light, then it is possible to lose a power unit earlier during the take-off run than if it is heavy and still comply with the regulations. Thus, ignoring all other considerations, V_1 depends on aeroplane weight and increases as weight increases. There is a further consideration which must be taken into account. If all

power units continue to operate throughout the whole take-off, then there is an upper weight limit imposed as by sub-para 9a. A typical graph is shown in Fig 4, the vertical portion of the graph is due to the 'all power units operating' limitation.

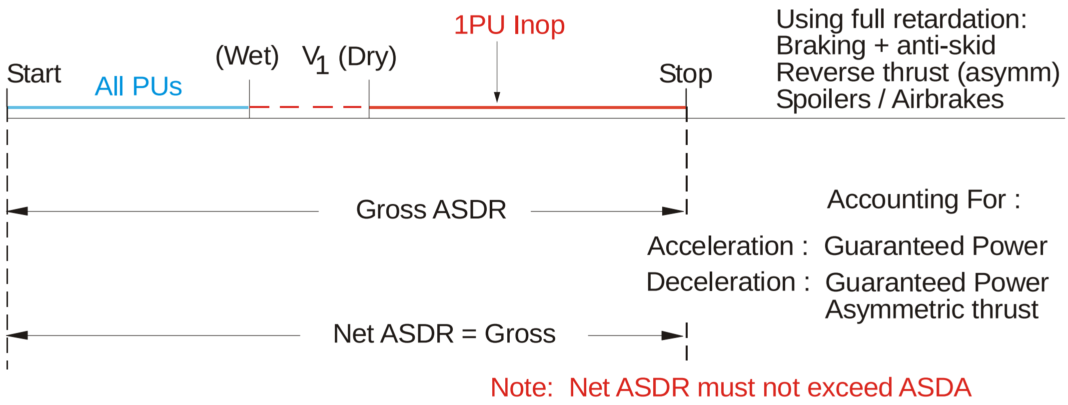
2-11 Fig 4 Take-off Run Analysis



12. **Accelerate Stop Distance Required (ASDR)**. The ASDR shall be the greatest of:

a. The sum of the gross distance to accelerate with all power units operating from the starting point to the decision point (V_1) appropriate to a dry runway and the gross distance to stop with the critical power unit inoperative from the decision point on a dry hard surface using all available means of retardation (see Fig 5).

2-11 Fig 5 ASDR

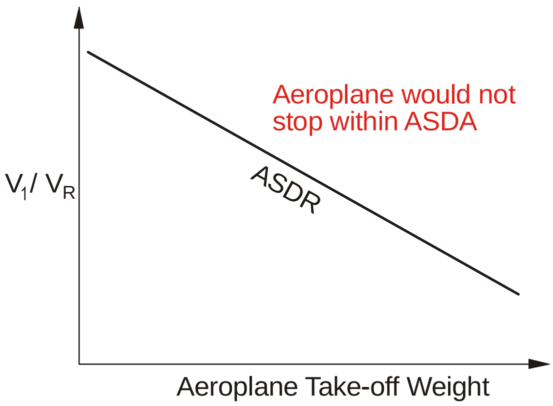


b. The sum of the gross distance to accelerate with all power units operating from the starting point to the decision point (V_1) appropriate to a wet runway and the gross distance to stop with the critical power unit inoperative from the decision point on the reference wet hard surface using all available means of retardation (see Fig 5).

13. ***ASDR Compliance.*** In order to comply with Air Navigation (General) Regulation 7(2) therefore, the ASDR should be the greatest of sub-paras 12a or b and must not exceed ASDA.

14. **ASDR Analysis**. In accelerate stop distance analysis there are 2 variables, V_1 and weight, but this time, if the aeroplane is light, then a power unit can be lost later during the take-off run than if the aeroplane is heavy. Thus, with accelerate stop distance (and ignoring all other considerations), V_1 depends on aeroplane weight and decreases as aeroplane weight increases. A typical graph is shown in Fig 6.

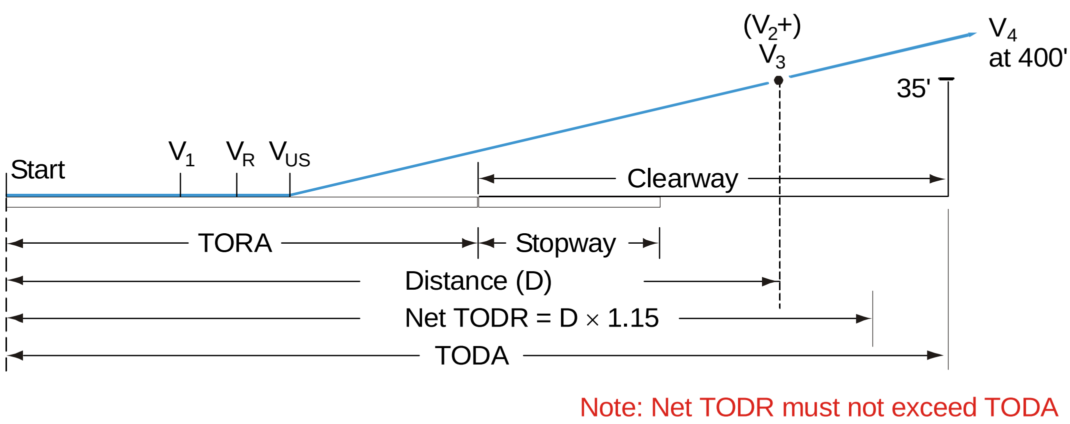
2-11 Fig 6 Accelerate Stop Distance Analysis



15. **Take-off Distance Required (TODR)**. The TODR shall be the greatest of:

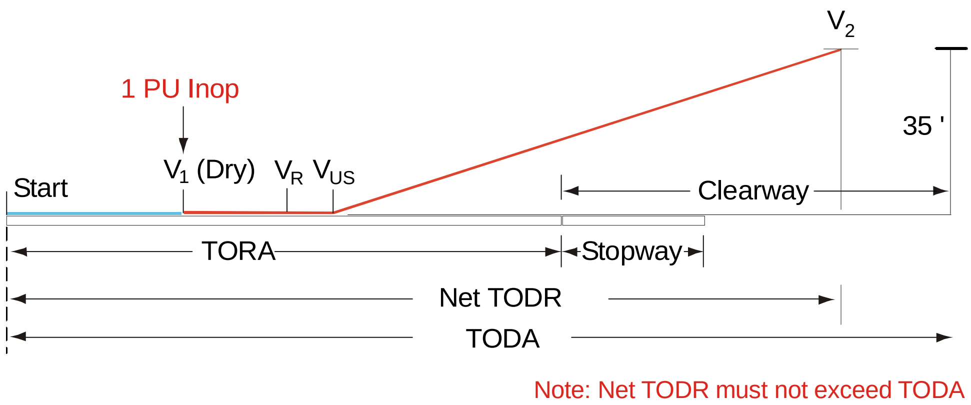
a. **All Power Units Operating.** This is 1.15 times the gross distance to accelerate with all power units operating from the starting point to the Rotation Speed (V_R), to effect a transition to climbing flight and attain a screen height of 35 ft; the speed at 35 ft shall not be less than take-off safety speed (V_2) and shall be consistent with the achievement of a smooth transition to the steady initial climb speed (V_4) at a height of 400 ft (see Fig 7).

2-11 Fig 7 TODR - All Power Units Operating



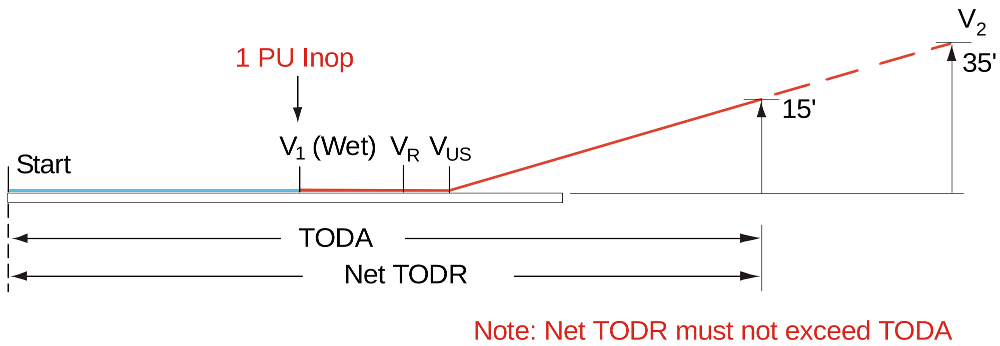
b. **One Power Unit Inoperative**. This is the gross distance to accelerate with all power units operating from the starting point to the decision point (V_1), appropriate to a dry runway, then to accelerate with the critical power unit inoperative to the rotation speed (V_R) and thereafter to effect a transition to climbing flight and attain a screen height of 35 ft at a speed not less than take-off safety speed (V_2) (see Fig 8).

2-11 Fig 8 TODR - One Power Unit Inoperative



c. **One Power Unit Inoperative - Wet Runway**. This is the gross distance to accelerate with all power units operating from the starting point to the decision point (V_1), appropriate to the rotation speed (V_R) and thereafter to effect a transition to climbing flight and attain a screen height of 15 ft in a manner consistent with the achievement of a speed not less than take-off safety speed (V_2) at 35 ft, the failure of the power unit being recognized at V_1 appropriate to a wet runway (see Fig 9).

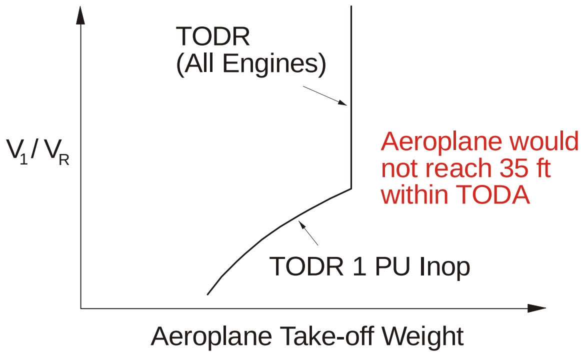
2-11 Fig 9 TODR - One Power Unit Inoperative (Wet Runway)



16. **TODR Compliance*.*** In order to comply with Air Navigation (General) Regulation 7(2) therefore, the TODR should be the greatest of sub-paras 15a, b or c and must not exceed the TODA.

17. **TODR Analysis*.*** In take-off distance analysis there are the same 2 variables as before, ie V_1 and aeroplane weight. If the aeroplane is light, a power unit can be lost earlier than if the aeroplane is heavy. Thus with take-off distance and ignoring all other considerations, V_1 depends on weight and increases as weight increases. As in the case of take-off run, a further consideration must be taken into account. If all power units continue to operate throughout the whole take-off, this simply places an upper limit on the aeroplane weight just as the similar requirement did in the case of take-off run. A typical graph of the limiting value of weight is shown in Fig 10.

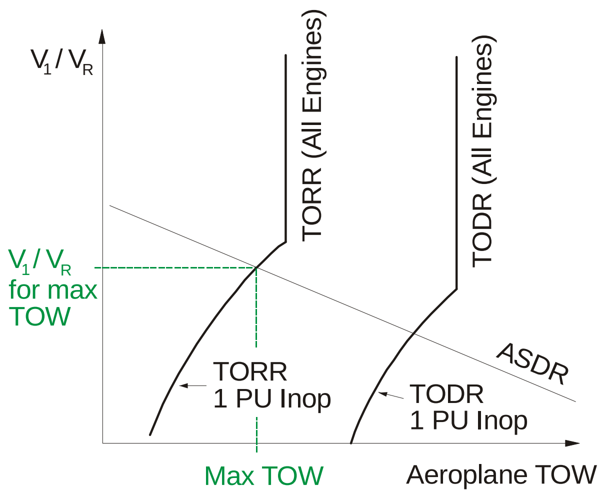
2-11 Fig 10 Take-off Distance Analysis



Final Take-off Analysis

18. If the analysis of the 3 field length requirements (Figs 4, 6 and 10) are plotted on a common axis, the composite graph of Fig 11 results. This will provide maximum permissible take-off weight and its associated V_1 speed which completely satisfies the regulations with reference to the field length requirements.

2-11 Fig 11 Final Take-off Analysis



19. The disposition of the 3 graphs depends entirely on the airfield dimensions and, although in Fig 11 maximum permissible take-off weight and V_1/ V_R ratio were determined by the take-off run 'one power unit inoperative' case, they might equally well have been determined by either of the take-off distance limitations.

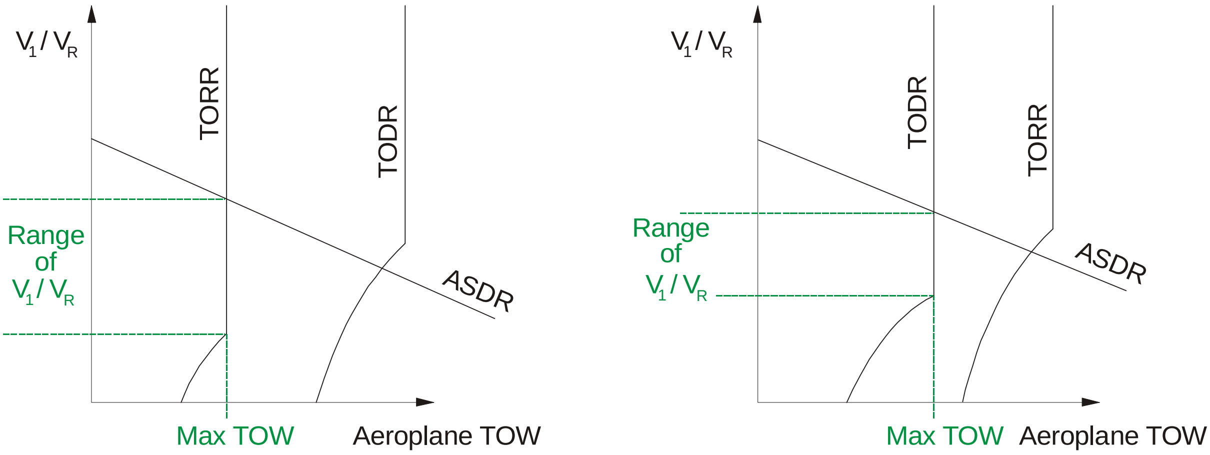
20. There will be a range of V_1/ V_R values (see Fig 12) in those cases where maximum permissible take-off weight is limited by:

a. WAT limitation.

b. Take-off run, all engines operating.

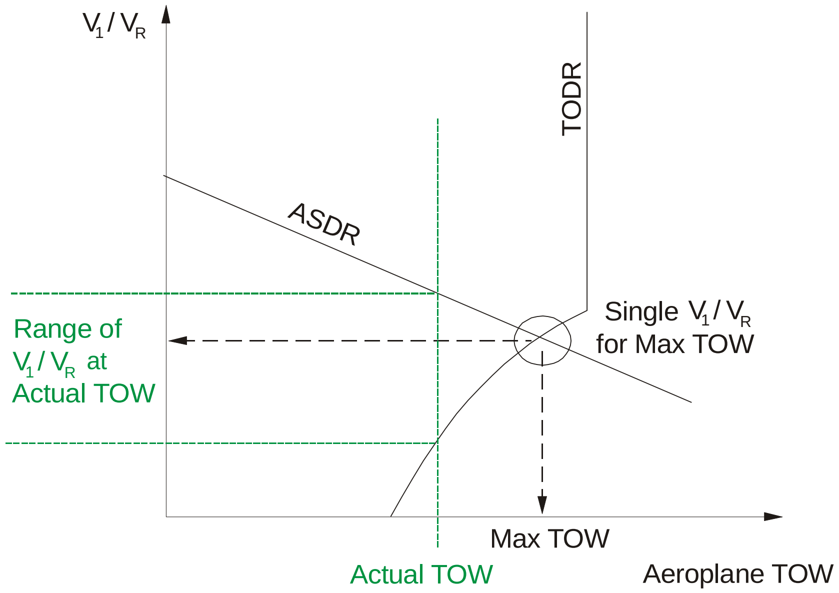
c. Take-off distance, all engines operating.

2-11 Fig 12 Range of V_1 Values - All Power Units Operating Limited



There will also be a range when the actual take-off weight is less than the maximum permissible (see Fig 13). The decision to use high or low V_1 depends upon the aeroplane type and operating techniques.

2-11 Fig 13 Range of V_1 Values - Weight Below Maximum Permissible



21. When field lengths are unbalanced it is necessary to consider individually take-off run, accelerate stop distance and take-off distance available. If, however, ASDA and TODA are of the same length, or assumed to be so, then the field is said to be 'balanced' and compliance with these 2 distance requirements can be achieved simultaneously for the case of power unit failure at V_1. From a balanced field chart it is possible, in one step, to determine:

a. Maximum take-off weight to comply with accelerate stop distance with one power unit inoperative.

b. The V_1/V_R ratio appropriate to this weight and common to the 2 distances. It is now only necessary to check the required take-off run for this weight and the V_1/V_R ratio with the take-off distance 'all power units operating' to ensure they are not limiting.

22. When TORA, ASDA and TODA are all equal, then take-off run can never be the limiting factor, but it still remains necessary to check take-off distance 'all power units operating'. If 'clearway' does exist and the balanced field length graphs are used, a penalty in take-off weight will be imposed. This may be justified, however, by the speed and simplicity of using this method.

23. Performance planning for the take-off stage is now completed by selection of the most limiting weight from the factors considered so far, ie:

a. C of A limit.

b. WAT limit (take-off).

c. Field length requirements.

d. Other factors to be considered that may affect the aircraft weight are:

(1) Brake heating limitations.

(2) Tyre speed and pressure limitations.

(3) Crosswind limitations.

(4) Use of air bleeds for anti-icing or air conditioning systems.

(5) Reduced performance due to slush, snow or standing water on the runway.

(6) Noise abatement regulations.

(7) Aerodrome pavement strength.

(8) Anti-skid system inoperative.

(9) Reverse thrust inoperative.

(10) Flap settings.

(11) Use of water injection or water methanol.

CHAPTER 12 - NET TAKE-OFF FLIGHT PATH

Introduction

1. The take-off stage of performance planning (Volume 2, Chapter 11) is complete at the end of the take-off distance required (TODR), at which point the aircraft is 35 ft above the datum surface level, in take-off configuration. The net take-off flight path segment extends from this point, to a safe height of 1,500 ft above the same datum. During this phase of flight, the aircraft must change to en route configuration, and all obstacles must be cleared by stipulated safety margins.

2. Air Navigation (General) Regulation 7(3) requires the aeroplane to clear all obstacles in its path by a vertical interval of at least 35 ft. The net take-off climb, like all other portions of scheduled performance, is planned on the net performance with one power unit inoperative.

3. Turns in the climb path are legal, but regulatory limitations and restrictions must be observed. The Operating Data Manual (ODM) will provide scheduling for such turns. The obstacle domain will curve accordingly. If it is intended that the aeroplane shall change its direction of flight by more than 15º, the vertical interval is increased to give at least 50 ft clearance from any obstacle encountered during the change of direction. Turns must only be planned within climbing segments, since the radius of turn cannot be calculated for those segments where the aeroplane is accelerating.

4. Unless obstacle height requires further extension of the take-off flight path, the net take-off flight path is completed when a height of 1,500 ft has been attained, and transition to the en route configuration completed.

5. In determining the net take-off flight path with one power unit inoperative, the following factors must be considered:

a. The weight of the aeroplane at the commencement of the take-off run. This weight is referred to as the take-off weight (TOW).

b. The altitude of the aerodrome.

c. The air temperature at the aerodrome.

d. The average slope of the take-off distance available (when appropriate).

e. Not more than 50% of the reported wind component opposite to the direction of take-off or not less than 150% of the reported wind component in the direction of take-off.

f. The height of an obstacle (when appropriate).

g. The distance to an obstacle (when appropriate).

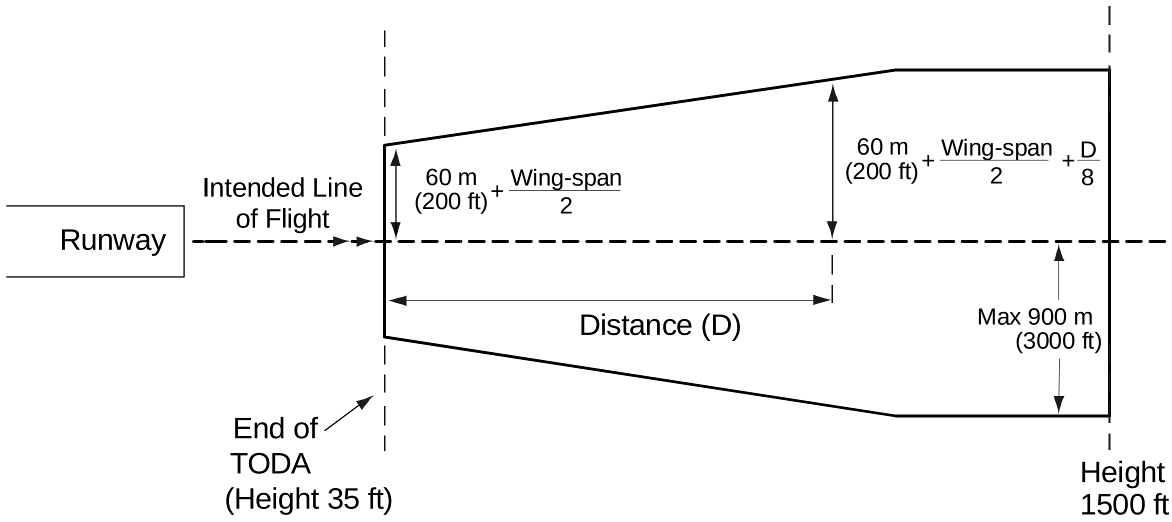
Legal Definition of an Obstacle

6. To comply with para 2, an obstacle shall be deemed to be in the path of the aeroplane if the distance from the obstacle to the nearest point on the ground below the intended line of flight of the aeroplane does not exceed the lesser of the following:

a. A distance equal to 60 m, plus half the wing-span of the aeroplane, plus one eighth of the distance from that point to the end of the TODA, measured along the intended line of flight (see Fig 1).

b. 900 m.

2-12 Fig 1 The Obstacle Domain (Performance Group A)

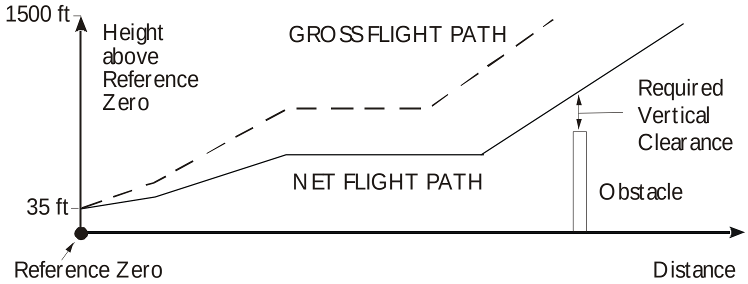


7. The domain within which an object can be considered a legal obstacle can therefore be plotted for a straight take-off as in Fig 1. If the aeroplane changes its direction of flight by more than 15º, the obstacle domain maintains the same dimensions but curves with the curved intended line of flight.

Use of Net Climb Performance

8. Having ascertained that an obstacle is within legal bounds, it is necessary to establish that the required clearance is obtained using net climb performance with one power unit inoperative. This is obtained by using the gross climb performance, reduced by a safety margin as specified in BCARs. The net flight path will therefore be below that achieved on gross power (Fig 2).

2-12 Fig 2 Comparison of Gross and Net Flight Paths

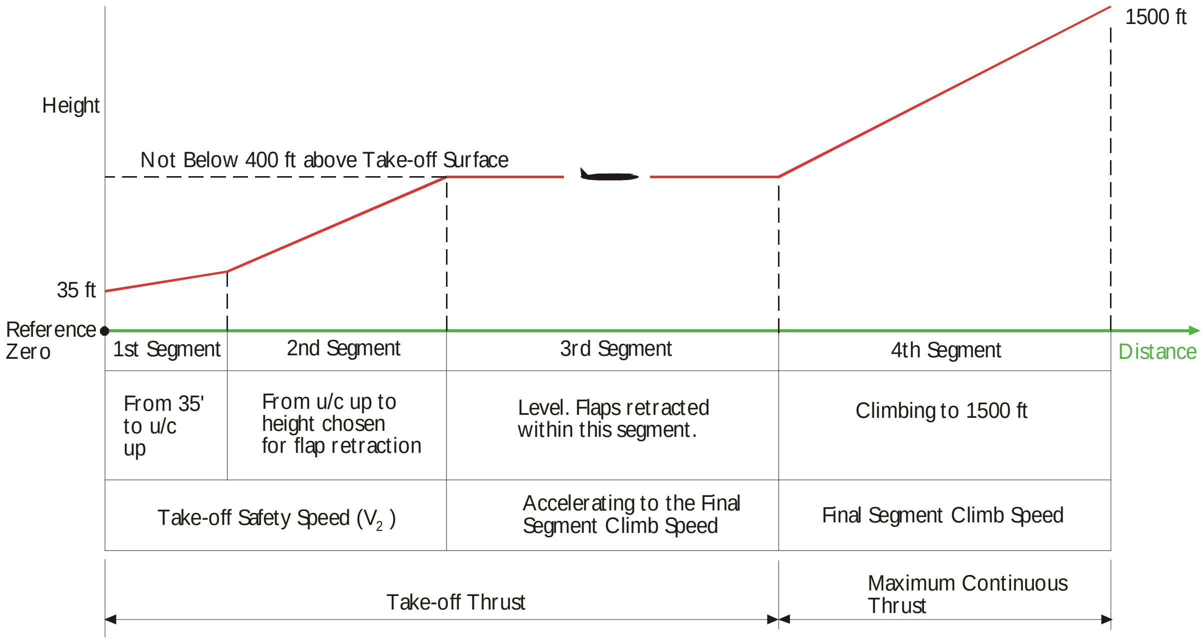


The Net Take-off Flight Path Profile

9. The net take-off flight path is divided into a series of segments (maximum 6 - some aircraft use a smaller number of segments).

10. **A Typical 4-segment Profile**. A typical 4-segment net flight path is illustrated in Fig 3. Some ODMs may use speed and power settings that differ from those stated.

2-12 Fig 3 Net Take-off Flight Path Segments (One Power Unit Inoperative)



a. **Reference Zero**. Reference Zero is the point 35 ft directly below the aeroplane at the end of TODR. It is the datum to which the co-ordinates of the various points in the take-off flight path are referred.

b. **First Segment**. This segment extends from the 35 ft point, to the point where the landing gear is fully retracted and (if applicable) the propeller of the failed power unit is feathered. This segment is flown at, or above, the Take-off Safety Speed (V_2).

c. **Second Segment**. This segment is flown at a minimum of V_2, and extends from the end of the first segment to the height chosen for the initiation of flap retraction. The height chosen for flap retraction depends on the aeroplane. The ODM will give one of the following as the preferred height:

(1) The attainment of a minimum height of 400 ft.

(2) The attainment of the maximum level-off height (MLOH). The MLOH results from any limitation if flap retraction is delayed in order to achieve clearance of obstacles close to Reference Zero.

(3) The attainment of the maximum height that can be reached within the time limit imposed by use of take-off thrust (typically 5 minutes after brake release).

(4) The attainment of a height of 1,500 ft.

d. **Third Segment**. This segment assumes a period of level flight at take-off thrust, during which the aircraft accelerates and the flaps are retracted in accordance with the recommended speed schedule.

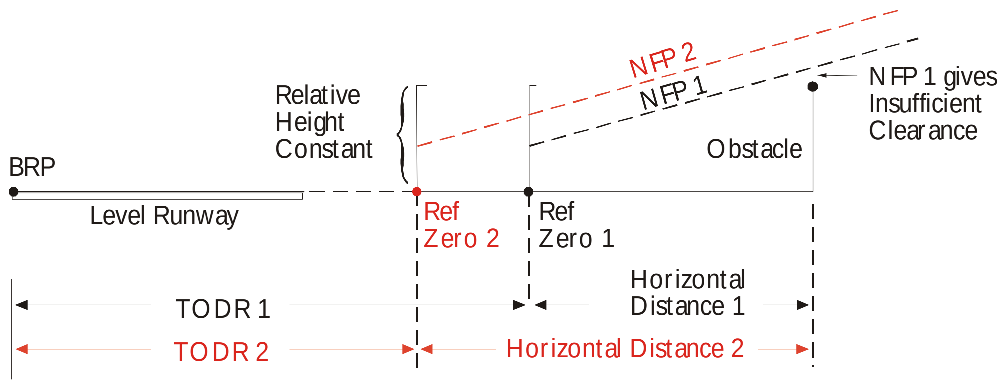
e. **Fourth (or Final) Segment**. This segment extends from the level-off height to a net height of 1,500 ft or more. This segment is flown at the final segment climb speed using maximum continuous thrust.

Obstacle Datums

11. It is essential to use a common datum in horizontal and vertical planes when comparing obstacle location and elevation with runway data. In the UK AIP, obstacles are listed by latitude and longitude, with height above mean sea level. In other sources, obstacle location may be given in alternate formats, for example, as a distance and height relative to the start of the take-off run (known as the Brake Release Point (BRP)). The operator must take extreme care, and if necessary reduce the position of reference zero and that of any obstacles to a common datum.

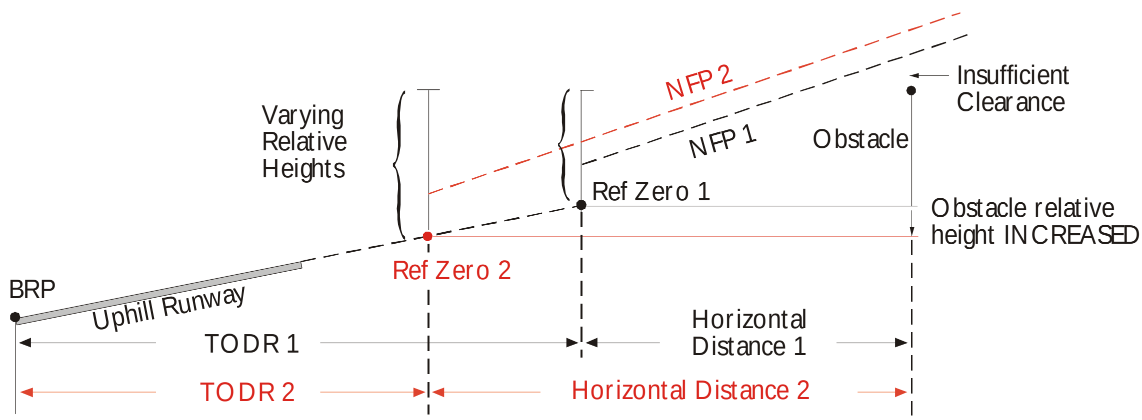
12. **Runways with Zero Slope**. If the average slope of the take-off distance is zero, then the relative heights of BRP, reference zero and any obstacles will remain constant. Only the horizontal distance between them alters with a change of take-off distance required (TODR). In Fig 4, the first calculation is made at maximum permitted TOW, resulting in the first take-off distance required (TODR 1) and its associated net flight path (NFP 1). As this gives insufficient vertical clearance from the plotted obstacle, a second calculation is made, based upon a reduced TOW (TODR 2). The reference zero is now at a greater horizontal distance from the obstacle, and NFP 2 will result in an increased vertical clearance. If the vertical clearance is still insufficient, then the process is repeated at a third, further reduced TOW, and so forth, until sufficient clearance is achieved.

2-12 Fig 4 Reference Zero - Take-off Distance with Zero Slope

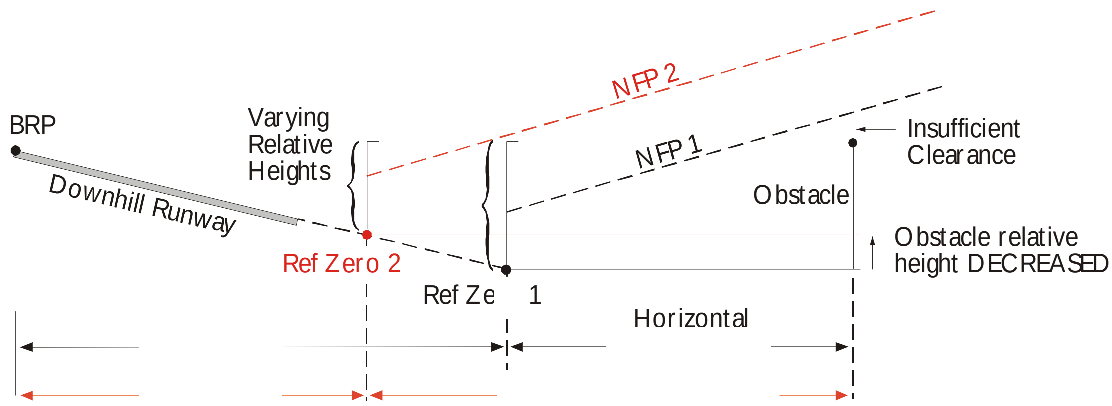


13. **Runways with a Slope**. When a slope exists in the take-off distance, either up or down, not only will the horizontal distance vary, but also the relative heights of the obstacles and reference zero will need careful consideration (see Figs 5 and 6)

2-12 Fig 5 Reference Zero - Take-off Distance with Uphill Slope



2-12 Fig 6 Reference Zero - Take-off Distance with Downhill Slope



Determination of Obstacle Clearance

14. The initial calculation for the net flight path, to determine positive vertical clearance (35 ft or 50 ft) from obstacles, is based on the maximum permitted TOW determined from the take-off performance calculation (see Volume 2, Chapter 11). If, on using this weight, the required vertical obstacle clearance is obtained, then the net take-off flight path imposes no limitations.

15. If the required obstacle clearance is not obtained on the initial calculation, then in general, the climb gradient will need to be increased, and this is achieved by a reduction in the TOW.

16. The net take-off flight path segment calculations can be determined by one of 2 methods:

a. **Scale Plot of Net Take-off Flight Path**. A scale plot of the aeroplane net take-off flight path can be constructed (see Volume 2, Chapter 13) showing aeroplane net height against horizontal distance travelled, measured along the intended line of flight, either straight or curved. Any obstacle within the obstacle domain is drawn on this scale plot at the appropriate height and horizontal distance, in order to assess the vertical clearance. If, on the scale plot, the legal clearance is not achieved, then the TOW must be reduced and a new scale plot constructed using a lower weight. This procedure must be repeated until the legal vertical clearance is obtained.

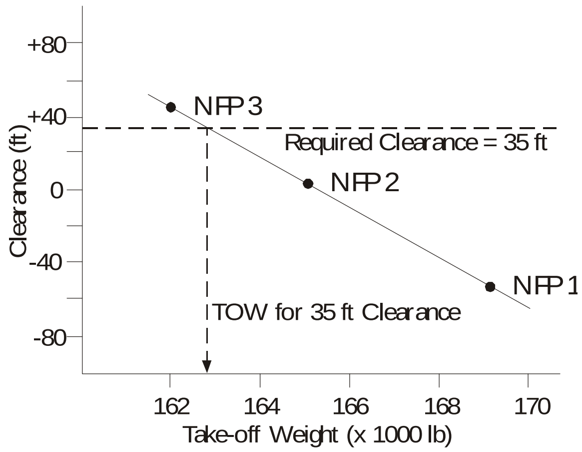
b. **Obstacle Clearance Charts**. In some ODMs, clearance is determined by using Obstacle Clearance Charts, which show net flight path profiles, reduced by 35 ft, as a function of take-off climb gross gradient at 400 ft. Knowing the distance and height difference between an obstacle and reference zero, these dimensions can be plotted on the appropriate obstacle clearance chart to find the required climb gradient. If the climb gradient for the planned TOW is insufficient to clear the obstacle, the TOW must be reduced to produce the required gradient. Alternatively, in some ODMs, use may be made of the 'increased V_2 speed' method. This method uses all or some of the excess field length to increase the take-off speed and thus improve the climb gradient achievable.

17. Allowances for fuel burn-off and performance change with height are accounted for in ODM net flight path charts. Therefore, the weight input for charts will be TOW.

18. **Scale Plot - Redefining Reference Zero**. Where further scale plots are constructed, as per para 16a, reference zero will be redefined at each TOW, at the end of TODR. Having repositioned reference zero, it will also be necessary to re-adjust obstacles horizontally and vertically as applicable.

19. **Scale Plot - Determining Precise TOW**. With small adjustments of TOW, the relationship between reduced TOW and increased obstacle clearance can be assumed to be linear. In order to determine the maximum TOW with precision, a graph showing vertical clearance can be plotted for 2 or 3 flight paths at different TOWs. The axes of the graph will show TOW against clearance. In the example in Fig 7, clearances have been determined for TOWs of 169,000 lb, 165,000 lb and 162,000 lb (NFP 1 to 3 respectively). Interpolation shows that the maximum permitted TOW to achieve 35 ft vertical clearance is 162,700 lb.

2-12 Fig 7 Graph for Determination of Maximum TOW



Increase in Height of Final Segment

20. If it is necessary, or desired, to demonstrate obstacle clearance after take-off to a height in excess of 1,500 ft, then the final climb segment may be scheduled to a greater height, or alternatively the normal en route one power unit inoperative net data can be used from the 1,500 ft point onwards.

CHAPTER 13 - CONSTRUCTION OF NET TAKE-OFF FLIGHT PATH SCALE PLOT

Introduction

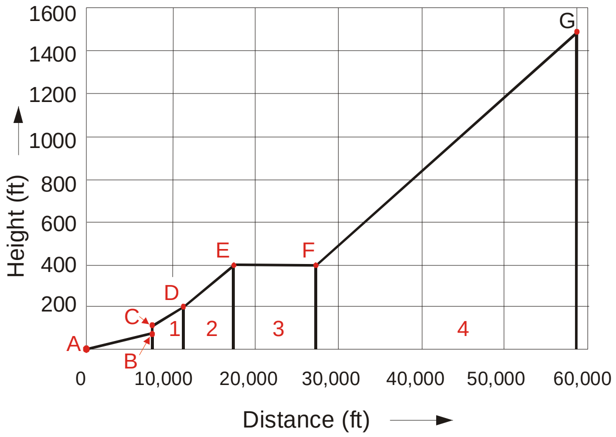
1. The more modern flight manuals contain obstacle clearance charts which do not require the construction of a net take-off flight path scale plot to calculate the maximum take-off weight to ensure legal obstacle clearance. However, there are many ODMs in current use which do not contain obstacle clearance charts.

2. There are various methods of constructing the scale plot of the take-off net flight path and the choice of method used is open to the operator, but the following is suggested as being the simplest. Before this or any other method is used, the basic principles described in Volume 2, Chapter 12 must be understood.

Construction Sequence

3. Position obstacles in relation to the 'brakes-off' point as regards both distance and height (runway obstruction data is normally quoted in relation to the 'brakes-off' point). Both net flight path and obstacles can now be plotted from the origin of the graph, i.e. 'brakes-off'. The method of plotting the net flight path is explained below with reference to Fig 1.

2-13 Fig 1 Construction of Net Take-off Flight Path



4. Plot the TODR (A-B) using the runway slope to calculate the height gained/lost by the end of TODR in relation to the 'brakes-off' point.

5. The aeroplane will have achieved a 35 ft screen height at the end of the TODR. Add 35 ft to the height at point B and plot point C.

6. From the graphs, determine segment 1 gradient (if applicable).

7. From the graphs, determine the height gained during segment 1.

8. From paras 6 and 7, calculate the horizontal distance travelled during segment 1, unless obtained directly from a graph.

9. Add the values of height gained and horizontal distance travelled in segment 1 to the values of height and distance at the end of the TODR segment. This gives the total height gained and distance travelled since 'brakes-off'.

10. Plot these values on the graph (point D) and join this point to the end of the TODR segment (point C). The TODR and segment 1 are now accurately depicted on the graph.

11. From the graphs determine the segment 2 gradient.

12. From the graphs determine height gained during segment 2.

13. From paras 11 and 12 calculate the horizontal distance travelled during segment 2, ie:

$$\text{Distance travelled} = \frac{\text{Change in Height} \times 100}{\text{Gradient}\,\%}$$

14. Add the values obtained in paras 12 and 13 to the previous total values of height and distance to obtain total height and distance from the 'brakes-off' point at the end of segment 2.

15. Plot these values on the graph (point E) and join this point to the end of segment 1 (point D). The graph is now complete to the end of segment 2.

16. From the graphs, determine the distance travelled in segment 3.

17. Add this value to the previous total to obtain the total distance travelled (the height remains unchanged).

18. Plot these values on the graph (point F) and join this point to the end of segment 2 (point E).

19. From the graphs, determine the segment 4 gradient.

20. From the graphs, determine the height of the end of the fourth segment which will be either 1,500 ft above the end of TODR (in which case the net flight path is complete) or a lesser height determined by the 5 minute power point.

21. From paras 19 and 20, calculate the distance travelled during segment 4.

22. Add this value to the previous total value to obtain the total distance travelled since 'brakes-off' and plot point G.

23. If segment 5 is required, gradient will be determined, height and distance increments calculated and summed as for previous segments to obtain total values.

24. If segment 6 is required to reach 1,500 ft, establish the segment 6 height increment and gradient and calculate the height increment as before.

CHAPTER 14 - EN ROUTE PERFORMANCE

Introduction

1. En route performance planning concerns the portion of the flight which starts at 1,500 ft, ie at the end of the net take-off flight path and ends at 1,500 ft above the landing surface.

En Route Obstacle Clearance

2. The Air Navigation (General) Regulation 7(4) requires that, in the event of power unit failure, the aeroplane will clear all obstacles within 10 nm either side of the intended track by a vertical interval of at least 2,000 ft with the remaining power unit(s) operating at maximum continuous power. The intended track may be the planned route or on any planned diversion to any suitable aerodrome at which the aeroplane can comply with the condition in the Regulation relating to an alternate aerodrome. On arrival over such an aerodrome, the gradient of the specified net flight path with one power unit inoperative shall not be less than zero at 1,500 ft above the aerodrome.

3. For aeroplanes with three or more power units, condition (5)(a) of the Air Navigation (General) Regulation 7 must be complied with whenever the aeroplane on its route or any planned diversion therefrom is more than 90 minutes flying time in still air at the 'all power units operating' economical cruising speed from the nearest aerodrome at which it can comply with the condition in the Regulation relating to an alternate aerodrome. This requires that the aeroplane, in the event of any two power units becoming inoperative, be capable of continuing flight with all other power unit(s) operating at maximum continuous power, clearing by a vertical interval of at least 2,000 ft, all obstacles within 10 nm either side of the intended track to a suitable aerodrome. On arrival over such an aerodrome, the gradient of the specified net flight path with two power units inoperative shall not be less than zero at 1,500 ft above the aerodrome. The speed to be used for calculating 90 minutes flying time is that scheduled in the Operating Data Manual (ODM) for the application of the Air Navigation Regulations relating to flight over water.

4. Provided that, where the operator of the aeroplane is satisfied, taking into account the navigation aids which can be made use of on the route, that tracking can be maintained within a margin of 5 nm, then 5 nm can be substituted for 10 nm in paras 2 and 3.

5. In assessing the ability of the aeroplane to comply with en route obstacle clearance requirements, account must be taken of meteorological conditions for the flight and the aeroplane's ice protection system must be assumed to be in use when appropriate. Account may be taken of any reduction of the weight of the aeroplane which may be achieved after the failure of a power unit or power units by such jettisoning of fuel as is feasible and prudent in the circumstances of the flight and in accordance with the Flight Manual.

6. Current requirements do not differentiate between those cases where the rules are met by a positive performance and those where the procedure used allows the aircraft to 'drift down' to the calculated stabilizing altitude. In the latter case, the legislation permits the assumption that the pilot always knows precisely where he is and immediately takes the appropriate action in the event of engine failure.

7. In view of the relative infrequency with which a combination of engine failure, significant obstructions and inadequacy of navigational aids is likely to be met, it was considered unjustifiable to introduce specific accountability for the drift-down case into the Regulations. However, information to cover drift-down is included in the Operations Manual. One method of satisfying the requirement is to

assume that the aeroplane passes over the critical obstruction with a clearance of not less than 2,000 ft after an engine failure occurs no closer than the nearest fix from a radio or internal navigation aid. The take-off weight of the aeroplane may need to be adjusted in order to provide this capability.

8. When compliance is met by 'drift-down' procedure, the maximum permissible altitude that can be assumed at the start of the drift down is the least of:

a. Maximum permissible altitude for power unit restarting.

b. The planned altitude to fly.

9. Once the maximum permissible altitude is established, the drift-down of the aeroplane is calculated using the en route net-gradient of climb 'one or two power units inoperative' as applicable. This drift-down is plotted and the vertical clearance of 2,000 ft is checked against all obstacles defined in paras 2, 3 and 4. If this clearance is not achieved, either the aeroplane weight must be adjusted until the required clearance exists or, alternatively, the aeroplane must be re-routed to avoid critical obstacles. It should be noted that, in certain circumstances, it is possible for the aeroplane to 'drift up'.

10. The en route net gradient of climb graphs in the ODM are entered with temperature (ISA deviation), aeroplane altitude (pressure) and weight to extract a percentage gradient, which can be positive or negative.

11. The rules governing two-engine aeroplanes are laid down in ANR 7(5)(b) and (c) and are subject to frequent changes.

Calculation of En Route Flight Path After Power Failure

12. The method used for calculating and plotting the en route flight path after power unit failure is to:

a. Determine the maximum height from which the drift-down will commence (see para 8).

b. Determine the required obstacle clearance.

c. Select the vertical spacing to be used in the calculation e.g. 2,000 ft. The number of points plotted should be sufficient to enable a smooth curve to be drawn.

d. For the mid-height of the band chosen and associated temperature, obtain from the appropriate graph the en route net gradient.

e. Calculate the horizontal distance travelled during descent using the following formula:

$$\text{Gradient \%} = \frac{\text{Change in Height}}{\text{Change in Distance}} \times \frac{100}{1}$$

$$\therefore \text{ Change in Distance} = \frac{\text{Change in Height}}{\text{Gradient \%}} \times \frac{100}{1}$$

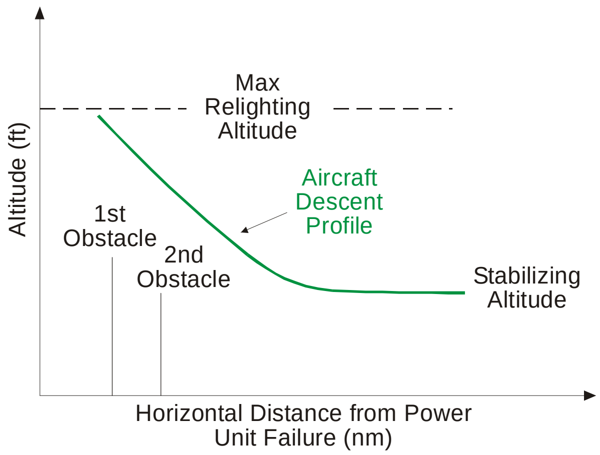
$$\therefore \text{ Change in nautical miles} = \frac{\text{Change in Height} \times 100}{\text{Gradient} \times 6080\text{ ft}}$$

$$\therefore \text{ For 1,000 ft vertical band} = \frac{1000 \times 100}{\text{Gradient} \times 6080}$$

$$= \frac{16.44}{\text{Gradient}}$$

f. The horizontal distance travelled for each vertical band is plotted against the lower altitude of the band and not the mid-height. A drift-down plot is shown in Fig 1.

2-14 Fig 1 Plot of Drift Down En Route Flight Path



Effect of Wind on Drift Down Path

13. The horizontal distances calculated from the gradients extracted from the ODM are, in effect, still air distances. To obtain the true distance/gradient the forecast wind component must be applied; a head component decreases the distance travelled (increases the gradient), whilst a tail component will give an increased horizontal distance (decreased gradient):

Example:

Net gradient of climb	= –3.0%
TAS	= 420 kt
Headwind component	= 60 kt

The ground speed is 360 kt; then the true gradient relative to the ground is given by:

$$-3.0 \times \frac{420}{360} = -3.5\%$$

and the associated change in horizontal distance would then be calculated by using a new gradient of –3.5%.

An alternative method is not to adjust the gradient, but simply to adjust the horizontal distance:

$$\text{Corrected Distance} = \frac{\text{Calculated Distance} \times \text{G/S}}{\text{TAS}}$$

This calculation can be solved on a DR computer (see Volume 9, Chapter 8). Set Groundspeed on the outer scale against True Airspeed on the inner scale. Set Calculated Distance on the inner scale and read off the Corrected Distance on the outer scale.

CHAPTER 15- LANDING REQUIREMENTS

Introduction

1. The final set of requirements which needs investigation concerns landing distance criteria which deal with destination and alternate aerodromes separately. The landing requirements cover the approach and landing stage of performance planning, commencing at a height of 1,500 ft above the landing surface to coming to a stop on the runway.

2. As with the take-off, requirements can be treated under similar headings, i.e.:

a. C of A limit (landing).

b. WAT limit (landing).

c. Landing distance requirements.

Landing Planning Considerations

3. **C of A Limit**. The C of A limit specifies the maximum structural landing weight; it is absolute and must not be exceeded except in emergency. This is normally listed in the aeroplane Operating Data Manual (ODM) under 'Limitations'.

4. **WAT Limit**. Air Navigation (General) Regulations 7(6) requires that "weight will not exceed the maximum landing weight for the altitude and the expected air temperature for the estimated time of landing at the aerodrome of intended landing and at any alternative aerodrome". This is the WAT limit, which is designed to ensure compliance with the minimum requirements specified in BCARs, which lay down specified gradients of climb or descent with one power unit inoperative or all power units operating. The aeroplane will be certificated under one of the following requirements:

a. Continued approach - all power units operating.

b. Continued approach - one power unit inoperative.

c. Discontinued approach - one power unit inoperative.

d. Balked landing - all power units operating.

5. The limit is readily obtainable from a single graph in the ODM, entering with arguments of aerodrome pressure altitude and temperature, then extracting maximum weight for these conditions. The WAT limit graph takes no account of landing distance available.

Landing Distance Requirements

6. Air Navigation (General) Regulations 7(7) states that landing distances required, respectively specified as being appropriate to aerodromes of destination and alternate aerodromes, do not exceed at the aerodrome at which the aircraft is intended to land or at any alternate aerodrome, as the case may be, the landing distance available, after taking into account the following factors:

a. The landing weight.

b. The altitude at the aerodrome.

c. The temperature in the specified international standard atmosphere appropriate to the altitude at the aerodrome:

d. A level surface in the case of runways usable in both directions, provided that the landing distances available are the same in both directions.

e. The average slope of the runway in case of runways usable in only one direction, or of differing landing distances.

f. Still air conditions in the case of the most suitable runway for a landing in still air conditions.

g. Not more than 50% of the forecast wind component opposite to the direction of landing or not less than 150% of the forecast wind component in the direction of landing in case of the runway that may be required for landing because of forecast wind conditions.

7. Currently, an aeroplane's landing distance performance may conform to one of 2 sets of landing distance required criteria known as Arbitrary Landing Distance and Reference Landing Distance. The main differences between these 2 requirements are to be found in the associated conditions and the safety factors to be applied. The set of requirements to which the particular aeroplane type will operate, will be stated in the ODM.

8. **Arbitrary Landing Distance**. This is the gross horizontal distance required to land on a dry hard surface from a screen height of 50 ft and come to a complete stop, following a steady descent to the 50 ft height point at a gradient of descent not greater than 5% (3º glide path) and at a constant speed of not less than the greatest of:

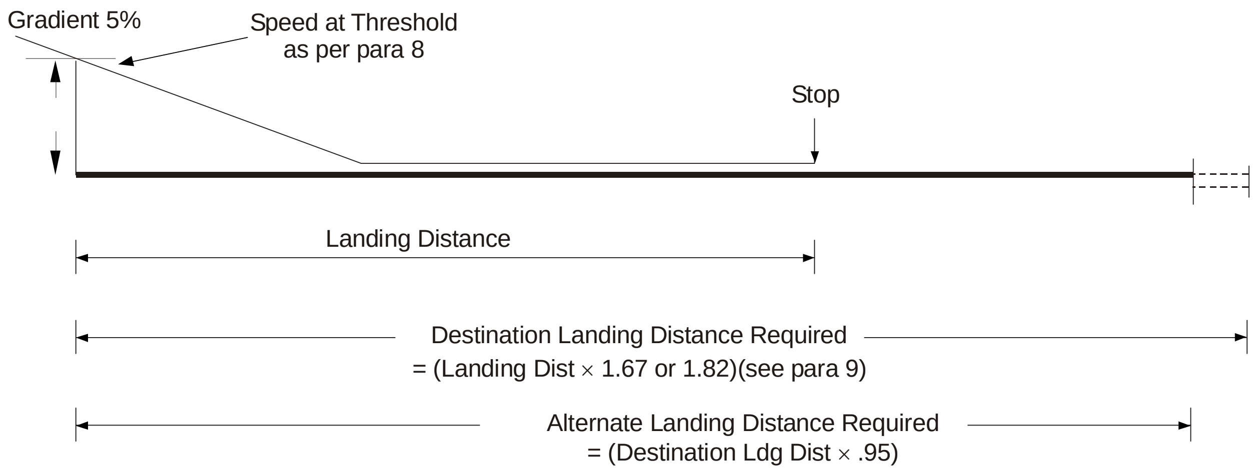
a. $1.3 \times$ VMSO.

b. VAT_0.

c. VAT_1 – 5 kt.

9. **Arbitrary Landing Distance Required**. The landing distance required appropriate to destination aerodrome shall be the arbitrary landing distance (determined in accordance with para 8) $\times$ 1.82 for aeroplanes having turbojet power units and not fitted with effective reverse thrust; $\times$ 1.67 for all other aeroplanes. The landing distance required appropriate to alternate aerodrome shall be the landing distance required appropriate to destination aerodrome multiplied by 0.95 (see Fig 1).

2-15 Fig 1 Arbitrary Landing Distance Required - Destination and Alternate Aerodromes



10. **Reference Landing Distance**. This is the gross horizontal distance required to land on a reference wet hard surface from a height of 30 ft and come to a complete stop. The speed at a height of 30 ft is the maximum threshold speed (VAT_0 + 15 kt); touch down is at the Reference Touch Down Speed, after a steady gradient of descent to the 30 ft point of not greater than 5% (3º glide path) under limiting operational conditions of ceiling and visibility.

11. **Reference Landing Distances Required**. The landing distance required appropriate to a destination aerodrome (Fig 2) shall be the greater of:

a. The reference landing distance 'all power units operating' (determined in accordance with para 10) × the 'all power units operating' field length factor of:

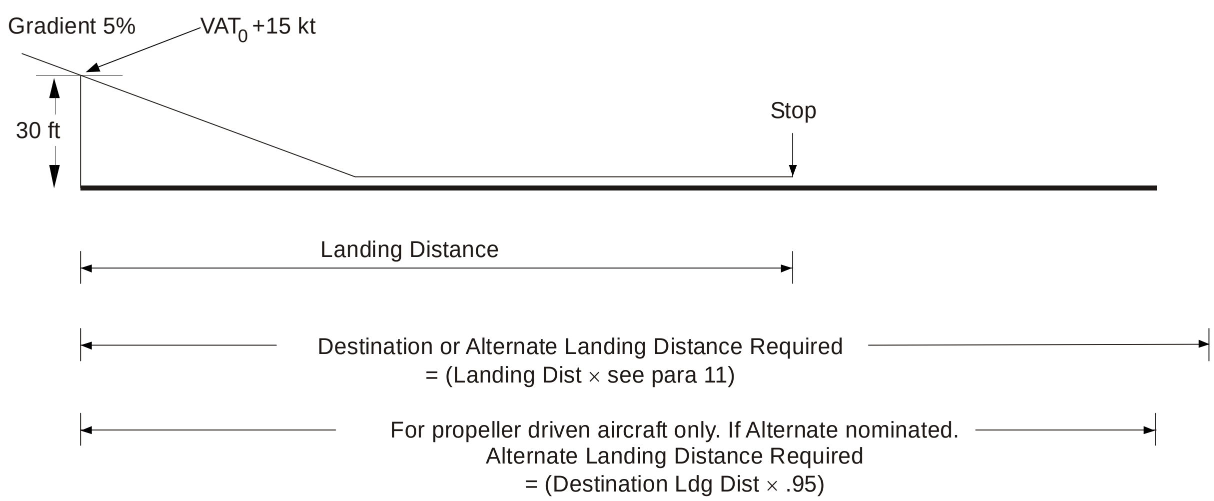
$$1.24 - 0.1\frac{CDG}{CDA} \text{ or 1.11, whichever is the greater.}$$

b. The reference landing distance 'one power unit inoperative' (determined in accordance with para 10) × the 'one power unit inoperative' field length factor of:

$$1.19 - 0.2\frac{CDG}{CDA} \text{ or 1.08, whichever is the greater.}$$

CDG and CDA are the coefficients of drag when in contact with the ground and when airborne respectively.

2-15 Fig 2 Reference Landing Distances Required - Destination and Alternate Aerodromes



12. **Limiting Landing Weight**. Paras 8 to 11 covered the landing distance requirements for the destination and alternate aerodrome. Each must be considered separately and the most limiting will govern the maximum permissible landing weight (C of A and WAT limits apart). When comparing the destination and alternate to obtain the limiting landing weight, allowance must be made for fuel burn-off between destination and alternate.

13. **Calculation**. Procedure for determining the limiting landing weight is given below:

a. **Jet Aircraft - Destination Aerodrome**.

(1) The landing distance required must not exceed the landing distance available on the runway most suitable for landing in still air conditions (using runway slope if required).

(2) The landing distance required must not exceed the landing distance available on the runway which may have to be used in forecast wind (factored) conditions (using runway slope if required).

(3) Compare the weights obtained in (1) and (2) above with those of the C of A and WAT limits; the least weight will be the maximum permissible landing weight for a destination aerodrome.

b. **Jet Aircraft - Alternate Aerodrome**. The procedure to resolve the maximum permissible landing weight at an alternate aerodrome is identical to that for the destination aerodrome.

c. **Propeller-driven Aircraft - Destination Aerodrome**.

(1) The landing distance required must not exceed the landing distance available (using destination scale) on the runway most suitable for landing in still air conditions (using runway slope if required).

(2) The landing distance required must not exceed the landing distance available on the runway which may have to be used in forecast wind (factored) conditions using the destination scale if no alternate has been nominated in the flight plan, or the alternate scale if an alternate has been nominated (using runway slope if required).

(3) Compare the weights obtained in (1) and (2) above with those of the C of A and WAT limits; the least weight will be the maximum permissible landing weight for a destination aerodrome.

d. **Propeller-driven Aircraft - Alternate Aerodrome.** The procedure to resolve the maximum permissible landing weight at an alternate aerodrome is identical to that for the destination aerodrome except that the landing distance required is calculated by use of the alternate scale throughout.

14. **Multiple Runways.** If aerodromes have multiple runways, then each runway will need to be treated individually, unless the best runway can be determined by inspection. Generally, the still air case will always be the more limiting unless a tail wind prevails, or in the multi-runway case, the forecast wind requires that a shorter runway be considered (e.g. if the crosswind component were outside the aeroplane limits for landing on the longest equivalent runway).

Summary

15. Having now decided on the maximum permissible landing weight, this could impose a restriction on take-off weight since maximum take-off weight can never exceed maximum landing weight plus burn-off fuel from departure to destination or alternate aerodrome.

Note: In this chapter reference has been made to landing distance required or the maximum permissible landing weight; these 2 terms are complementary and amount to the same thing.

CHAPTER 16- OPERATING DATA MANUAL (ODM)

Introduction

1. Traditionally Scheduled Performance and flight planning information is presented to UK military aircrew in the Operating Data Manual (ODM), Topic 16 of the Aircraft Document Set. The ODM is issued for a specific aircraft type, fixed or rotary wing. It is used in conjunction with the handling techniques and operating procedures in the Aircrew Manual, and subject to the overriding limitations contained in the Release to Service. In content the ODM is broadly equivalent to the performance section of a civilian Aircraft Flight Manual (AFM) and is regulated by MAA Regulatory Article (RA) 1310. Defence Aircrew Publications Squadron provide Subject Matter Expertise for the provision and content of ODM together with performance advice.

2. The ODM contains all the performance information necessary for the performance, and flight, planning of any flight the aircraft is likely to be required to undertake. The ODM calculations provide performance limitations, e.g. the maximum weight at which a take-off can be made from a specific runway in the relevant ambient conditions, and these limitations must be considered mandatory.

3. The ODM will typically state the regulatory basis for the data contained and many of the regulatory required margins will already be applied to reduce the calculations required. The data represented in the ODM is to the Normal Operating Standard (NOS) (see Volume 2, Chapter 9 Paragraph 11). NOS is broadly equivalent to civilian standards and provides an equivalent standard of safety. For some types, particularly fast-jets, NOS will not replicate civilian standards but will be as near to them as can be practically achieved for the type. To provide increased operational capability the ODM may also contain data to the Reduced Operating Standard (ROS) and/or to the Military Operating Standard (MOS). This increased capability is achieved by the deliberate erosion of performance safety margins, making ROS and MOS operations approximately 10 times and 100 times the risk of NOS respectively. Typically MOS operations provide no allowance for engine failure. Due to the increased risk inherent in the use of ROS or MOS data, explicit authorization is required, which is regulated via MAA RA1330.

4. Dependent of aircraft type the ODM will consist of one or more volumes. A fast-jet ODM can normally be contained in one book. For heavy fixed wing aeroplanes it is customary for the ODM to be divided into two books. Book 1 contains the Scheduled Performance data and Book 2 Flight Planning data. A helicopter ODM is often augmented by Rapid Planning Data for in-flight use.

5. Many in service aircraft have a means of providing performance planning data to aircrew that either compliments or replaces the traditional ODM. Some of these alternates are:

a. Civilian aircraft types will typically be operated using the civilian AFM, either hard copy or supplied electronically via interactive screens on the aircraft or on Electronic Flight Bags.

b. American supplied types may be operated from the American supplied documents.

c. Flight planning may be available via flight planning software and some types have type specific scheduled performance electronic performance planning aids.

d. Quick-look performance data is often contained within the Flight Reference Cards (FRCs) or Flight Crew Checklists (FCCs).

However performance planning data is provided, it remains a fundamental requirement that it is consistent with the certified source data, either contained within the ODM, AFM or other specified document.

Presentation of Data

6. Performance planning as outlined in the previous chapters is a lengthy process when using basic methods. When the field lengths were 'balanced' (i.e. ASDA=TODA), the maximum take-off weight was produced in a reduced time, albeit with some penalty to take-off weight if clearway does in fact exist.

7. In the D and X and the more recent D and R graphs, comparison is made between the unbalanced lengths of TODA/ASDA and TORA/ASDA. By adjustment in the construction of the graphs, the above-mentioned unbalanced lengths are equated into equivalent balanced field lengths, the least of which is taken to be limiting. Associated with each equivalent balanced field length is either an X or a V_1/V_R value, both serving the same function (the X being converted to a V_1/V_R ratio at a later stage). The X or V_1/V_R ratio used is the one which is associated with the limiting D (or R) value. This need not necessarily be the smallest value of X or V_1/V_R. A single graph gives the maximum weight for a limiting value chosen (D or R). In some cases a separate graph may be supplied for both D factor and R factor. Conversion of V_1/V_R ratio or X to V_1 completes the take-off weight problem as far as runway criteria is concerned except for the checking of the all power units operating limitation and maximum permissible V_1/V_R ratio. This solution may be presented as an individual graph, actual TODA being used and not the D factor. Volume 2, Chapter 17 contains more detailed information on D and X graphs and the method of using them.

8. In some ODMs, the 'all power units operating' case and the 'one power unit inoperative' case are both presented on one graph, thus giving the most rapid solution when using ODMs. It must be stressed that, since this gives field length limiting values, no account is taken in calculation for maximum permissible take-off weight or limitations which can be imposed by possible net take-off flight path, en route terrain clearance and landing distance requirement restrictions.

Regulated Take-off Graphs (RTOGs)

9. A further step in the simplification of the take-off performance calculation is the Regulated Take-off Graph (RTOG) (Fig 1). This is a simple graph, for an aeroplane type, constructed for individual runways at a chosen aerodrome taking account of variable atmospheric conditions (i.e. temperature, wind and pressure) and field length limitations (i.e. take-off field lengths, elevation and slope). Entry into the graph against temperature, through the reported wind component, produces the maximum take-off weight, V_1 and V_R. The data may also be presented in table form.

10. A book of these graphs is produced to cover all commonly used aerodrome runways. The graphs may incorporate modifications demanded by noise limitations, tyre speed limitations, and net take-off flight path with instructions, if necessary, for executing a planned, or emergency, turn. It is possible to match strange aerodrome criteria, for which no graph has been prepared with those for which a graph is produced, or with specially prepared balanced field length graphs. This obviates the need to calculate the performance plan from first principles. Where the runway criteria is such that the WAT limit imposes the only restriction, then the runway may appear as an entry in a table of limiting conditions. The table generally lists:

a. The runway.

b. Aerodrome dimensions.

c. Limiting wind component.

d. Either a correction to V_R to obtain V_1, or the V_1/V_R ratio.

Conclusion

11. The quickest and most accurate method of solving performance problems is to use a computer with specially written software. Such software has been in use by aeroplane manufacturers and large fleet operators for some time and is now more generally available. Aircrew should note that emergency turn data produced electronically often does not adhere to an approved standard instrument departure profile.

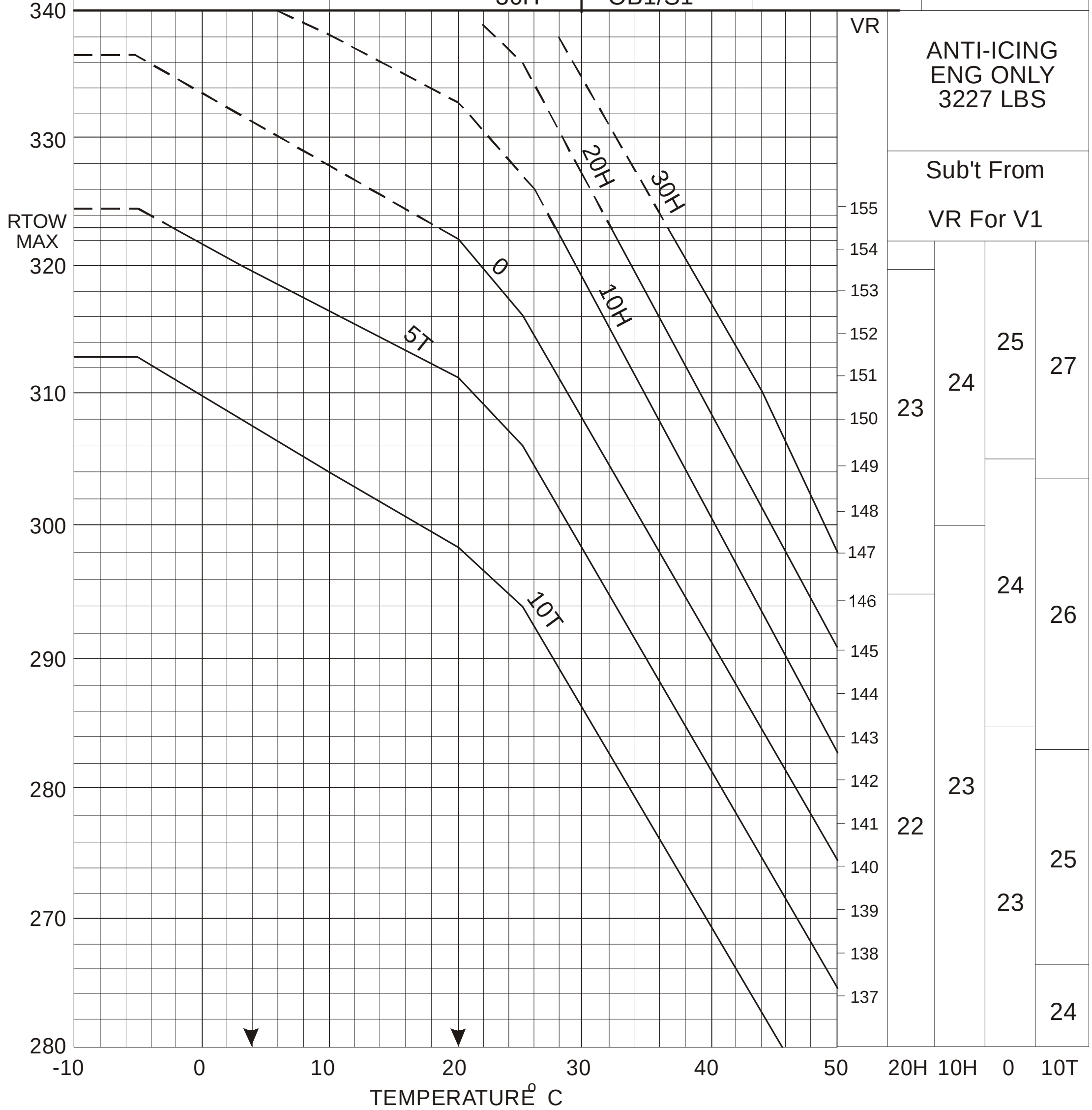
12. The main end-product in performance planning is finding the value of the maximum permissible regulated take-off weight. It should be clear, from this and the preceding chapters, that this weight may be determined by the take-off distance, the net take-off flight path, the en route terrain clearance, or the landing distance limitations, depending upon the particular circumstances.

2-16 Fig 1 Example of an RTOG

LIVERPOOL				RUNWAY 27	ISSUE JUL 91
TORA	7500	ASDA	7602	TODA	8100
LDA	7500	SLOPE	0.25DN	ELEV	77

OBSTACLE	1	2	3	4	5	6
DISTANCE	8100					
HEIGHT	-12					

LIMITS	WIND				
		10T	OB1/S1		
		5T	OB1/S1		
		0	OB1/S1		
		10H	OB1/S1		
		20H	OB1/S1		
		30H	OB1/S1		



PRESSURE HEIGHT CORRECTIONS			LANDING WEIGHT
TEMPERATURE RANGE			10T WIND 223171 LBS
5 °C	15 °C	25 °C	7T WIND 233000 LBS
+716 LB per - 100ft -630 LB per + 100 ft	+648 LB per - 100ft -727 LB per + 100 ft	+896 LB per - 100ft -1178 LB per + 100 ft	LDA 7500 FT

CHAPTER 17 - D AND X GRAPHS

Introduction

1. With the Civil Aviation Authority's standard performance charts as used in the licensing examination being presented in a form which is typical of that used in modern flight manuals, many operators will be using an aeroplane Operating data manual (ODM) which still uses the D and X (or D and R) method of presenting the field length data to be used to determine a maximum weight for the specified conditions. In these charts, the two variables D and X are utilized in an intermediate stage of the calculations, but the graphs can be used without an exact knowledge of their physical meaning.

General Description

2. Although it is not necessary to understand the significance of D and X in order to use the graph successfully, the following points are added for interest:

a. D represents the equivalent balanced field length, taking account of slope, wind etc.

b. X represents a correction to the respective deviation from the balanced field power failure speed ratio.

c. The values of X are ratios of V_1 and V_R for a balanced field, but the value of 100 is appropriate to the balanced field length.

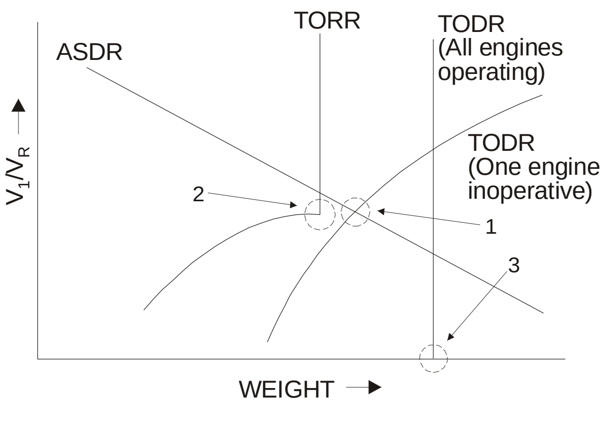
d. Fig 1 shows the normal plot of the values of ASDR, TORR (all engines and one engine inoperative), TODR (all engines and one engine inoperative) and the following points should be noted:

(1) X is 100 when TODA and ASDA are equal in still air with nil slope. Thus with a TODA of 6,000 ft and an ASDA of 6,000 ft in still air with nil slope, the D and X values obtained from this graph would be 6,000 and 100 respectively.

(2) In the TORA/ASDA graph, the kink in the D values represents the point where the all engines TORR becomes limiting.

(3) TODR (all engines operating) gives the value of weight which satisfies the all engines operating factorized TOD requirements.

2-17 Fig 1 Significant Point on Final Take-off Analysis Plot



3. Graphs are usually supplied for the following cases:

a. D and X for TODA/ASDA.

b. D and X for TORA/ASDA.

c. D for TODA (all engines operating). (Note: Some aeroplanes do not use this system, but supply a graph to check that the TODR with all engines operating does not exceed the TODA).

d. Take-off weight for a given value of D.

e. Power failure speed ratio for a given value of X.

f. Limitations of power failure speed ratio (brake energy considerations).

Detailed Instructions

4. Resolution of take-off weight is as follows:

a. Enter the first graph with TODA and ASDA appropriate to the runway to be used, allowing for runway slope and wind as necessary. The intersection of the two lines so determined gives values of D and X which should be noted.

b. Enter the next graph TORA and ASDA and, working through runway slope and wind component, extract second values of D and X.

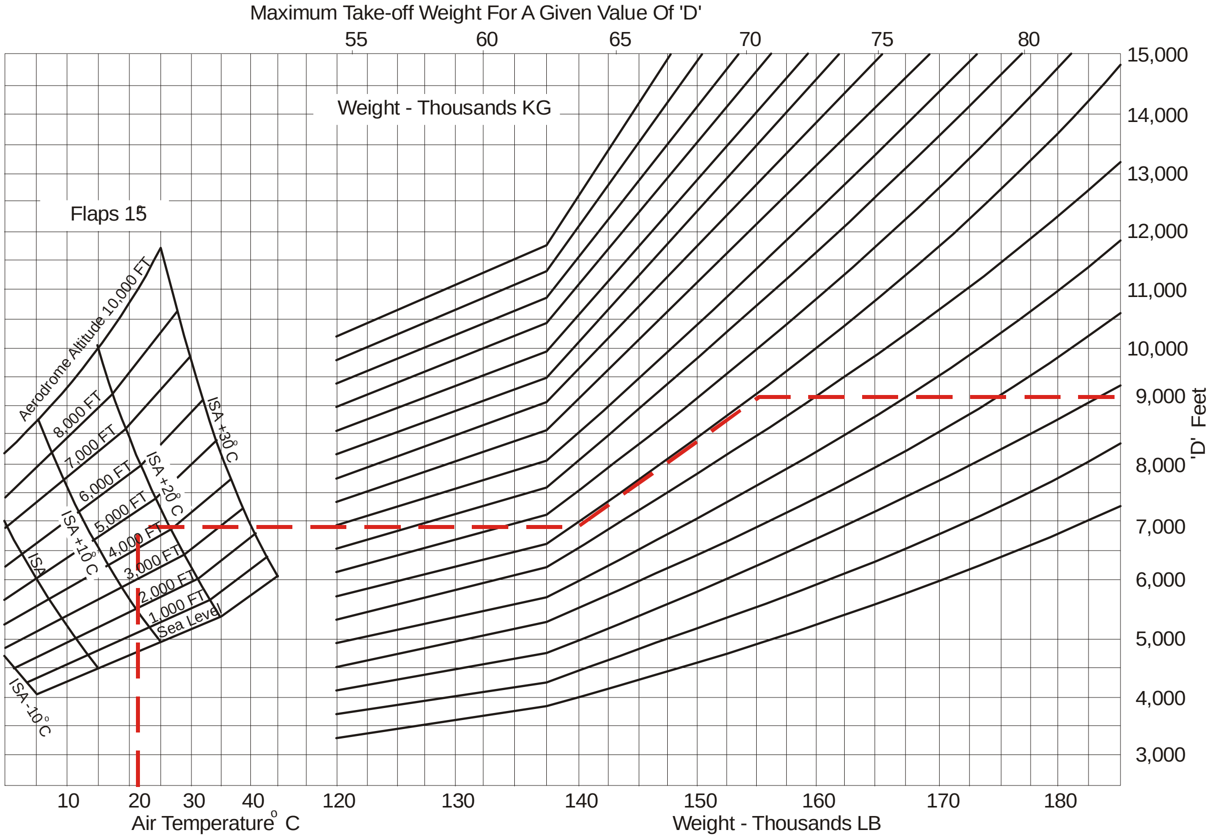
c. Enter the third graph (where supplied) and obtain the value of D for the all engines operating case.

d. Compare the three values of D; the lowest value represents the critical case and that value of D is then used to obtain the maximum take-off weight. If this value of D is obtained from the two graphs showing ASDA/TORA or ASDA/TODA use the value of X associated with the appropriate D. If the lowest value of D comes from the take-off distance all engines operating case, then the value of X is taken from the graph giving the next lowest value of D. In those ODMs where a value of D is not obtained for the take-off distance (all engines), then it is necessary, once the take-off weight has been obtained, to check that the TODR (all engines) does not exceed the TODA.

e. The next graph is entered with aerodrome altitude and temperature; the take-off weight corresponding to the critical value of D is thus obtained (see Fig 2).

f. The value of the power failure speed ratio corresponding to the critical value of X is then obtained. It is now necessary to check that the power failure speed ratio does not exceed the limiting value for the brake energy consideration.

2-17 Fig 2 Example of Max TOW/D Chart



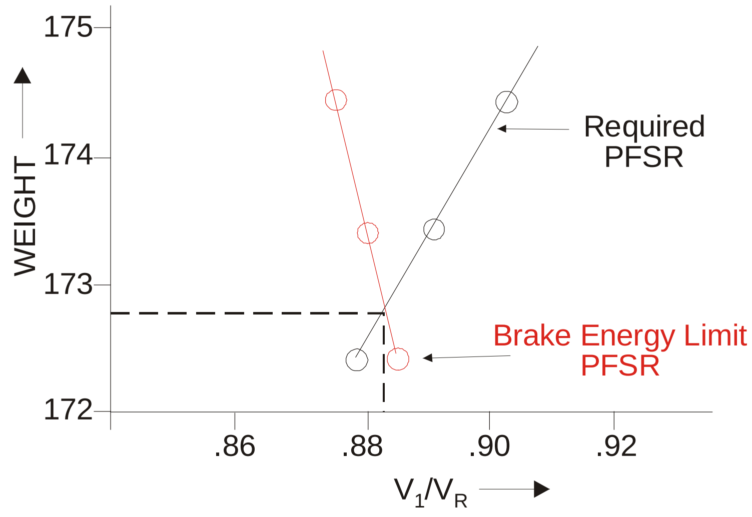
5. If it is found that the take-off weight and power failure speed ratio do not meet these requirements, recalculation is then necessary.

6. In the brake energy limitation case, this can be done in one of the following ways:

a. By having a graph showing the correction to take-off weight for brake energy limitations. With this graph one enters with the case that is limiting (ie either TODA/ASDA or TORA/ASDA) and the calculated power failure speed ratio (PFSR). Against the maximum permitted value of PFSR a percentage correction to take-off weight is obtained to satisfy the brake energy requirement.

b. Reducing the value of ASDA by an arbitrary amount (say 500 ft) and recalculating D and X. A new take-off weight is then obtained together with a new value of PFSR. If brake energy is limiting, a new maximum value of PFSR is obtained with the new take-off weight. This process is repeated for other values of ASDA and a graph is plotted for both sets of PFSR against weight. The intersection of the two lines so obtained then gives maximum take-off weight and associated PFSR as shown in Fig 3.

2-17 Fig 3 Plot of PFSRs Against Take-off Weight



7. The intersection of the two lines (in Fig 3) gives the take-off weight and the corresponding PFSR. From this the take-off distance (all engines) operating requirement must again be checked.

Range of V_1 Using D And X Graphs

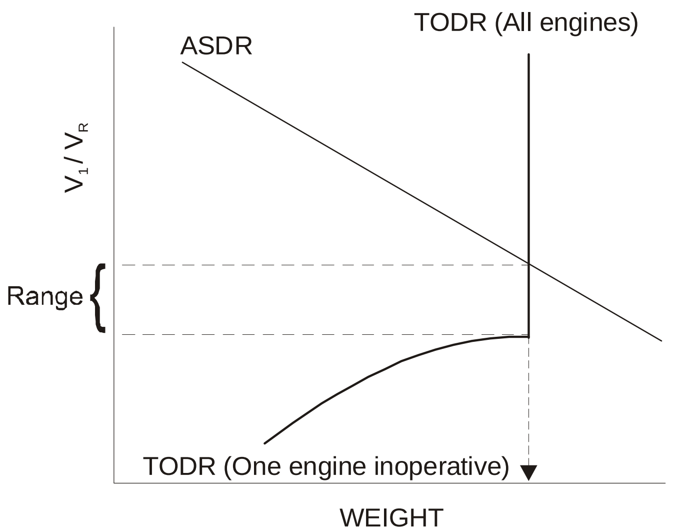
8. It is possible from the D and X graphs to obtain a range of V_1 under certain conditions:

a. **Take-off Weight Equal to Maximum Take-off Weight**.

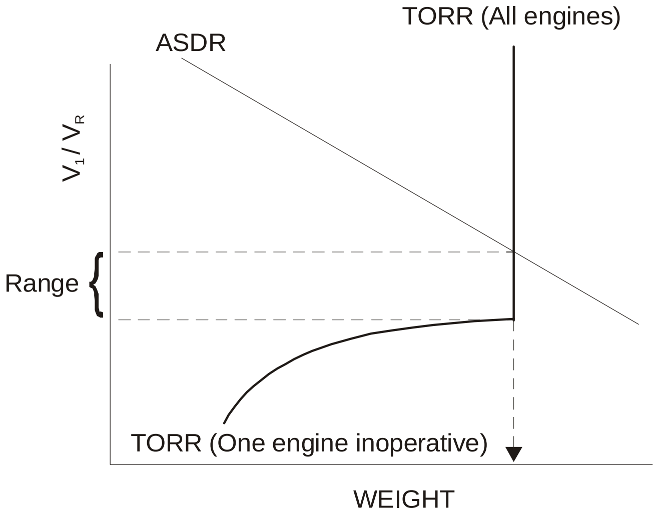
(1) When TOD (All Engines) is Limiting (Fig 4a).

(2) When TOR (All Engines) is Limiting (Fig 4b). In this case, using ASDA the X value would be read as the intersection of the TORA and ASDA lines (see Point A in Fig 5). For the same take-off weight the X value can be as low as Point B (Fig 5). Hence a useable range of X exists giving a range of V_1 if desired.

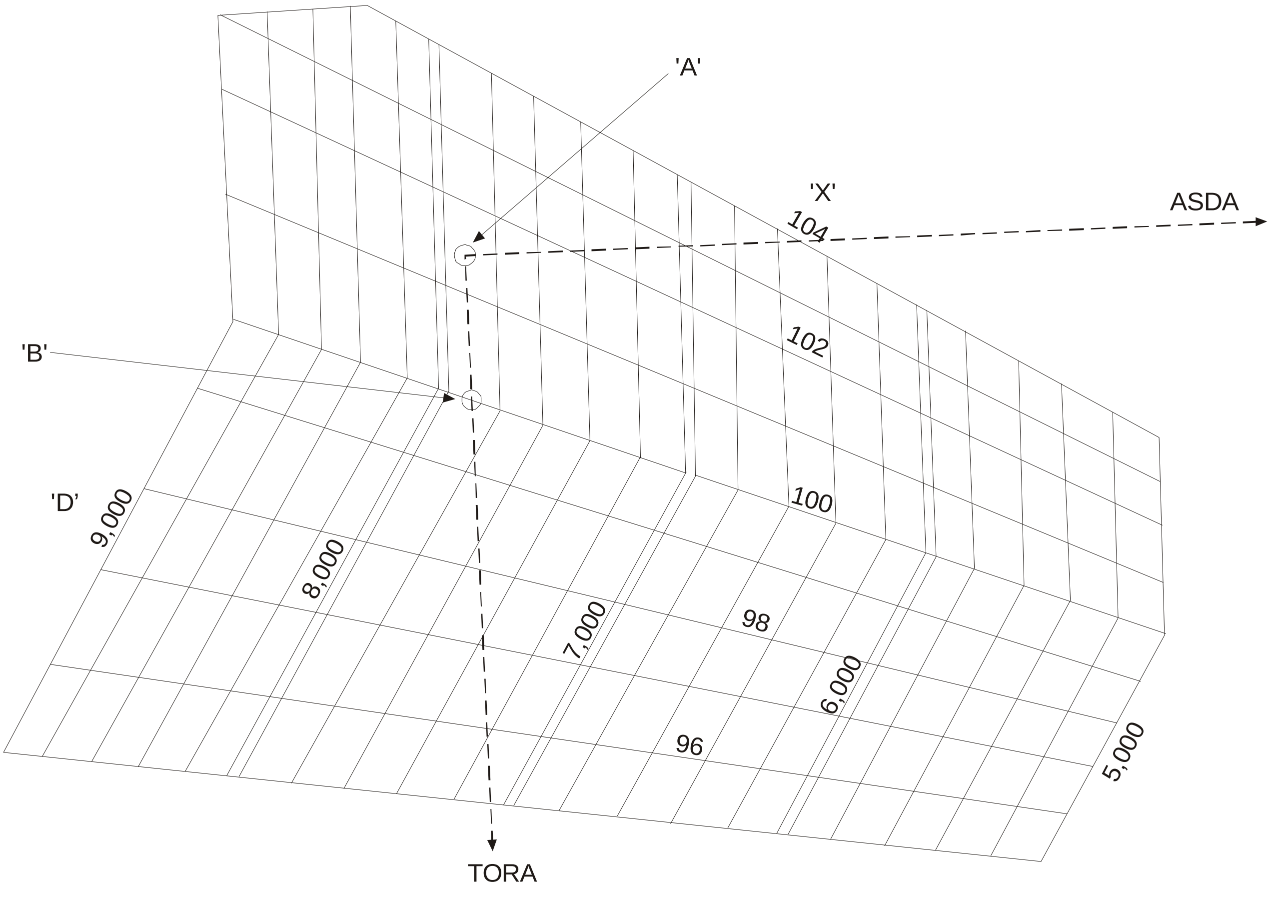
2-17 Fig 4 Take-off Distance (All Engines) Limiting



2-17 Fig 5 Take-off Run (All Engines) Limiting



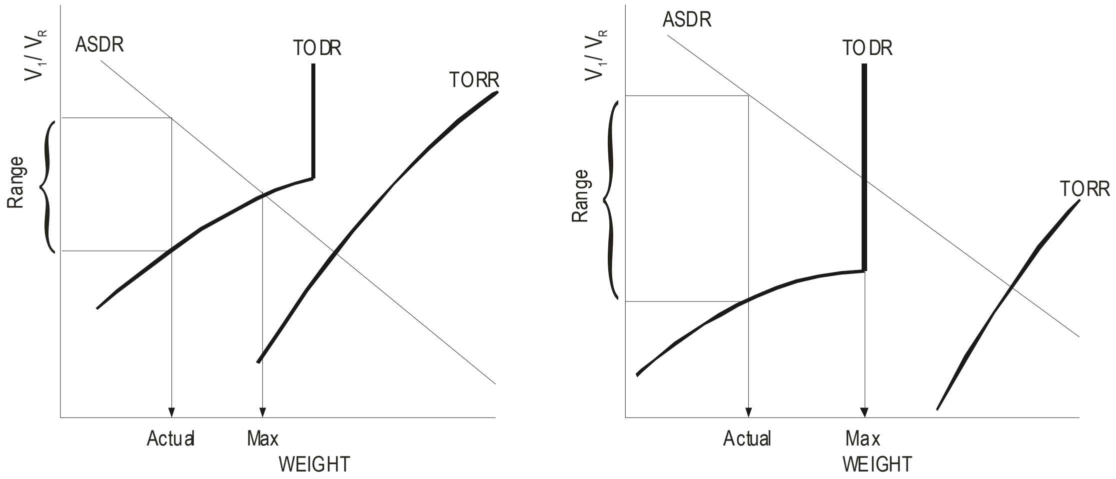
2-17 Fig 6 D and X Matrix



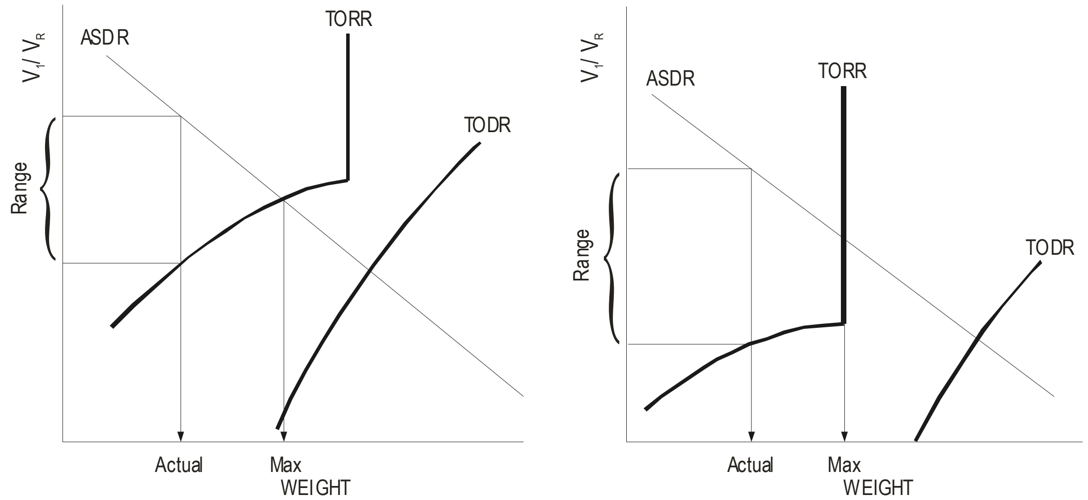
b. **Take-off Weight Less Than Maximum Take-off Weight.** There will always be a range of V_1 when the take-off weight is less than the maximum take-off weight. The usable range will be governed by take-off run or take-off distance whichever is limiting in determining maximum take-off weight (see Fig 6).

2-17 Fig 7 Final Take-off Analysis - Less than Maximum Take-off Weight

a Take-off Distance Limiting



b Take-off Run Limiting



9. **Method**.

a. Enter the graph to find D appropriate to the actual take-off weight.

b. Enter whichever graph was limiting on maximum take-off weight case (i.e. TORA or TODA).

c. Enter with TORA or TODA and go through to meet the required value of D and read off the value of X (this is the minimum value of X).

d. Re-enter with ASDA and go up through the graph to meet the required value of D and read off the value of X (this is the maximum value of X).

e. With these two values of X, find the maximum and minimum V_1 and ensure that the maximum permitted value is not exceeded.

Note: The maximum weight given by field length consideration may not always be the maximum permissible take-off weight for the flight. D and X graphs do not take account of other performance regulations e.g. WAT limitations, net flight path, etc.

10. The explanation and method of use, stated in the previous paragraphs, hold good for the more recent D and R graphs, where the main change is that instead of extracting X values, to be converted to a V_1/V_R ratio at a later stage, a V_1/V_R ratio is obtained directly.

CHAPTER 18 - GRAPHS

Introduction

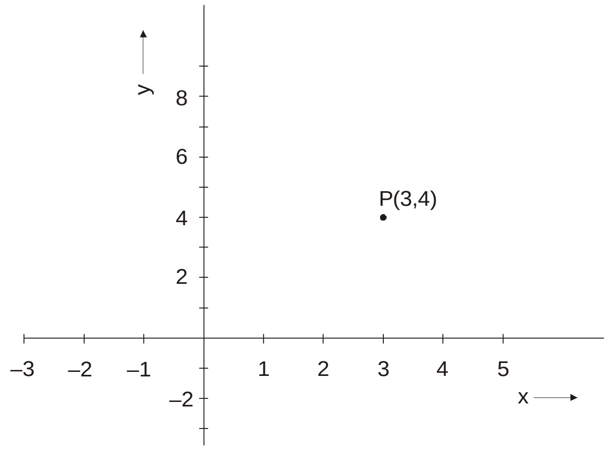
1. The term 'graph' is usually applied to a pictorial representation of how one variable changes in response to changes in another. This chapter will deal with the form of simple graphs, together with the extraction of data from them, and from the 'families of graphs' and carpet graphs that are frequently encountered in aeronautical publications. Although not strictly a 'graph', the Nomogram will also be covered. The pictorial representation of data in such forms as histograms, frequency polygons, and frequency curves will be treated in Volume 13.

Coordinate Systems

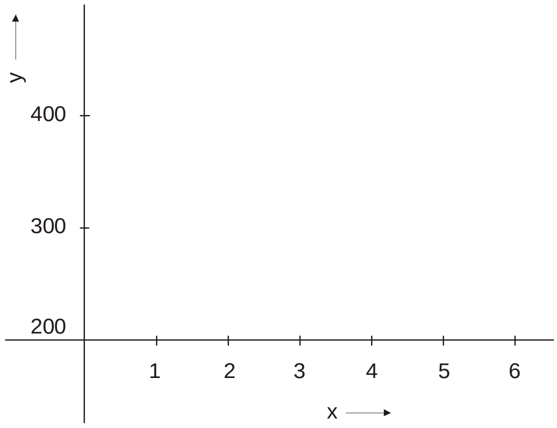
2. Graphs are often constructed from a table of, say, experimental data which gives the value of one variable, x, and the experimentally found value of the corresponding variable, y. In order to construct a graph from this data it is necessary to establish a framework or coordinate system on which to plot the information. Two such coordinate systems are commonly used: Cartesian coordinates and Polar coordinates. Both systems will be described below, but the remainder of this chapter will be concerned only with the Cartesian system.

3. **Cartesian Coordinates**. Cartesian coordinates are the most frequently used system. Two axes are constructed at right angles, their intersection being known as the origin. Conventionally the horizontal 'x' axis represents the independent variable; the vertical 'y' axis represents the dependent variable, i.e. the value that is determined for a given value of x. Any point on the diagram can now be represented uniquely by a pair of coordinate values written as (x,y) provided that the axes are suitably scaled. It is not necessary for the axes to have the same scale. Thus, in Fig 1, the point P has the coordinates (3,4), i.e. it is located by moving 3 units along the x axis and then vertically by 4 'y' units. It is sometimes inconvenient to show the origin (0,0) on the diagram when the values of either x or y cover a range which does not include 0. Fig 2 shows such an arrangement where the x-axis is scaled from 0 but the corresponding values of y do not include 0. The intersection of the axes is the point (0,200). It should be noted from Fig 1 that negative values of x or y can be shown to the left and below the origin respectively.

2-18 Fig 1 Cartesian Coordinates

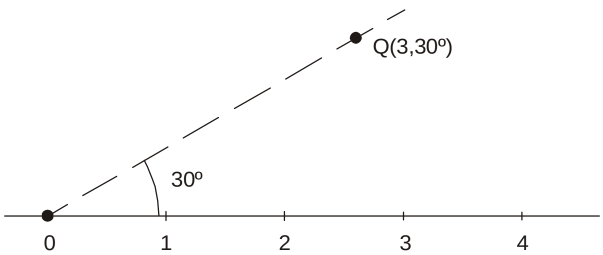


2-18 Fig 2 Cartesian Coordinates - Displaced Origin



4. **Polar Coordinates**. Polar coordinates specify a point as a distance and direction from an origin. Polar coordinates are commonly encountered in aircraft position reporting where the position is given as a range and bearing from a ground beacon; they are also used in certain areas of mathematics and physics. As with Cartesian systems it is necessary to define an origin, but only one axis or reference line is required. Any point is then uniquely described by its distance from the origin and by the angle that the line joining the origin to the point makes with the reference line. The coordinates are written in the form (r,θ), with θ in either degree or radian measure. Conventionally, angles are measured anti-clockwise from the reference line as positive and clockwise as negative. Fig 3 illustrates the system. Point Q has the coordinates (3, 30º) or (3, –330º) in degree measure; $(3, \frac{\pi}{6})$ or $(3, \frac{-11\pi}{6})$ in radian measure.

2-18 Fig 3 Polar Coordinates



The Straight Line Graph

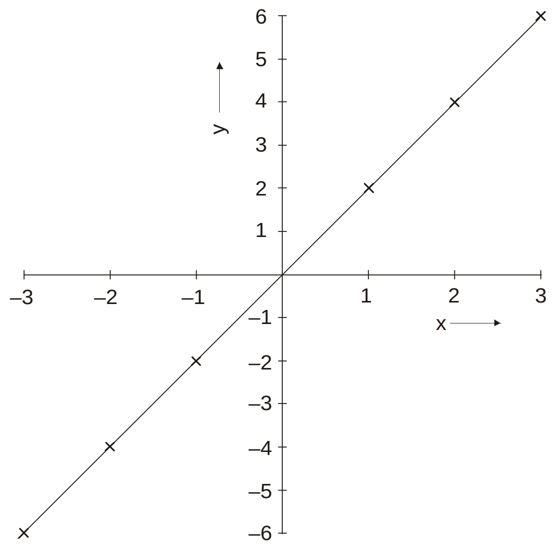
5. Table 1 shows a series of values of x and the corresponding values of y. Fig 4 shows these points plotted on a graph.

Table 1 Values of x and y

x	–3	–2	–1	0	1	2	3
y	–6	–4	–2	0	2	4	6

It will be seen that all the points lie on a straight line which passes through the origin. It is clear from the table of values that if the value of x is, say, doubled then the corresponding value of y is also doubled. Such a relationship is known as direct proportion and the graphical representation of direct proportion is always a straight line passing through the origin. In general the value of y corresponding to a value of x may be derived by multiplying x by some constant factor, m, ie: $y = mx$. In the example, m has the value 2, i.e. $y = 2x$. Because such a relationship produces a straight-line graph, it is known as a linear relationship and y = mx is known as a linear equation. Such relationships are not uncommon. For example the relationship between distance travelled, (d), speed, (s), and time, (t) is given by d = st. This would be a straight-line graph with d plotted on the y-axis and t on the x-axis.

2-18 Fig 4 Graph of y = 2x



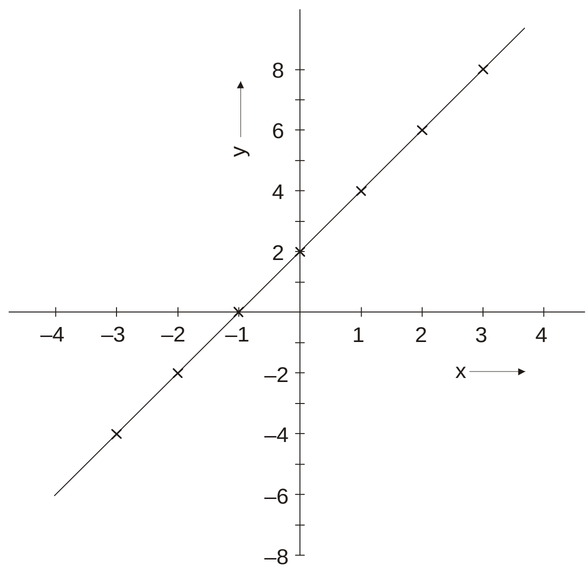
6. It is of course possible for a straight line through the origin to slope down to the right rather than up to the right as in the previous example. In this case positive values of y are generated by negative values of x and the equation becomes: y = – mx

7. Consider now the values of x and y in Table 2, and the associated graph, Fig 5.

Table 2 Values of x and y

x	–3	–2	–1	0	1	2	3
y	–4	–2	0	2	4	6	8

2-18 Fig 5 Graph of y = 2x + 2



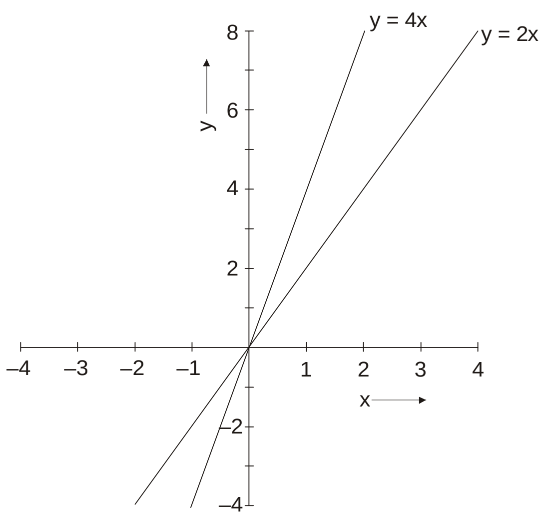
Clearly the graph is closely related to the previous example of y = 2x. In essence the line has been raised up the y-axis parallel to the y = 2x line. Investigation of the table of values will reveal that the relationship between x and y is governed by the equation: $y = 2x + 2$, and in general, a graph of this type has the equation: $y = mx + c$, where c is a constant. It will be apparent that the equation $y = mx$ is identical to the equation $y = mx + c$ if a value of 0 is attributed to the constant c. Thus, $y = mx + c$ is the general equation for a straight line, m and c being constants which can be positive, negative or zero. A zero value of m generates a line parallel to the x-axis. The value of c is given by the point at which the line crosses the y-axis and is known as the intercept.

8. **Gradient**. Consider Fig 6 which shows two straight-line graphs: $y = 2x$ and $y = 4x$. Both lines pass through the origin and the essential difference between them is their relative steepness. The line $y = 4x$ shows y changing faster for any given change in x than is the case for $y = 2x$. The line $y = 4x$ is said to have a steeper gradient than the line $y = 2x$. The gradient is defined as the change in y divided by the corresponding change in x, ie $\frac{y}{x}$. Rearranging the general equation for a straight line ($y = mx$), to make m the subject gives $m = \frac{y}{x}$, i.e. the constant m is the gradient of the straight line. As the line $y = mx + c$ has been shown to be parallel to $y = mx$, this clearly has the same gradient, given by the value of m. In the equation:

$$\text{distance} = \text{speed} \times \text{time}$$

'speed' is equivalent to 'm' in the general equation, and it is apparent that the gradient, speed, represents a rate of change - in this case the rate of change of distance with time. This concept of the gradient representing a rate of change will become important when dealing with calculus in Volume 13, Chapter 14.

2-18 Fig 6 Graphs of y = 2x and y = 4x



Non-Linear Graphs

9. Not all relationships result in straight-line graphs, indeed, they are a minority. A body falling to earth under the influence of gravity alone falls a distance y feet in time t seconds governed by the equation:

$$y = 16t^2$$

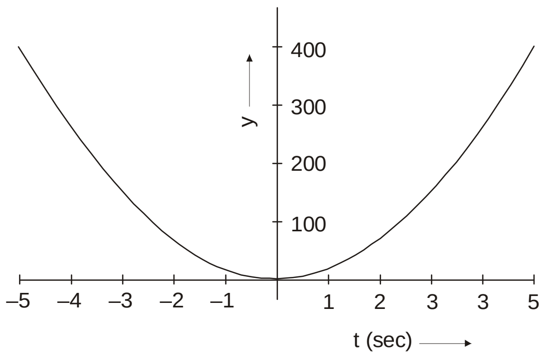
Table 3 shows a range of values of t with the corresponding values of y, and Fig 7 the associated graph.

Table 3 Values of t and y

t	0	1	2	3	4	5
y	0	16	64	144	256	400

Although not relevant in this example, notice that negative values of t produce identical positive values of y to their positive counterparts. The graph is therefore symmetrical about the y-axis and the shape is known as a parabola. The constant in front of the t^2 term determines the steepness of the graph.

2-18 Fig 7 Graph of $y = 16t^2$

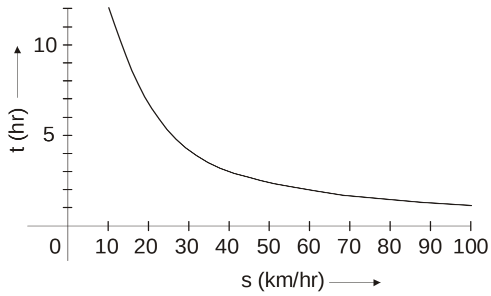


10. Consider now the problem "How long will it take to travel 120 km at various speeds?" This can be expressed as the equation:

$$t = \frac{120}{s}$$

where t = time in hours and s = speed in km/hr. This is an example of inverse proportion, ie an increase in s results in a proportional decrease in t. If values are calculated for s and t, and a graph is plotted, it will have the form illustrated in Fig 8 known as a hyperbola.

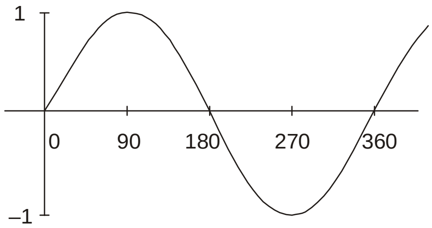
2-18 Fig 8 Graph of $t = \frac{120}{s}$ - Inverse Proportion



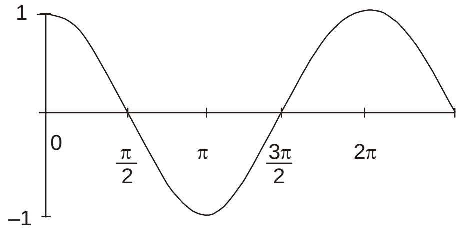
11. Graphs of y = sin x and y = cos x will be encountered frequently. The shapes of the graphs are shown below (Fig 9).

2-18 Fig 9 Graphs of sin and cos

2-18 Fig 9a y = sin x



2-18 Fig 9b y = cos x



The sine graph is shown with the x-axis scaled in degrees while the cosine graph has the x-axis scaled in radians. Either is correct; the radian form is frequently encountered in scientific texts. Sketches of these graphs are useful when trying to determine the value and sign of trigonometric functions of angles outside of the normal 0º to 90º range. Notice that both graphs repeat themselves after 360º (2π radians).

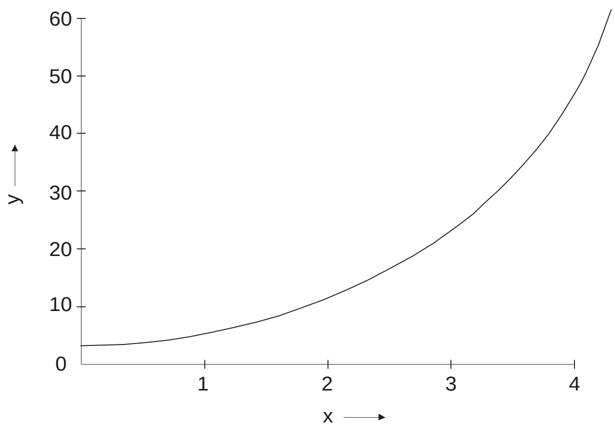
12. Finally, it is worth considering the graph that describes the relationship:

$$y = e^{ax}$$

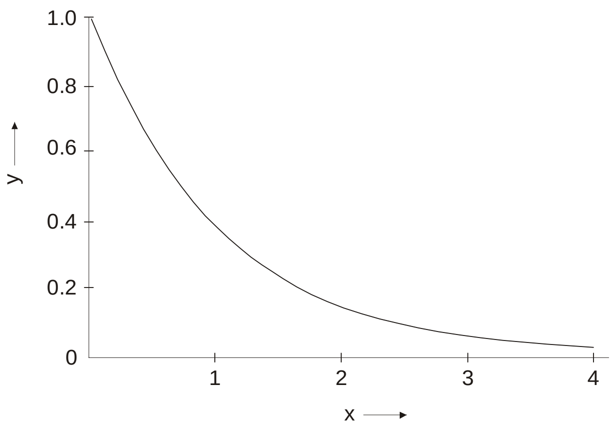
where a is a positive or negative constant. This form of equation is very common in science and mathematics and variants of it can be found in the description of radioactive decay, in compound interest problems, and in the behaviour of capacitors. The irrational number 'e' equates to 2.718 to 4 significant figures. The graph of $y = e^x$ is shown in Fig 10a and that of $y = e^{-x}$ in Fig 10b. The significant point about these graphs, which are known as exponential graphs, is that the rate of increase (or decrease) of y increases (or decreases) depending upon the value of y. A large value of y exhibits a high rate of change. It is also worth noting that there can be found a fixed interval of x over which the value of y doubles (or halves) its original value no matter what initial value of y is chosen. This is the basis of the concept of radioactive decay half-life. The interval is equivalent to $\frac{0.693}{a}$ where a is the constant in the equation $y = e^{ax}$.

2-18 Fig 10 Exponential Graphs

8-7 Fig 10a y = e^{x}



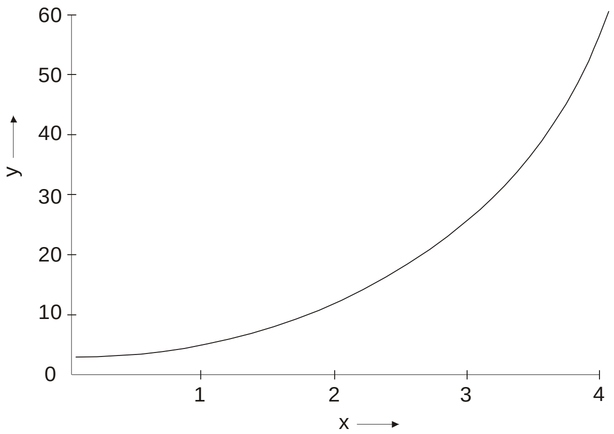
8-7 Fig 10b y = e^{-x}



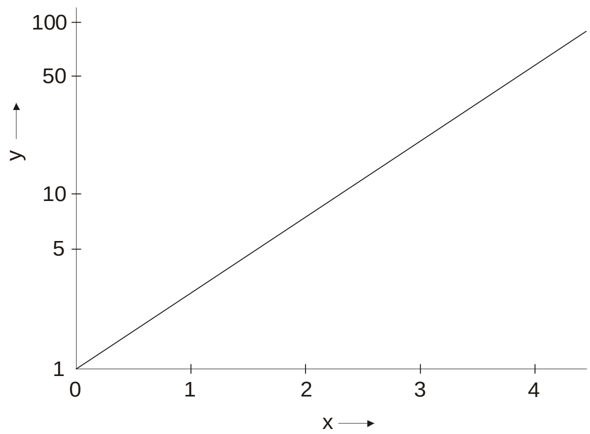
13. **Logarithmic Scales**. Clearly plotting and interpreting from exponential graphs can be difficult. The problem can be eased by plotting on a graph where the x-axis is scaled linearly while the y-axis has a logarithmic scale. This log-linear graph paper reduces the exponential curve to a straight line. A comparison between the linear and log-linear plots of $y = e^x$ is shown in Fig 11.

2-18 Fig 11 Comparison Between Linear and Log-linear Plots

2-18 Fig 11a $y = e^x$ (Linear Scales)



2-18 Fig 11b $y = e^x$ (Log-linear Scales)

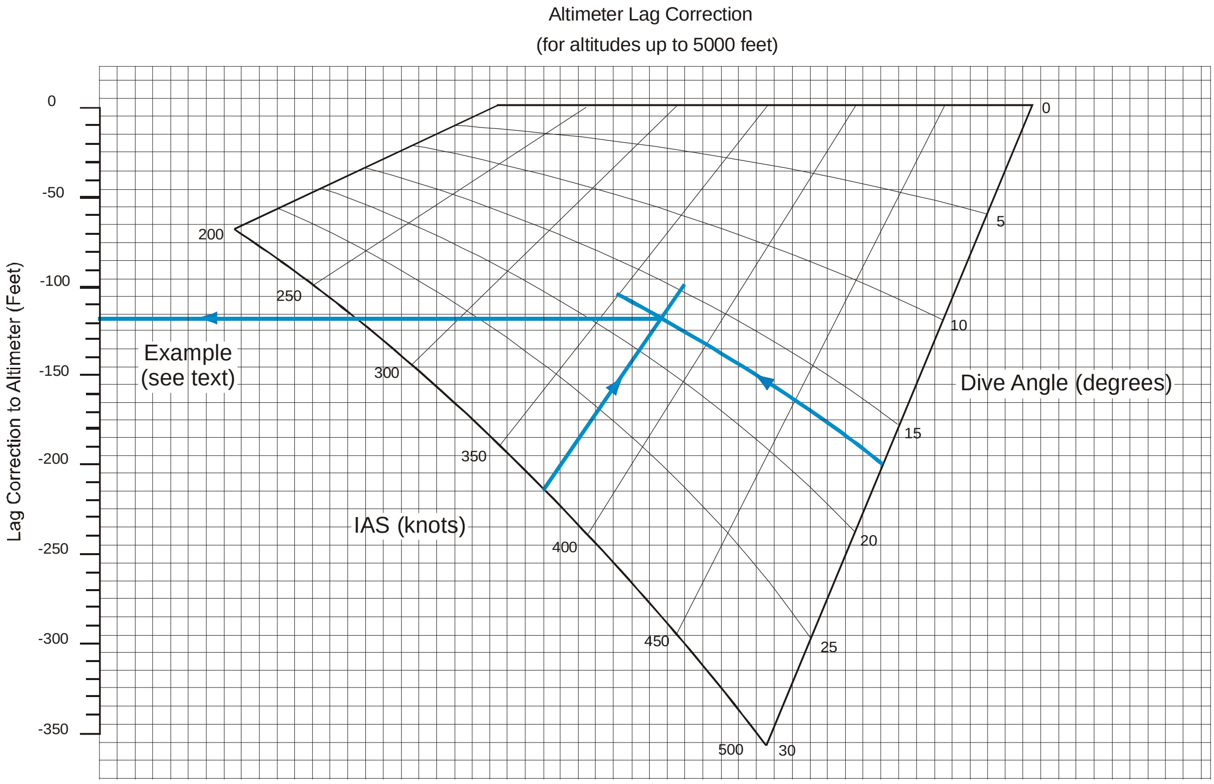


The Presentation and Extraction of Data

14. So far this chapter has been concerned with the mathematical background to simple graphs. More commonly graphs will be encountered and used as a source of data, especially in the field of flight planning and aircraft performance. Whilst occasionally these graphs will be either the simple forms already described or variations on these forms, more often rather complex graphs are used as being the only practical way of displaying complex relations. Two such types of complex graph will be described here in order to establish the method of data extraction. Finally, the nomogram will be discussed.

15. **Carpet Graphs**. An example of a carpet graph is shown in Fig 12.

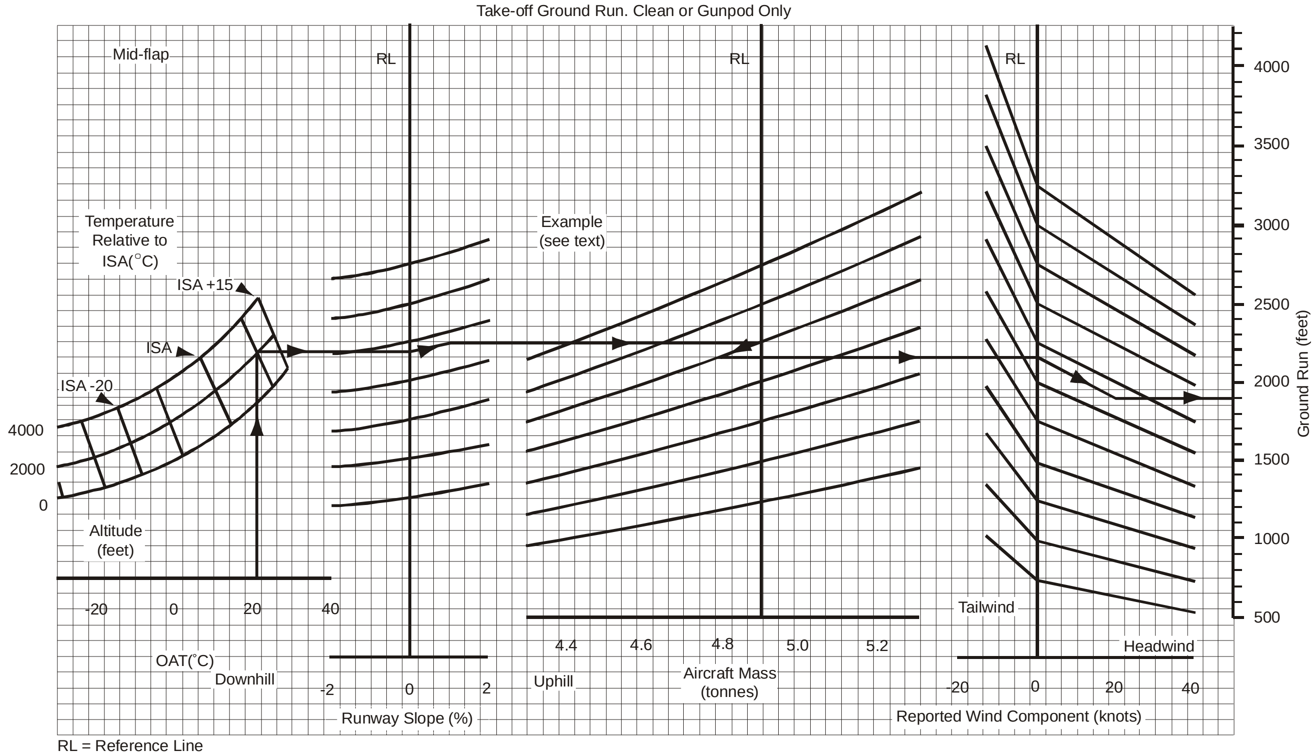
2-18 Fig 12 Carpet Graph



The aim of the graph is to indicate the lag in the altimeter experienced in a dive. Unlike the graphs already discussed where one input, 'x', produced one output, 'y', the carpet graph has two inputs for one output. The output is on the conventional 'y' axis but there is no conventional 'x' axis, rather there are two input axes. On the right hand edge of the 'carpet' diagram are figures for dive angle whilst on the bottom edge are figures for indicated air speed. To use the graph it is necessary to enter with one parameter, say dive angle, and follow the relevant dive angle line into the diagram until it intersects the appropriate IAS line. Intermediate dive angle and IAS values need to be interpolated, thus in the example values of 17º and 375 kt have been entered. From the point of intersection a horizontal line is constructed which will give the required lag correction figure where it intersects the 'y' axis, –118 feet in the example.

16. **Families of Graphs**. It is often necessary to consider a number of independent factors before coming to an end result. In this situation a family of graphs is frequently used to present the required information. Fig 13 shows such a family designed for the calculation of the aircraft's take-off ground run. Apart from the aircraft configuration which is indicated in the graph title, there are five input parameters. There will very often be a series of related graphs with variations in the title, for example in this case there will be another family of graphs for an aircraft with wing stores. It is clearly important that the correct set is selected. The method of using the graph will be described with reference to the example.

2-18 Fig 13 Family of Graphs



17. At the left end is a small carpet graph. Starting with the value of outside air temperature (21º) proceed vertically to intersect the altitude line (2,000 feet). Alternatively enter the 'carpet' at the intersection of the altitude and temperature relative to ISA. From this intersection proceed horizontally into the next graph to intersect the vertical reference line, marked RL. From this point parallel the curves until reaching the point representing the value of runway slope as indicated on the bottom scale (1% uphill). From here construct a horizontal line to the next graph reference line. Repeat the procedure of paralleling the curves for aircraft mass (4.8 tonnes) and then proceed horizontally into the last graph for head/tail wind (20 kt head) which is used in the same manner. Finally the horizontal line is produced to the right hand scale where the figure for ground run can be read (1,900 feet).

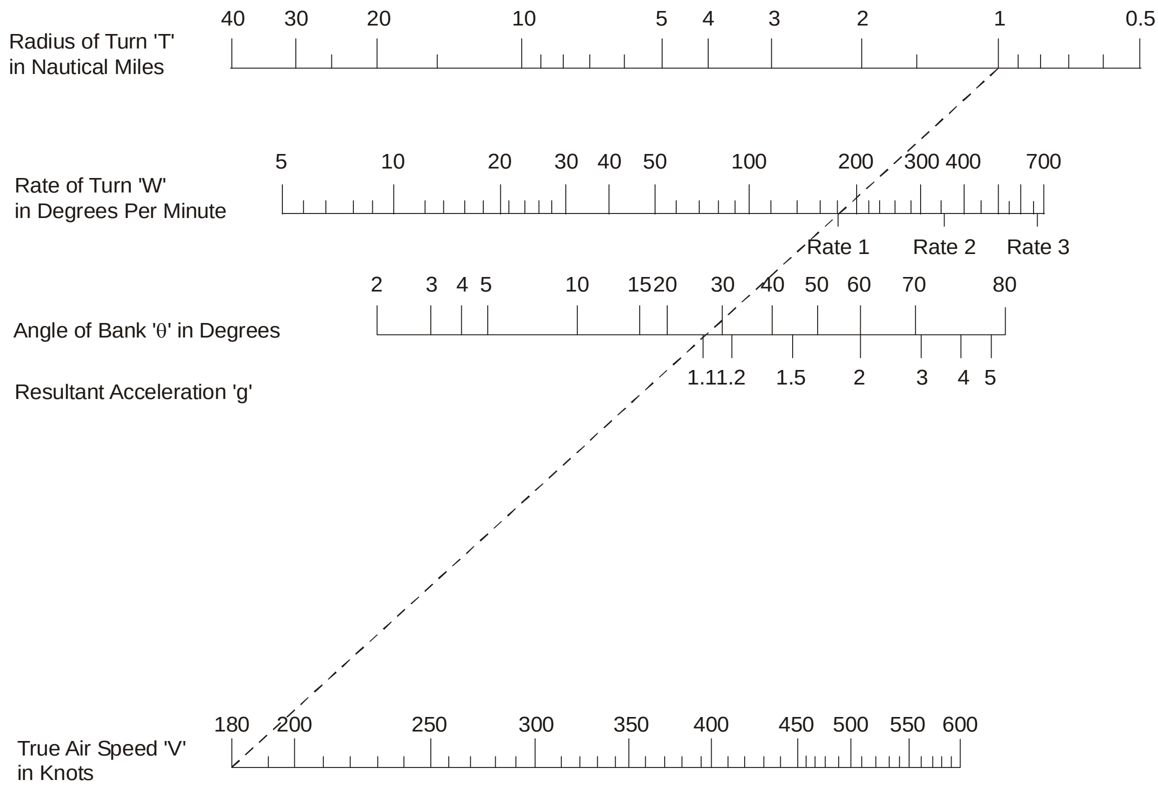
18. **The Nomogram**. The nomogram is not strictly a graph but a diagrammatic way of solving rather complex equations. There are usually two input parameters for which one or two resultant outputs may be derived. Fig 14 shows a nomogram for the determination of aircraft turning performance. The equations involved are:

$$\text{Rate of turn TAS} = \frac{\text{TAS}}{\text{Radius of Turn}}$$

$$= \text{Acceleration} \times \tan \text{Bank Angle}$$

This nomogram consists of four parallel scaled lines. Two known values are joined by a straight line, and the intersection of this line with the other scales gives the unknown values. In the example illustrated, an input TAS of 180 kts with a Rate 1 turn gives a resultant of 1.1 g, and a radius of turn of 1nm.

2-18 Fig 14 A Nomogram



The variables are related by the equation $V_W = \frac{V}{T} = g \tan \theta$

To use the Nomogram join two known values by a straight line and the intersection of this line on its projection with the other scales give the unknown values

Example : TAS (V) = 180 Kt
Rate of Turn (W) = 1
Angle of Bank (θ) = 25°
Using the Nomogram (see dotted line)
Resultant Acceleration (g) = 1.1
Radius of Turn (T) = 1 nm

CHAPTER 19 - REDUCED THRUST TAKE-OFF

Introduction

1. In recent years, a reduced thrust take-off procedure has been developed in order to improve engine reliability and to conserve engine life. Clearly, it can be used only when full take-off thrust is not required to meet the various performance requirements on the take-off and initial climb out. Those aeroplanes approved for this procedure have Operating Data Manual (ODM) appendices containing all the information necessary. This procedure is known by differing names e.g. Graduated Power Take-off (British Airways), Factored Take-off Thrust (Royal Air Force), Variable Take-off Thrust and many others.

2. A take-off can be limited by many considerations. The three which are significant in terms of reduced thrust are:

a. Take-off field lengths.

b. Take-off WAT limitation (engine-out climb gradients).

c. Net take-off flight path (engine-out obstacle clearance).

3. Where the proposed take-off weight (TOW) is such that none of these considerations is limiting, then the take-off thrust may be reduced, within reason, until one of the conditions becomes limiting.

4. The following general principles apply:

a. Whilst the procedure is optional and is generally used at the Captain’s discretion, it is not unreasonable for an operator to require it to be used when there is no resultant loss of safety.

b. Under Federal Aviation Regulations (USA national regulations) the thrust reduction must not exceed 10%, whilst the CAA require that a minimum thrust, easily identified by the pilot, will be limiting.

c. The reduced thrust procedure should not be used in either of the following conditions:

(1) Contaminated runway surface.

(2) When specified in the relevant ODM or Aircrew Manual.

d. The overall procedure includes a method for periodically checking that the engines are capable of producing full take-off thrust.

Method

5. There are several possible methods that can be used, but the most usual is the 'assumed temperature' method. A temperature higher than the actual ambient temperature is determined, at which the actual TOW would be the Regulated Take-off Weight (RTOW), all other parameters having their actual values. The take-off thrust appropriate to this higher temperature is then used for the take-off. This method ensures that all the take-off performance requirements are met at the reduced thrust, with a small additional margin because the density effect of the actual ambient temperature is lower than that of the assumed temperature. While this means that, in the event of a continued take-off following an engine failure, the whole take-off would be good in terms of performance, it is nevertheless recommended that full take-off power be restored in the event of an engine failure above V_1.

6. There are some particular variations applying to aeroplanes with contingency power ratings; details are given in ODMs. All ODM appendices on reduced thrust contain a worked example. Finally, remember that the performance consideration which is limiting at full thrust (field length, WAT curve, net flight path) is not necessarily the limiting one at reduced thrust.

Safety Considerations

7. Although by using reduced thrust take-offs the aeroplane is now being operated at or near a performance limited condition, more often, there is no increase in risk because:

a. The 'assumed temperature' method of reducing thrust to suit the take-off weight does so at a constant thrust/weight ratio; the actual take-off distance, take-off run and accelerate stop distances at reduced thrust are less than at full thrust and full weight by approximately 1% for every 3 °C that the actual ambient temperature is below the assumed temperature.

b. The accelerate stop distance is further improved by the increased effectiveness of full reverse thrust at the lower temperature.

c. The continued take-off after engine failure is protected by the ability to restore full power on the operative engines.

d. A significant percentage of take-offs are at weights close enough to the RTOW not to warrant the use of reduced thrust.

e. The excess margins on lighter weight take-offs are largely preserved by the maximum thrust reduction rule (or minimum thrust requirement).

CHAPTER 20 – ENGINE-OUT FERRYING

Introduction

1. Engine-out ferrying is the process of flying a multi-engine aeroplane from one place to another with one of its engines inoperative. It is a convenience to an operator, in that it avoids the necessity of holding a costly spare engine at all stations on the operator's route network. It is sometimes called '3-engine ferry', which is not inaccurate for a 4-engine aeroplane. Three-engine aeroplanes can also be ferried with one engine out so long as they meet the rules; some have demonstrated this. The safety level on an engine-out ferry is, of course, lower than on all-engines operation, hence passengers may not be carried. Depending on the performance and handling qualities of the type, and its certification standard, the actual risk can vary from very little more than that associated with a normal take-off to considerably more than that with a normal take-off.

2. Under British Civil Airworthiness Requirements (BCARs), the ferrying take-off procedure gives full protection against failure of a further engine from a controllability point of view and limits the risk of inadequate performance to a period covering the maximum stop speed to the speed at the 'gear locked up' point. A failure in this period would result in an overrun during an abandoned take-off and a very low screen height during a continued take-off. Those aeroplanes approved for this procedure have Operating Data Manual (ODM) appendices containing all the information necessary.

Performance Operating Limitations

3. Ferrying is a specialized operation of an aeroplane certificated in Group A Transport Category in which an acceptable level of safety is achieved by setting against power-unit unserviceability, operational restrictions as to use, weight and flight technique.

4. **Weather Conditions**. The visibility and cloud ceiling prevailing at the aerodrome of departure and forecast for the destination and alternate aerodromes shall be not less than 1nm and 1,000 ft respectively (BCARs).

5. **Route Limitation**. No place along the intended track shall be more than 2 hours flying time in still air at the ferrying cruising speed from an aerodrome at which the aeroplane can comply with Ferry Landing Distance Required relating to an alternate aerodrome.

6. **Weight Restrictions**. It is always necessary for the take-off weight to be restricted, for obvious reasons. The restriction is not great on a high performance aeroplane but is very much more on a low performance 4-engine aeroplane, and can be crippling on a 3-engine aeroplane. If there is any doubt, the solution is to take-off light and fly shorter legs, within reason.

7. **Ferrying TORR**. The Ferrying TORR shall be the gross distance to accelerate from 'brakes off', to the Ferrying Rotation Speed, and thereafter to effect the transition to climbing flight and attain a screen height of 35 ft at a speed not less than Ferrying Take-off Safety Speed.

8. **Ferrying TODR**. The Ferrying TODR shall be the gross distance at para 7, multiplied by 1.18. The Ferrying TODR shall not exceed the accelerate stop distance available at the aerodrome at which the take-off is to be made.

9. **Ferrying Take-off Net Flight Path.** The Ferrying Take-off Net Flight Path, which extends from a height of 35 ft to a height of 1,500 ft, shall be the gross flight path with one serviceable power unit inoperative reduced by a gradient of 0.5% and plotted from a point 35 ft above the end of the Ferrying TODR to a point at which a height of 1,500 ft above the aerodrome is reached. It shall clear vertically by 35 ft all obstacles lying within 200 ft/60 m plus half the aeroplane wing-span either side of the intended track. In assessing the ability of the aeroplane to satisfy this condition, it shall be assumed that changes of direction shall not exceed 15º.

10. **En Route Terrain Clearance**. The ability of the aeroplane to comply with the requirements shall be assessed on the one serviceable power unit inoperative net data.

11. **Ferrying Landing Distances Required**. The Ferrying Landing Distance Required is calculated in the following manner:

a. **Destination Aerodromes.** The Ferrying Landing Distance Required shall be the normal landing distance required, multiplied by 1.05.

b. **Alternate Aerodromes**. The Ferrying Landing Distance Required appropriate to alternate aerodromes shall be the Ferrying Landing Distance appropriate to destination aerodromes (determined in accordance with para 11a), multiplied by 0.95.

12. **Take-off Technique**. The take-off technique is established and published in the ODM. This will include details of crosswind limitations, runway surface conditions and the establishment of a V_I speed (V_I as used in this context is a convenient shorthand expression for a decision speed in relation to engine ferry - it does not imply performance protection in its usual context in relation to normal all-engines operation).

CHAPTER 21- LOAD BEARING STRENGTH OF AIRFIELD PAVEMENTS

Introduction

1. An aerodrome pavement should be strong enough for an aeroplane to be operated without risk of damage to either the pavement or the aeroplane and, in normal circumstances, the pavement should last long periods without requiring major maintenance.

2. It is necessary to classify both pavements and aeroplanes in such a way that the load bearing capacity of the pavement can readily be compared with the load exerted by the aeroplane.

Pavements

3. **Pavement Types**. There are two main types of aerodrome pavement:

a. **Rigid Pavement**. Rigid pavement is the term used when the bearing strength is derived from concrete slabs.

b. **Flexible Pavement**. Flexible pavement consists of a series of layers of compacted substance; the surface top layer is usually of bituminous material.

4. **Stress Effects***.* The stress effect on a pavement, caused by an aircraft, will vary with the all-up weight (AUW), tyre pressure (and thus contact area), number and spacing of wheels, and the type and thickness of the pavement. Aircraft with a multi-wheel arrangement are able to spread their weight better than those with a single wheel arrangement.

5. **Runway Surfaces***.* The surface of a runway should not have irregularities that might affect aircraft steering or otherwise adversely affect the take-off or landing of an aircraft. The surface should provide good braking action, and the coefficient of friction in both wet and dry conditions must reach a satisfactory standard.

6. **Aircraft Compatibility**. During flight planning, the officer authorizing the flight, and the aircraft captain, should consider compatibility of aircraft type with pavement strengths for aerodromes of intended operation. Aerodrome pavement strengths are published in the Aeronautical Information Publication (AIP), Military AIP, and En-Route Supplements (ERS). The main methods of comparing aircraft loading against pavement strength are explained within this chapter, listed in mainly chronological order, to assist explanation of their development.

The Load Classification Number (LCN) System

7. The Load Classification Number (LCN) system is an early system, based on aircraft having wheels in single units with a minimum tyre contact area of 200 in^2. The derivation of the LCN number is from the ratio between two values. The first value (the 'standard value') is the load required to produce a failure of a given surface when acting over an area of 530 in^2. The second value is the load required to produce a failure on the same surface, but applied over a lesser specified area. The ratio between these two values is expressed as a percentage and is known as the LCN. By comparing the wheel loading of an aeroplane with the LCN of an aerodrome pavement, it is possible to determine whether the pavement is sufficiently strong for that particular aeroplane. The load exerted by the aeroplane depends on:

a. Aeroplane AUW.

b. Tyre pressure (and thus contact area).

8. **Application of the LCN System**. The aircraft LCN must not exceed the aerodrome LCN if the number of aircraft movements is to be unrestricted.

9. **Aircraft with Multiple Wheel Units.** The introduction of increasingly heavier aeroplanes with their associated multiple wheel units and higher tyre pressures, i.e. smaller contact areas, complicated the original calculations. In order to obtain a simple figure on which comparisons could be made, the concept of Equivalent Single Wheel Loading was introduced.

Equivalent Single Wheel Loading (ESWL)

10. In order for aircraft with multi-wheel undercarriages and higher tyre pressures to be classed within the LCN system, the concept of Equivalent Single Wheel Loading (ESWL) was introduced. The ESWL of a particular group of two or more closely spaced wheels is calculated, at the tyre pressure of the assembly, enabling it to be compared with a single wheel unit. Thus, a direct, though complex relationship exists between ESWL and LCN, and conversion can be achieved by means of an appropriate table or graph.

The Load Classification Group (LCG) System

11. The Load Classification Group (LCG) system is a development from, and improvement on, the earlier LCN system. It is intended to take account of local LCN variations by placing pavement load bearing capacity into groups, each of which embrace a range of LCN values as shown in Fig 1. For example, it can be seen that a pavement with an LCN of 70 would belong within LCG III.

2-21 Fig 1 Correlation of LCN and LCG

LCN	LCG
101 upwards	I (Heavy ac)
76 - 100	II
51 - 75	III
31 - 50	IV
16 - 30	V
11 - 15	VI
01 - 10	VII (Light ac)

12. **The Pavement Load Classification Group**. In this system, the bearing strength of a pavement, derived from characteristics and type of construction, is indicated by placing it in a Load Classification Group (LCG).

13. **Aircraft Groupings.** Aircraft are also given an LCG value, which is promulgated in ODMs or Aircrew Manuals. If the aircraft weight is published as an LCN, then appropriate pavement LCG requirement can be read from Fig 1.

14. **Application of the LCG System.** Provided that the LCG of the aircraft corresponds to that of the pavement, unlimited use of the pavement area is permitted, i.e. an aeroplane whose LCG places it in Group IV may be operated continuously on pavements of LCGs I, II, III or IV.

Example: A fully-loaded Tucano T Mk 1 aircraft falls within LCG VII. The ERS entry for Cranwell Rwy 09/27 (Fig 2) shows that the pavement for runway 09/27 is LCG IV. The Tucano T Mk 1 aircraft can therefore operate without restriction from that runway.

2-21 Fig 2 ERS Extract – Cranwell

CRANWELL England RAF N53 01.82 W000 28.99 Elev 218ft EGYD / **---**
TC, UK(L)1/2/5, UK(H)2/6, EU(H)SP1/2/1-OAT Barnsley ASRgn London FIR

TEL **PSTN** (01400) 261201 Ext 7377 **DFTS** 95751
TIME PPR H24. 0815-1730(A) Mon-Thu, 0815-1700(A) Fri, or as required by 3FTS.
0845-1700(A) Sat-Sun, or as required by EMUAS or 7 AEF only. ATZ opr hr H24.
RWY **09/27** (082.69ºT/-0.65%) 6,831ft/2,082m, Asphalt/Concrete, LCG IV; 4, 6, 7, 12, 13, 15**(a)**

15. **Aerodrome LCG Overload**. Because of the safety factor incorporated, the published LCG for the pavement always represents a safe loading less than the maximum that the pavement can carry. It is therefore possible to allow occasional/infrequent use of a pavement by an aeroplane with an LCG one category heavier than the published pavement LCG, e.g. an aeroplane whose LCG places it in Group IV may be operated occasionally on pavements of LCG V. It can be seen from Fig 3 that at all weights, apart from the overload case, the C130 can operate unrestricted from RW 09/27 at Cranwell. At overload weight, with an LCG of III, the C130 can operate from the runway occasionally but such operations are at the discretion of the aerodrome operator. An aircraft with an LCG category two or more heavier than the published aerodrome LCG will probably only be allowed to operate at the aerodrome in an emergency.

2-21 Fig 3 LCN/LCG – Hercules C130 Mk 3 & C130 Mk 4

	ESWL		*LCN*	*LCG*
Operating Weight Empty	*lbf*	*kgf*		
	23,237	10,540	23	**V**

	ESWL		*LCN*	*LCG*
Max Landing Weight	*lbf*	*kgf*		
	39,278	17,817	42	**IV**

	ESWL		*LCN*	*LCG*
Max Take-off Weight	*lbf*	*kgf*		
	47,374	21,489	50	**IV**

	ESWL		*LCN*	*LCG*
Overload	*lbf*	*kgf*		
	51,511	23,365	55	**III**

16. **Recent LCG Values.** Instances may occur where the latest LCG value for a pavement does not agree with the last published LCN. This is because the calculations of LCG are, in many cases, based on a new evaluation of pavement strength.

The Aircraft Classification Number (ACN)/Pavement Classification Number (PCN) System

17. **ICAO Requirements.** ICAO Standard Practices now recommend that Member States adopt uniform and standardized systems for reporting of pavement strengths. For pavements intended for use by aeroplanes having a maximum total weight authorized (MTWA) in excess of 5,700 kg (12,500

lbs), ICAO recommends the use of the Aircraft Classification Number (ACN)/Pavement Classification Number (PCN) system. (For aircraft with MTWA of 5,700 kg or less, see para 22a.)

18. **The ACN.** The ACN is a number expressing the relative effect of an aircraft mass on a pavement for a specified sub-grade strength. It is calculated by taking into account the weight of the aircraft, the pavement type and the sub-grade category. ACN values for aircraft (one for rigid pavements and one for flexible pavements) are promulgated in relevant Aircrew Manuals. The ACN values for British military aircraft are also published in the Flight Information Handbook. The ACN values for the Hercules C130 are shown in Fig 4 as an example.

2-21 Fig 4 ACN – Hercules C130 Mk 4 & C130 Mk 5

		Aircraft Classification Number (ACN)							
		Rigid Pavement Subgrades				Flexible Pavement Subgrades			
Load Condition	Tyre PSI	High A	Medium B	Low C	Ultra Low D	High A	Medium B	Low C	Ultra Low D
155007 lbs (1)	105	27	28	31	33	24	27	29	33
81320 lbs (2)	105	11	12	13	13	9	11	11	14
Overload 174165 lbs	114	30	31	34	37	26	30	32	37

(1) Maximum Total Weight Authorised (MTWA)
(2) Operating Weight Empty (OWE)
If the aircraft is operating at an intermediate weight at take-off or landing, the ACN value can be calculated by a linear variation between the tabled weights.

19. **The PCN.** The PCN expresses the bearing strength of pavement, for unrestricted operations. It is reported as a five part code, promulgated in a set order, e.g. PCN 23/F/B/X/U. The five parts of the code are:

a. **The Numerical Value**. The numerical value represents the bearing strength of the pavement for an unrestricted number of movements. The numerical value is against a continuous scale, commencing at zero, with no upper limit.

b. **Pavement Type.** The reporting procedure for pavement type divides surfaces into rigid (R) or flexible (F).

c. **Sub-grade Strength Category.** The strength of the pavement sub-grade is measured and classified appropriate to the pavement type, as either high (A), medium (B), low (C) or ultra-low (D).

d. **Tyre Pressure Category.** Tyre pressures are divided into four groups - high (W) (no upper limit), medium X) (max 217 psi), low (Y) (max 145 psi), very low (Z) (max 73 psi). Thus, provision is made for the aerodrome authority to place a limit on the maximum allowable tyre pressure, if this is a particular constraint.

e. **Pavement Evaluation Method.** The pavement qualities may be determined by either a full technical evaluation (T) or, alternatively, by experience gathered from previous aircraft operations (U). Where the latter is used, the pavement must be monitored for performance in case it does not meet expectations and begins to deteriorate.

Using the decode at Fig 5, it can be seen that the previous example, PCN 23/F/B/X/U, describes a pavement of PCN 23; it is a flexible pavement, of medium sub-grade category, the maximum tyre pressure is Medium (max 217 psi), and the evaluation was carried out by experience of aircraft operations.

2-21 Fig 5 PCN Decode

Part	*Code*	*Details*
PCN	-	PCN Number
Pavement Type	R F	Rigid Flexible
Sub-grade Category	A B C D	High Medium Low Ultra-low
Tyre Pressure	W X Y Z	High (no limit) Medium (max 217 psi) Low (max 145 psi) Very low (max 73 psi)
Evaluation Method	T U	Technical By experience

20. **Using the ACN/PCN System.** Provided that the pavement PCN is equal to, or greater than, the aircraft ACN, unlimited use of the pavement is permitted.

Example: A Hercules C130 aircraft is due to land at Newcastle, refuel to max AUW, and take-off for the next sortie. Is there a performance limitation due to runway pavement strength? The PCN for Runway 07/25 at Newcastle is promulgated as PCN 65/F/B/W/T (Fig 5). This shows that the runway has a PCN of 65, the pavement is flexible, of medium sub-grade category, and there is no tyre pressure limit. These values were ascertained by technical evaluation.

2-21 Fig 6 ERS Extract - Newcastle Rwy 07/25

NEWCASTLE England Civ N55 02.25 W001 41.50 Elev 266ft EGNT / NCL
TC, UK(L)2/5, UK(H)2/6, EU(H)12/SP1/1-OAT Tyne ASRgn Scottish FIR

TEL (0191) 286 0966 (Switchboard), x3244 (ATC Supervisor)
TIME H24. ATZ ***ff***.
RWY **07/25** (065ºT/-0.35%) 7,641ft/2,329m, Grooved Asphalt, PCN 65/F/B/W/T; L6, 7, 11, 12, 13, 15**(a)**

Fig 4 shows that the C130's ACN for flexible pavement of medium sub-grade is 28 at MTWA, 12 at OWE and 31 at overload weight. The C130's tyre pressure (105/114 psi) is not relevant in this case, as there is no limit. The PCN at Newcastle (65) is greater than the C130's ACN at all operating weights and thus unlimited use of the runway pavement is permitted.

21. **Aerodrome PCN Overload.** Aerodrome authorities decide their criteria for permitting overload operations as long as pavements remain safe for use by aircraft. When the ACN exceeds the PCN by 50%, overload operations will probably only be allowed in an emergency.

Other Load Classification Systems

22. In addition to pavement classification systems described previously, the following methods of aerodrome surface classification may be encountered:

a. **ICAO Recommended - Aircraft less than 5,700 kg.** ICAO recommends that the bearing strength of a pavement intended for aircraft below 5,700 kg (12,500 lbs) MTWA, shall be reported by a simple statement, in words, of the maximum allowable aircraft weight and the maximum allowable tyre pressure. For example, a pavement strength may be published as:

"500 kg/0.75 MPa". (Note: 1 megapascal (MPa) = 145 psi).

b. **Acceptable Aeroplanes**. In this very basic system, a particular aeroplane e.g. A300, may be quoted as the limiting size of aeroplane which can be accepted on a pavement. This system takes no account of aeroplane wheel configuration or tyre pressures and therefore does not represent an accurate assessment of the actual pavement load bearing capability.

c. **Acceptable All-up Weight (AUW).** This system imposes an AUW limitation for aeroplanes using the pavement; it may be further qualified by reference to the undercarriage configuration e.g. "200,000 lb on a dual tandem undercarriage". Compatibility with other weights and configurations can be determined by comparing the aeroplane's LCN with the LCN of any other aeroplane which has a weight/undercarriage configuration similar to that quoted for the pavement.

Taxiways/Parking Areas

23. The airfield information published in No 1 AIDU En-Route Supplements gives strengths of runways only. The strengths of surfaces on taxiways and parking areas (which may differ from that of the runway) are promulgated in the Military AIP and the UK AIP.

Example: Figs 7 and 8 are extracted from the UK Mil AIP and show the apron/taxiway and runway data for RAFC Cranwell (EGYD).

2-21 Fig 7 Apron and Taxiway Data for RAFC Cranwell (EGYD)

<table>
<tr><td rowspan="6">1</td><td rowspan="6">Apron surfaces:</td><td colspan="2">Apron</td><td colspan="2">Surface</td><td colspan="2">Strength</td></tr>
<tr><td colspan="2">Hangars 265, 266, 534</td><td colspan="2">Concrete</td><td colspan="2">LCG IV</td></tr>
<tr><td colspan="2">Hangars 29, 30</td><td colspan="2">Concrete</td><td colspan="2">LCG IV</td></tr>
<tr><td colspan="2">East Apron to Hangar 30</td><td colspan="2">Asphalt</td><td colspan="2">LCG IV</td></tr>
<tr><td colspan="2">Rubb Hangar</td><td colspan="2">Concrete Block</td><td colspan="2">LCG VII</td></tr>
<tr></tr>
<tr><td rowspan="3">2</td><td rowspan="3">Taxiway width, surface and strength:</td><td>Taxiway</td><td colspan="2">Width</td><td colspan="2">Surface</td><td>Strength</td></tr>
<tr><td>Perimeter</td><td colspan="2">15m</td><td colspan="2">Asphalt/Concrete</td><td>LCG IV</td></tr>
<tr><td>Link to 08-26</td><td colspan="2">15m</td><td colspan="2">Asphalt</td><td>LCG IV</td></tr>
</table>

2-21 Fig 8 Runway Data for RAFC Cranwell (EGYD)

Designations Runway number	True and MAG bearing	Dimensions of Runway (m)	Strength (PCN) and surface of Runway and stopway	Threshold co-ordinates	Threshold elevation highest elevation of TDZ of precision APP Rwy
1	2	3	4	5	6
08	082·69° GEO 084·16° MAG	2081 x 45	LCG III Asphalt/Concrete	N53 01 43·43 W000 30 20·57	217·5ft TDZE 217·8ft
26	262·71° GEO 264·18° MAG	2081 x 45	LCG III Asphalt/Concrete	N53 01 50·93 W000 28 43·44	180·3ft TDZE 199·0ft
01	008·20° GEO 009·67° MAG	1461 x 45	LCG IV Asphalt/Concrete	N53 01 15·79 W000 29 07·53	176·6ft TDZE 186·8ft
19	188·20° GEO 189·67° MAG	1461 x 45	LCG IV Asphalt/Concrete	N53 02 02·60 W000 28 56·35	182·4ft TDZE 187·2ft

CHAPTER 22 – FIXED WING AIRCRAFT – WEIGHT AND BALANCE

Introduction

1. The principles of weight and balance are applicable to all aircraft. Aircraft carrying standard loads, e.g. training and air defence aircraft, are normally flown under a pre-computed weight and balance clearance, but transport aircraft, due to their variable load capacity, require a separate clearance for each sortie.

2. The basic weight and centre of gravity moment for each individual aircraft are recorded in the Form 700. The essential weight and balance data for particular types or marks of aircraft is contained in:

a. For transport aircraft - the Weight and Balance Data Book applicable to that aircraft type.

b. For all other types of fixed wing aircraft - the Aircraft Maintenance Manual.

For transport aircraft that may be used in the air transport support and air mobility roles, the weight and balance information required for preparing or loading the aircraft is repeated in the appropriate book in the AP 101B series which deals with air transport support and air mobility. If the information contained in the documents detailed in sub-paras a and b contradicts that given in the AP 101B series, then the Weight and Balance Data Book or the Aircraft Maintenance Manual is to be taken as the final authority. Additional information concerning aircraft loads may be found in AP 101A - 1101-1, Air Transport Operations Manual - Fixed-Wing Aircraft - General and Technical Information.

WEIGHT AND BALANCE

Aircraft Weight Limitations

3. A limitation is imposed on the all-up weight (AUW) at which any aircraft is permitted to operate. This limitation depends on the strength of the structural components of the aircraft and the operational requirements it is designed to meet. If these limitations are exceeded, the safety of the aircraft may be jeopardized and its operational efficiency impaired. Subject to certain flying restrictions, permission may be given to operate at an AUW in excess of the normal maximum. However, as such operation reduces the safety factor, thereby increasing the risk of structural failure in manoeuvre or when flying in turbulent conditions, permission is granted only in rare circumstances.

Effect of Increasing All-up Weight on Aircraft Performance

4. The effect on an aircraft's performance due to increasing AUW is to:

a. Increase the stalling speed, thereby increasing the take-off and landing runs.

b. Increase the aircraft's inertia, thereby reducing acceleration on take-off and deceleration on landing.

c. Reduce the rate of climb.

d. Lower the absolute ceiling and optimum range altitude.

e. Reduce the range and endurance.

f. Reduce manoeuvrability and asymmetric performance.

g. Increase wear on tyres and brakes.

Balance

5. Whilst it is important to ensure that the normal maximum AUW of an aircraft is not exceeded, the distribution of that weight, i.e. the balance of the aircraft, is equally important.

6. It is not possible to design an aircraft in which the lift, weight, thrust and drag forces are always in equilibrium during straight and level flight; the centre of pressure (CP) and the drag line vary with changes of angle of attack and the position of the centre of gravity (CG) depends on the load distribution. It is necessary, therefore, to provide a force to counteract the pitching moments that may be set up by these forces. This is the function of the tailplane, which, together with the elevators and trimmers, can offset any moment set up by the movement of the CP or the drag line. It is also able to counteract any of the unbalance or unstable tendencies caused by movements of the CG, provided that these are confined within the normal limits.

Effects of Unbalanced Loading

7. Incorrect loading of an aircraft can move the CG towards the normal fore-and-aft limits. This can have the following effects on the aircraft's performance:

a. **CG Too Far Forward.**

(1) The aircraft will become difficult to manoeuvre and heavy to handle, requiring a larger stick force than normal.

(2) Elevator authority may be insufficient for the round-out.

(3) There will be an increase in drag (and a consequent decrease in range and endurance) due to the increased nose-up trim required to maintain straight and level flight.

b. **CG Too Far Aft**.

(1) The aircraft becomes less stable (and may become unstable - leading to possible loss of control).

(2) The increased load on the tailplane may cause flutter in some aircraft.

(3) There will be an increase in drag (and a consequent decrease in range and endurance) due to the increased nose-down trim required to maintain straight and level flight.

Changes in Weight and Balance During Flight

8 The AUW of an aircraft is constantly changing during flight due to the consumption of fuel and oil, the release of ammunition and pyrotechnics, the release of missiles or bombs and the dropping of supplies or parachutists. All of these events will cause a reduction in the AUW. Conversely, if in-flight refuelling takes place, then the AUW will be increased.

9. Every alteration of the AUW will cause a movement of the aircraft's CG unless the CG of the item causing the change in weight is coincident with the CG of the loaded aircraft. The amount of the CG movement caused depends on the weight change of the item and its horizontal distance from the CG of the loaded aircraft. Any movement of personnel in the aircraft during flight will also cause a movement of the CG although this will not affect the AUW.

CENTRE OF GRAVITY

Centre of Gravity Limits

10. The optimum position for the CG of a loaded aircraft will change with the AUW. Even if the CG of the loaded aircraft is at its optimum position at take-off, in-flight changes in AUW cause the CG to move. Such movement may cause a progressive loss of efficiency, leading to a state of serious, and even dangerous, unbalance as the distance from the optimum position increases. Normally, only the fore-and-aft position of the CG is important on a conventional aircraft. If the lateral or vertical position is likely to have any material effect, then limits will also be specified for these positions.

11. The limits of permissible movement are always related, either directly or indirectly, to the centre of gravity datum point. In some cases, a second, or alternative, reference point is used. This is known as the 'weighing reference point' and is marked at a convenient position on the aircraft structure at a known and stated distance from the CG datum point. Measurements taken from this point can easily be related to the CG datum point.

12. The 'trim datum' is a point at a known distance from the CG datum. It applies to aircraft on which a trim sheet, or other weight and balance calculating device, is used.

Determination of Centre of Gravity

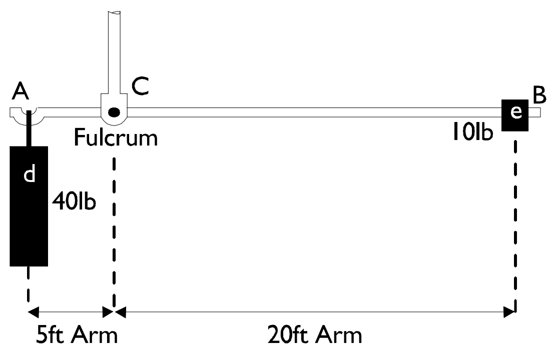
13. It is the aircraft captain's responsibility to ensure that the aircraft is loaded such that its CG lies within the authorized limits and will not move outside these limits during flight. To be capable of carrying out this responsibility the aircraft captain must be conversant with the method used for calculating the CG.

14. **The Principle of Arms and Moments**. The turning effect (moment) of any weight about a point of balance (the fulcrum) is directly proportional to its distance from that point (the arm). The moment of a large weight near the point of balance can, therefore, be equalled by that of a small weight at a greater distance from the fulcrum. From this, the following formula is derived:

$$\text{weight} \times \text{arm} = \text{moment}$$

15. In Fig 1, AB is a lever balanced about its fulcrum (C). If a 40 lb weight (d) is suspended from a point A, 5 ft from C, and a 10 lb weight (e) from a point B, 20 ft on the other side of C, the lever will remain balanced. This is because the positive moment 10 × 20 lb ft tending to turn the lever in a clockwise direction about point C is equalled by the negative moment 40 × 5 lb ft tending to turn the lever in an anti-clockwise direction about point C.

2-22 Fig 1 Principle of Arms and Moments



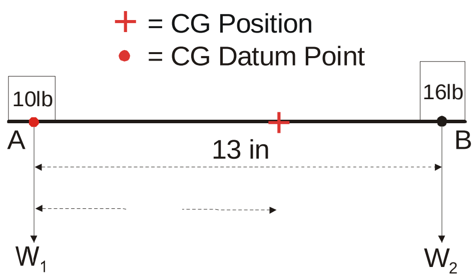
16. **Practical Application of the Principle**. In practice, the formula weight × arm = moment is used to determine the position of the CG of any aircraft by using the weights and arms of the various loads

in the aircraft. By multiplying these weights by their respective arms, their moments about the selected points are found. Since weight × arm = moment, it follows that arm = $\frac{\text{moment}}{\text{weight}}$. Therefore, the sum of the moments divided by the sum of the weights gives a resultant arm which, when measured from the reference point about which the moments of the individual weights were calculated, locates the CG of the loaded aircraft. As long as the position of the CG is within the authorized limits, the aircraft is safe to fly. However, as all aircraft have an ideal CG position, i.e. one which allows the aircraft to give its best flight performance, every endeavour should be made to distribute the payload so that this position is obtained.

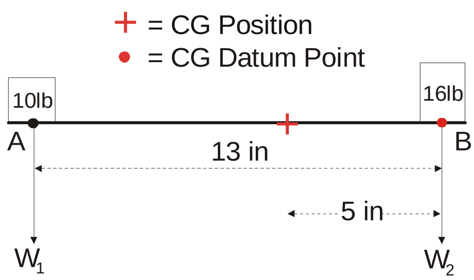
17. Fig 2 shows two weights, W_1 and W_2, of 10 lb and 16 lb respectively, located at points A and B, the ends of a thin rod whose weight is assumed to be negligible. The centres of the weights are 13 inches (in) apart. Since gravity is acting downwards through the centre of each weight, the resultant of the two weights, i.e. the combined weight of 26 lb, will act downward at some point between the two weights, this being the centre of gravity of the whole assembly.

2-22 Fig 2 Simple CG Calculation - Datum Point at End of Rod

a Datum at A



b Datum at B



18. The distance of this point from a stated reference point, ie a CG datum point, can be calculated by dividing the algebraic sum of the moments of the weight about the reference point by the sum of the weights. For example, Fig 2a assumes that the CG datum point is at A and, following the accepted conventions regarding positive and negative load arms and moments, the algebraic sum of the moments of the weights is:

Item	Weight (lb)	Load Arm (in)	Moment (lb in)
Weight W_1	10	0	0
Weight W_2	16	+13	+208
Totals	26		+208

The CG is therefore $\frac{+208}{26}$ = +8 in from A.

The positive sign indicates that it is to the right of A, so that the correct definition of the CG position is that it is 8 in to the right of A.

19. If the CG datum point is now assumed to be located at B (Fig 2b):

Item	Weight (lb)	Load Arm (in)	Moment (lb in)
Weight W_1	10	–13	–130
Weight W_2	16	0	0
Totals	26		–130

Now the CG position is $\frac{-130}{26}$ = –5 in from B.

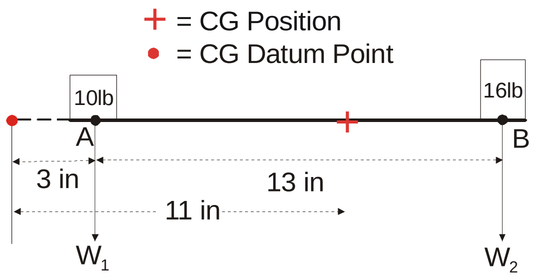
The negative sign indicates that it is to the left of B so that the correct definition of the CG position is 5 in to the left of B. Note that this is exactly the same position as before but expressed in relation to a different CG datum point.

20. Consideration will now be given to the use of a datum point other than at points A or B. Fig 3 shows the same weights and rod as before but with a CG datum point located away from the rod, 3 in to the left of A. Taking moments as before:

Item	Weight (lb)	Load Arm (in)	Moment (lb in)
Weight W_1	10	+3	+30
Weight W_2	16	+16	+256
Totals	26		+286

Now the CG position is $\frac{+286}{26}$ = +11 in, i.e. 11 in to the right of the datum point. This position is exactly the same as in the two previous examples.

2-22 Fig 3 Simple CG Calculation - Datum Point Away from Rod

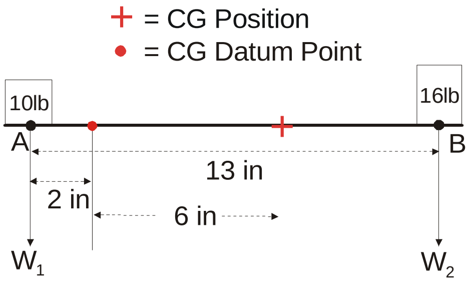


21. Finally, consideration will be given to the use of a datum point located on the rod and 2 in to the right of A (see Fig 4). The weights and the rod are exactly the same as before, but there are now both positive and negative load arms and moments. It is important to note how the algebraic sum of the moments is obtained:

Item	Weight (lb)	Load Arm (in)	Moment (lb in)
Weight W_1	10	–2	–20
Weight W_2	16	+11	+176
Totals	26		+156

The CG position is now $\frac{+156}{26}$ = +6 in ie 6 in to the right of the CG datum point. As the datum point itself is 2 in to the right of A, the CG position is 8 in to the right of A, ie the same position as before.

2-22 Fig 4 Simple CG Calculation - Datum Point on Rod



22. Summarizing, in the examples in paras 17 – 20, the data given consisted of the two weights and their corresponding load arms. From this, the algebraic sum of the longitudinal moments is calculated. The sum of the moments (remembering that sign is important) is then divided by the sum of the weights to give the longitudinal distance and the fore-and-aft direction of the new CG position from the CG datum point.

23. Exactly the same principles are applied to the determination of the basic longitudinal moment and basic CG position of an aircraft when the aircraft is weighed by the multi-point weighing method. The aircraft, in a condition as near to its basic condition as possible, is supported so that its longitudinal datum and lateral datum are horizontal on three or more weighing units arranged so that their individual readings can be resolved into two weights corresponding to the two weights in the previous examples. The horizontal distances from these two positions to the CG datum point are then determined either from data supplied by the manufacturer or by actual measurement. After an aircraft has been weighed, its basic weight and CG moment at that weight are recorded within the aircraft's Form 700.

AIRCRAFT LOADING

Trim Sheets

24. Trim sheets are supplied for use with a particular type of aircraft. Normally, with new types of aircraft, they are supplied initially by the manufacturers and bear no RAF identification other than the designation of the aircraft. After some experience has been gained in their use they may be modified, given an RAF Form number and made available through the usual channels. Trim sheets are used in conjunction with the Weight and Balance Data Book for transport aircraft.

25. Some arithmetical work is still required for the compilation of a trim sheet but this has been reduced to the minimum by the simplification of the necessary entries as follows:

a. Load index figures (index values) are used; these are the moments of the various items of load divided by a constant, e.g. 1,000. The basic moment of the aircraft is also divided by the same constant and is known as the basic index of the aircraft.

b. The weight, load arm and moment of each item of standard equipment which may be installed for the various roles have been calculated and recorded as index value changes in a Weight and Balance Data Book for the aircraft; some, or all, of this information may be given on the back of the trim sheet.

c. The cargo/passenger accommodation of the aircraft is divided into load compartments. The total weight in each compartment is recorded and dealt with as a separate entity.

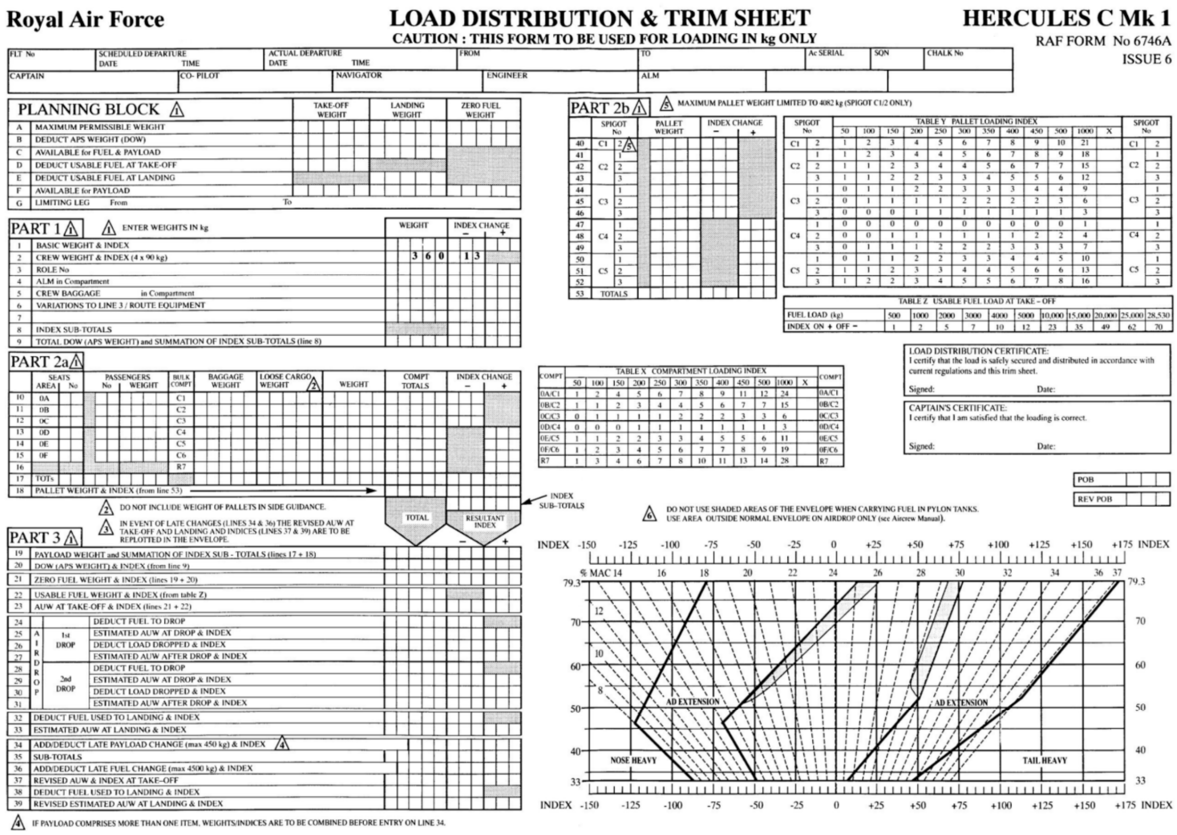
d. Load adjustment tables are provided on the trim sheet to enable the effect of adding or removing specific items of load, or stated increments of weight from various load compartments, to be read off as index changes, thus avoiding the need to calculate the moments involved.

These tables are of particular value if it is necessary to change the weight or position of any item of load after the aircraft has been loaded and the trim sheet completed.

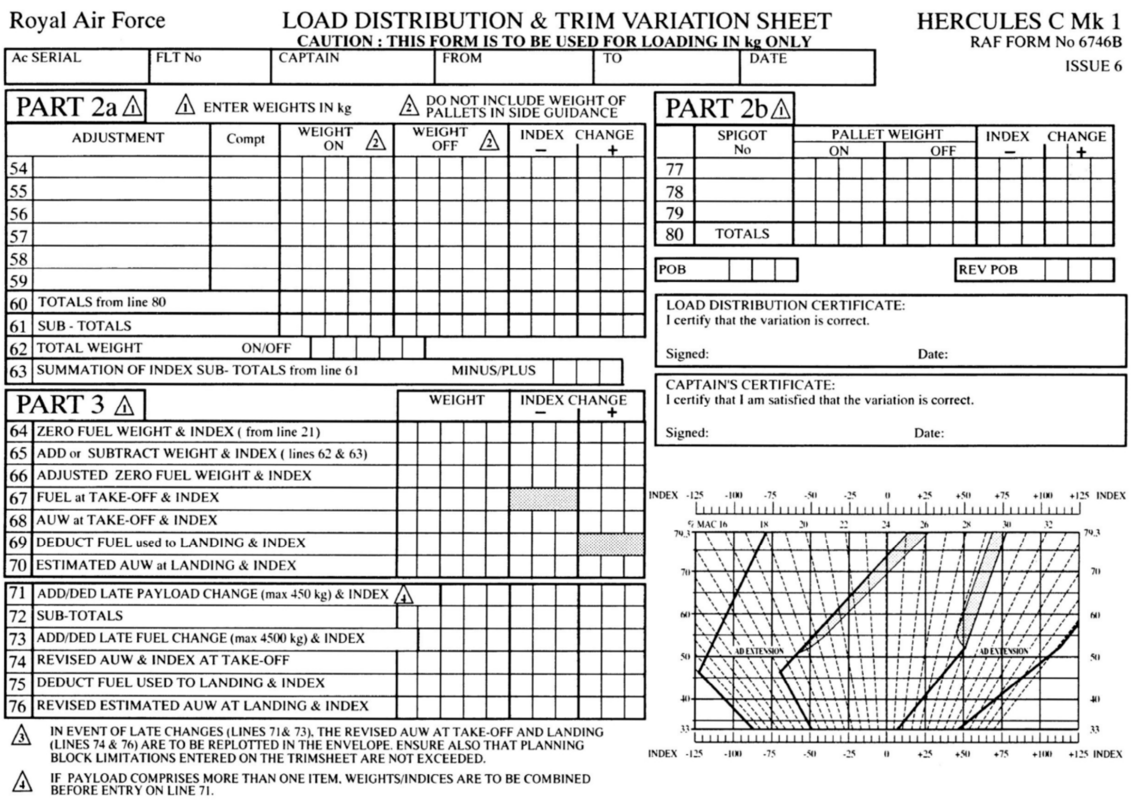
26. **Layout.** All trim sheets conform to a general pattern of layout although the actual detail may vary between aircraft. Trim sheets for a Hercules C Mk 1 aircraft (Forms 6746A & B) are shown at Figs 5 and 6 and these can be taken as typical examples.

Note: On the Hercules C Mk 1, all weights are in kilograms (kg). Some aircraft trim sheets require weights in pounds. The operator must take appropriate care.

2-22 Fig 5 Load Distribution and Trim Sheet



2-22 Fig 6 Load Distribution and Trim Variation Sheet



27. The top of the trim sheet (Fig 5) contains basic information such as the Flight Number, departure date, destination, etc. The planning block is completed prior to a particular task and is used as a guide for Air Movements to allocate payload to that flight.

28. The Part 1 (serial 1-9) of the trim sheet consists of the weight of the aircraft, crew, role equipment and other items which remain constant throughout the flight. This weight is termed the Aircraft Prepared for Service (APS) weight (also known as Dry Operating Weight (DOW)).

29. Part 2a (serial 10-18) and part 2b (serial 40-53), in conjunction with their relevant index tables (table X and Y), reflect the change in the position of the CG (expressed as an index value) as items of payload are loaded in various compartments of the aircraft.

30. Part 3 (serial 19-39) combines the APS weight with the payload weight to produce a Zero Fuel Weight (ZFW). Added to this ZFW is the usable fuel for the particular flight (the index values are shown in table Z) to produce an AUW for take-off. The estimated AUW at landing is determined by deducting the amount of fuel used during the flight.

31. The positions of the CG at take-off and landing are then plotted on the graph opposite the Part 3. This graph shows the permitted CG envelope of the Hercules C Mk 1. Both positions must lie within the permitted envelope.

32. The reverse of the trim sheet and the cover of the pad contain information on completion and use of the trim sheet.

33. The Form 6746A (Fig 5) is used for initial calculations. If there are late changes to the load and its distribution (e.g., extra passengers added at the final moments prior to take-off), then a Load Distribution and Trim Variation Sheet (Form 6746B) is used to calculate CG index changes (see Fig 6).

Cargo Restraint

34. All cargo carried in an aircraft must be secured against forces that may tend to move it from its allotted position in the aircraft. These forces may be caused by the acceleration and deceleration of the aircraft when taking-off or landing, by air turbulence, by control surface movements, or by the inertia of the cargo item should the aircraft crash-land or ditch. Under certain conditions these forces are of greater magnitude in one direction than others; the extreme instances occurring if the aircraft is suddenly slowed by landing on soft ground or by a crash-landing.

35. If an item of cargo is not properly secured it may move, with one or more of the following results:

a. Injury to personnel in the aircraft during a crash-landing, particularly those forward of the cargo, e.g. the flight-deck crew.

b. Movement of the aircraft CG position outside the permissible limits.

c. Structural damage to the aircraft.

d. Blockage of emergency exits.

e. Damage to other cargo items.

36. Small items of miscellaneous cargo may be safely secured by a net draped over them and secured to the aircraft structure, but bulky and heavy items such as large crates, vehicles and guns, etc must be more positively secured. This is done by the use of special equipment known collectively as 'tie-down equipment'. All transport aircraft are provided with tie-down points, which are part of the aircraft structure to which items of cargo can be secured.