Module - 4

COMPONENT LEVEL TESTING

Syllabus:
Compressor: Compressor MAP. Surge margin, Inlet distortions. Testing and Performance Evaluation. **Combustor:** Combustor MAP, Pressure loss, combustion light up test. Testing and Performance Evaluation. **Turbines:** Turbine MAP. Turbine Testing and Performance Evaluation. **Inlet duct & nozzles:** Ram pressure recovery of inlet duct. Propelling nozzles, after burner, maximum mass flow conditions. Testing and Performance Evaluation.

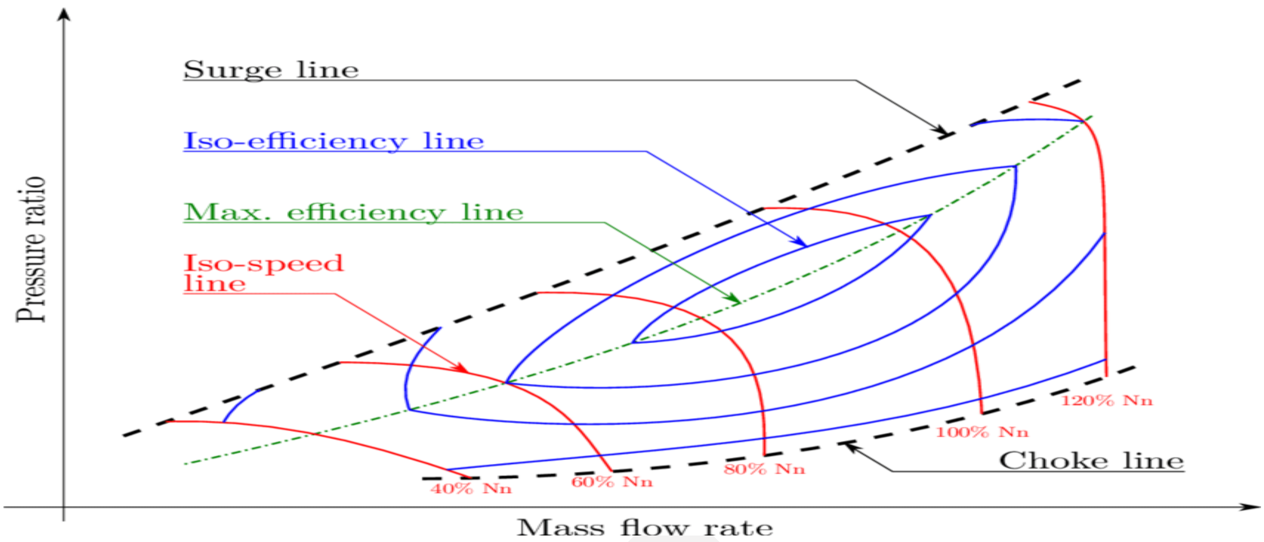
4.1 Axial flow compressors – off design performance

4.1.1 The compressor MAP

Once the compressor geometry has been fixed at the design point then the compressor map may be generated to define its performance under all off design conditions. The form of a map, sometimes called the characteristic or chic. Pressure ratio and isentropic efficiency are plotted versus referred flow for a series of lines of constant referred speed. For each referred speed line there is a maximum flow which cannot be exceeded, no matter how much pressure ratio is reduced. This operating regime is termed choke.

Ignoring second-order phenomena such as Reynolds number effects, for a fixed inlet flow angle and no rotating/tertiary stall or inlet distortion the following apply:

- For a fixed compressor geometry the map is unique.
- The operating point on the compressor map is primarily dictated by the components surrounding it as opposed to the compressor itself.
- Each operating point on the map has a unique velocity triangle (with velocity expressed as Mach number).
- Pressure ratio, CP.dT/T and efficiency are related and any two out of the three parameters may be used as the ordinates for the map. In fact any combination referred or full dimensionless groups will be suitable if they define flow, pressure ratio and temperature rise.



When Reynolds number falls below the critical value viscous flow effects have a second-order effect leading to lower flow, pressure ratio and efficiency at a speed. Low values may occur due to ambient conditions or due to linearly scaling a compressor to a smaller size. Reynolds number is in fact a fourth dimension to the map as shown in Fig. 5.6. As inlet temperature changes then the compressor geometry, and hence its map, may be modified due to thermal expansion and changing air properties. Differential radial growths between the discs/blades can cause tip clearance to change. Normally T1 effects are small and usually ignored. One important exception is HP compressor rig to engine differences, where the faster engine speed due to higher inlet temperature (than rig ambient) will change stress related growths.

4.1.2 Surge

At a given speed aerofoil rows may stall, that is to say the flow separates from the suction surface, as pressure ratio and hence incidence increase. For an aerofoil the point of stall is defined as the incidence at which the aerofoil loss coefficient reaches double its minimum value. In a multi-stage compressor stalled operation can be acceptable. For instance at low speeds following start up the front stages may well be stalled during normal operation, but steady state operation is possible as the rear stages are unstalled and stabilise the flow against the pressure gradient. However if the stall becomes severe, or is entered suddenly, a number of unacceptable flow regimes can result.

Inlet distortion (pressure and temperature)

Inlet distortion, which is spatial variation of inlet pressure or temperature, can significantly affect the overall compressor map. The most important effect is a reduction in the surge line. The method of parallel compressors is employed to evaluate this. Here the exit pressure and temperature are considered to be constant circumferentially. The map is then applied to two parallel streams as described below.

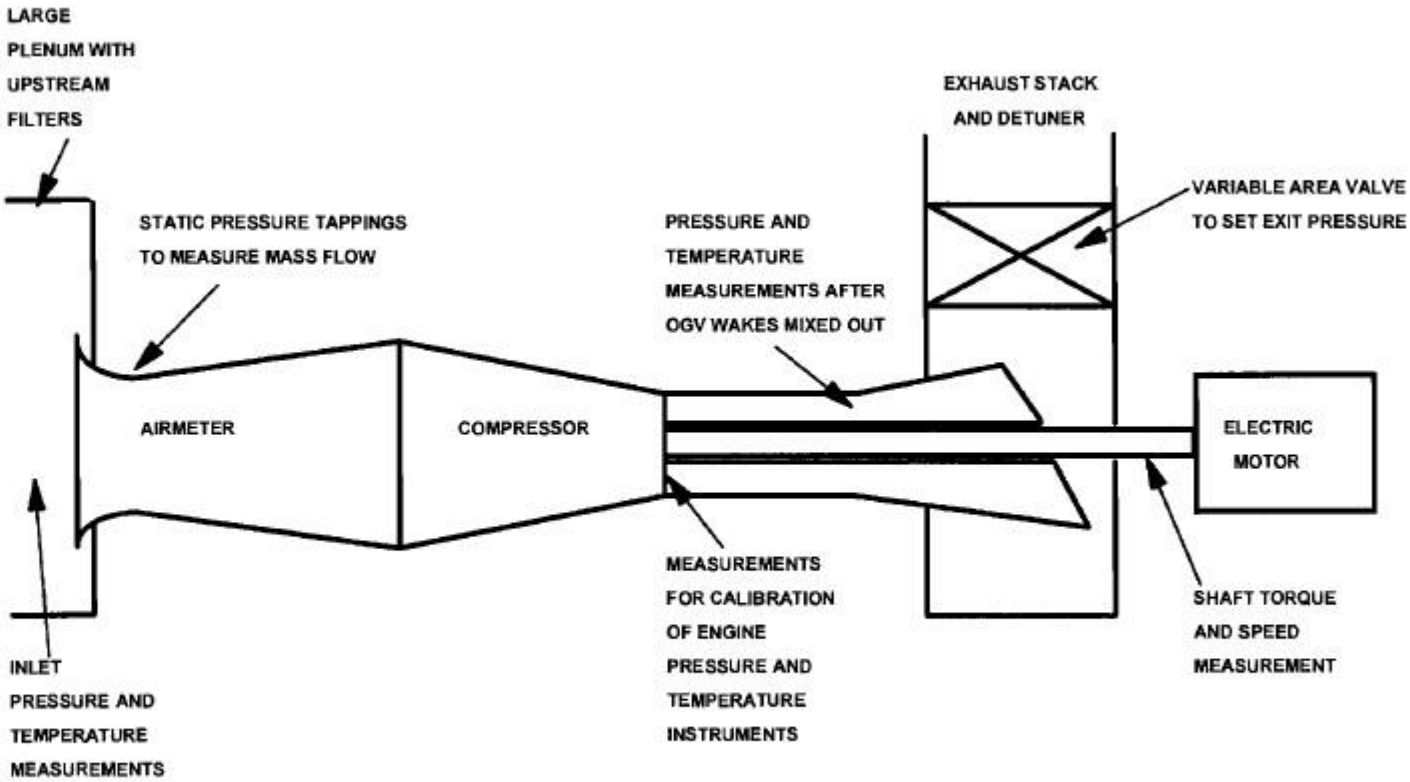
For aircraft engines in cross winds or at high angles of attack the inlet flow may be distorted circumferentially, leading to sectors where inlet pressure is significantly lower than the average.

Inlet temperature distortion may occur due to a number of reasons such as poor test bed design, or ingestion of thrust reverser exhaust or another engine’s exhaust. Again the method of parallel compressors may be used to determine additional surge margin required. In this instance it is the inlet capacity in the 1208 sector with the lowest temperature which is used for one stream, and the mean temperature in the remaining sector used for the second sector. This gives rise to a TC120 coefficient.

4.1.3 The compressor rig test

When a new compressor has been designed it may be tested on a rig prior to being built into an engine. This allows the compressor geometry to be optimized in a controlled environment, often before the rest of the engine hardware is available. There are so many design parameters involved with an axial flow compressor that unless the design is well within previous experience a rig test is essential.

The typical rig configuration is shown in Fig. 5.16. The compressor is driven by an electric motor which is controlled to a specified speed. Measurements are taken allowing flow, pressure ratio and efficiency to be calculated. The exit valve is then closed with the compressor speed maintained, forcing the pressure ratio to be increased and the flow decreased. This process is repeated until the surge line is encountered. A similar procedure is then followed for a number of speed lines. For each throttle setting varying the speed will produce a unique working line, akin to compressor operation within an engine.



Note:
Measurement of both shaft power input and temperature rise produces 'shaft' and 'gas path' efficiency levels respectively. Shaft efficiency will include disc windage. Gas path may not, if heated air exhausts separately.

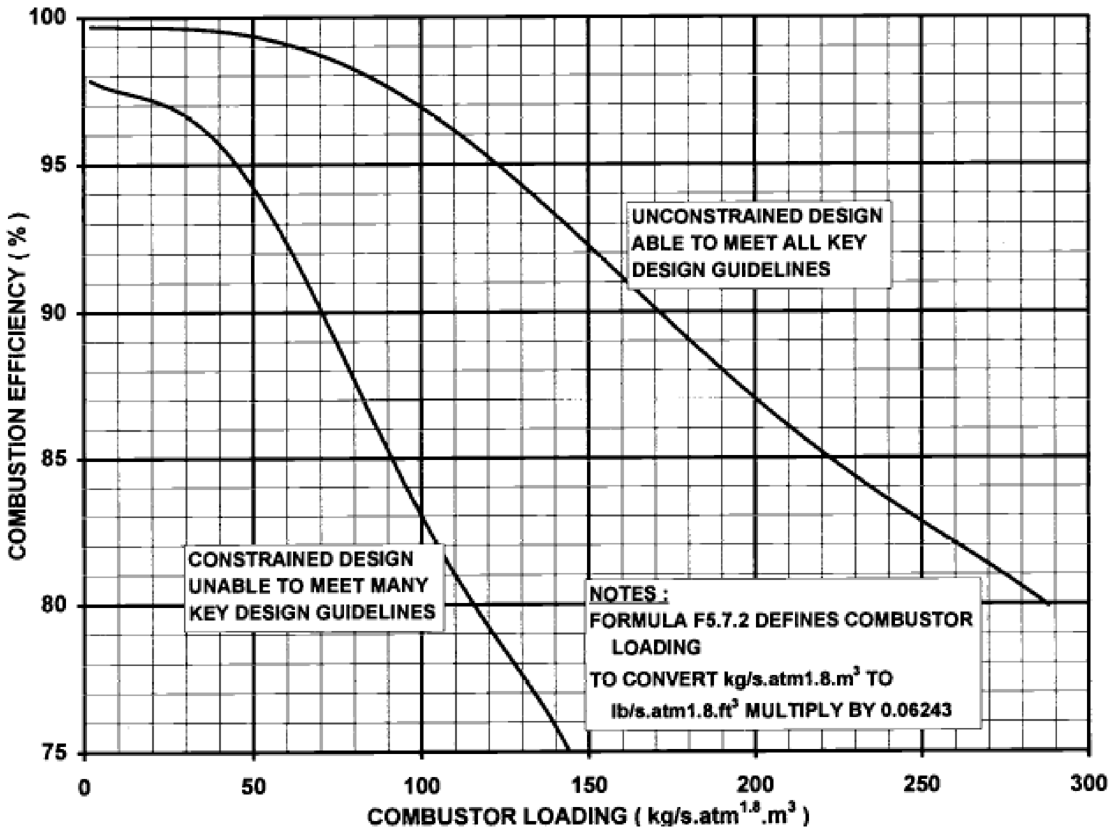
Fig. 5.16 Compressor rig layout.

4.2 Combustors – off design performance

4.2.1 Efficiency and temperature rise

Chart 5.5 may be used to determine efficiency for engine off design performance models, with a chosen curve digitized so that the model can interpolate along it using loading evaluated from the known inlet conditions and combustor volume. Formula F5.7.8 presents a polynomial fit for the unconstrained design which is able to meet all the key design guidelines. In fact, fuel air ratio is a third dimension to Chart 5.5 but its effect issmall and depends on the combustor design; no generic chart can be prepared and it may be ignored for early models.

Chart 5.5 Combustion efficiency versus loading.



4.2.2 Pressure loss

Cold and hot pressure loss may be derived from Formulae F5.7.9 and F5.7.10. The constants may be derived at the design point where percentage pressure loss as well as inlet and outlet parameters are known.

F5.7.9 Combustor cold loss (kPa) = fn(cold loss factor, inlet air flow (kg/s), inlet temperature (K), inlet pressure (kPa))

DPcold = Kcold * P31 * (W31 * SQRT(T31)/P31)^2

F5.7.10 Combustor hot loss (kPa) = fn(hot loss factor, inlet air flow (kg/s), inlet temperature (K), exit temperature (K), inlet pressure (kPa))

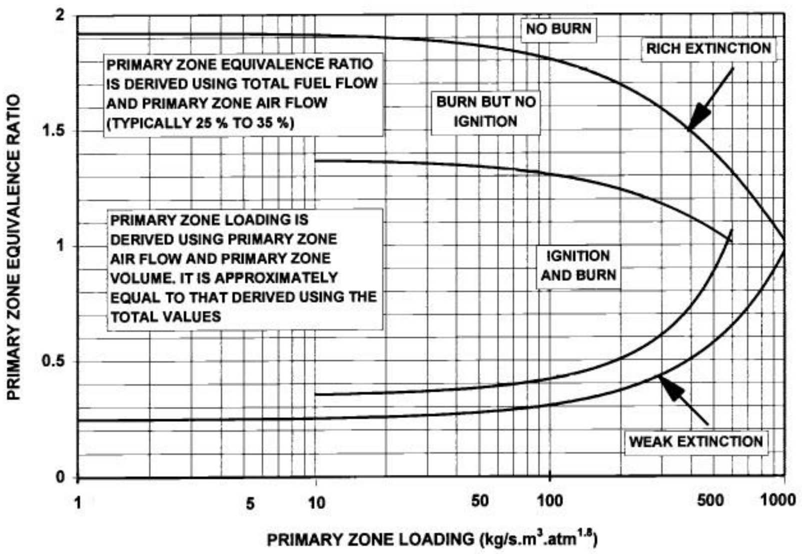
DPhot = Khot * P31 * (T4/T31 − 1) * (W31 * SQRT(T31)/P31)^2

4.2.3 Combustor stability

If fuel is injected correctly into a well-designed combustor then stability is primarily a function of velocity, absolute pressure and temperature. A low velocity aids flame stability, while high inlet pressure and temperature promote combustion by creating a closer density of air and fuel molecules or higher molecular activity. These three variables are all included in the loading parameter (velocity indirectly). For stability correlations loading is calculated using only the primary zone air flow and volume, as this is where combustion begins. In fact this will not be significantly different from that calculated using the total can volume and mass flow. Equivalence ratio is derived using the total fuel flow and primary zone air flow.

Chart 5.7 shows a generic combustor stability loop of primary zone equivalence ratio versus loading. There is a loading value of around 1000 kg/s atm1.8m3 beyond which combustion is not practical, this is primarily driven by velocity. As loading is reduced the flammable equivalence ratio band increases. Rich and weak extinction fuel air ratios may also be plotted versus primary zone exit velocity, as opposed to loading, so there are then families of curves for absolute pressure and temperature.

Chart 5.7 Combustion stability versus primary zone loading and equivalence ratio.



(a) Burn and ignition stability loops

The fraction of combustor entry air entering the primary zone is constant for off design operation, hence primary zone fuel air ratio may be derived from knowing total combustor inlet mass flow and fuel flow. Rich extinction is rarely encountered in an engine as over temperature of other components would normally precede it. However weak extinction is a threat, and since the exact curve is highly dependent upon the individual combustor design it must be determined by rig test. The levels in Chart 5.7 are a reasonable first indication, however.

A further instability called rumble can occur at weak mixtures. It is characterized by a 300–700 Hz noise generated by the combustion process.

4.2.4 Weak extinction versus ambient conditions and flight Mach number

As shown on Chart 5.7, for industrial, marine and automotive engines loading only increases marginally as the engine is throttled back to idle. Primary zone equivalence ratio typically falls from 1 to around 0.4 at idle, and additionally around 30 to 50% under fuelling relative to steady state occurs during a decel. Chart 5.7 shows that weak extinction is around 0.25 equivalence ratio, hence even at idle the permissible under fuelling would be around 40%. The decel schedule is set to prevent weak extinction, which is then not usually a threat given a well-designed system and anyway such a broad permissible band.

For aircraft engines high altitudes provide a more severe off design condition for weak extinction. The typical variation in loading and fuel to air ratio for a turbofan at key operating conditions are also illustrated on Chart 5.7. The worst case is usually a decel to just above idle at the highest altitude and lowest Mach number, however depending upon the idle scheduling this worst case can occur at an intermediate altitude. In contrast to an industrial engine, loading does increase significantly and hence great care must be taken to ensure that the stability loop is satisfactory throughout the operational envelope.

4.2.5 Starting and restarting – ignition, light around and relight

After dry cranking, fuel must be metered to the combustor and then ignited. Usually igniters are located in two positions, once ignition has been achieved the rest of the burners, or cans, must light around. A typical ignition loop is shown on Chart 5.7, again for an individual combustor it must be determined by rig test. Light off occurs with primary zone equivalence ratios in the range 0.35–0.75, depending partly on the loading, and immediate combustion efficiency is around 60–80%.

For aircraft engines the capability to relight within the restart envelope is essential. This is a particular challenge at high altitude and low flight Mach number where loading is high due to the low inlet pressure and temperature. The designing the combustor to have a loading of less than 300 kg/s atm1.8m^3 when windmilling at this flight condition is essential. Also it is vital to measure altitude relight performance in the rig test programme at the earliest opportunity.

4.2.6 The combustor rig test

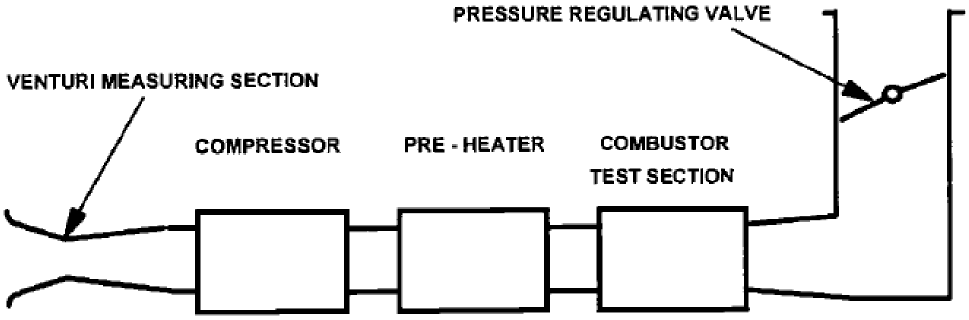


Fig. 5.26 Combustor rig test facility.

Figure 5.26 shows a typical combustor rig. Air enters through a venturi measuring. It is compressed, and if necessary heated to provide the inlet pressure and temperature per the engine condition being tested. It then passes into the combustor test section where the fuel is burned, and leaves via a diffuser and throttle valve. For cannular systems a single can may be tested reducing the size of the rig facility. For a new design of combustor a rig test is mandatory prior to any development engine testing, and as a minimum must establish and develop:

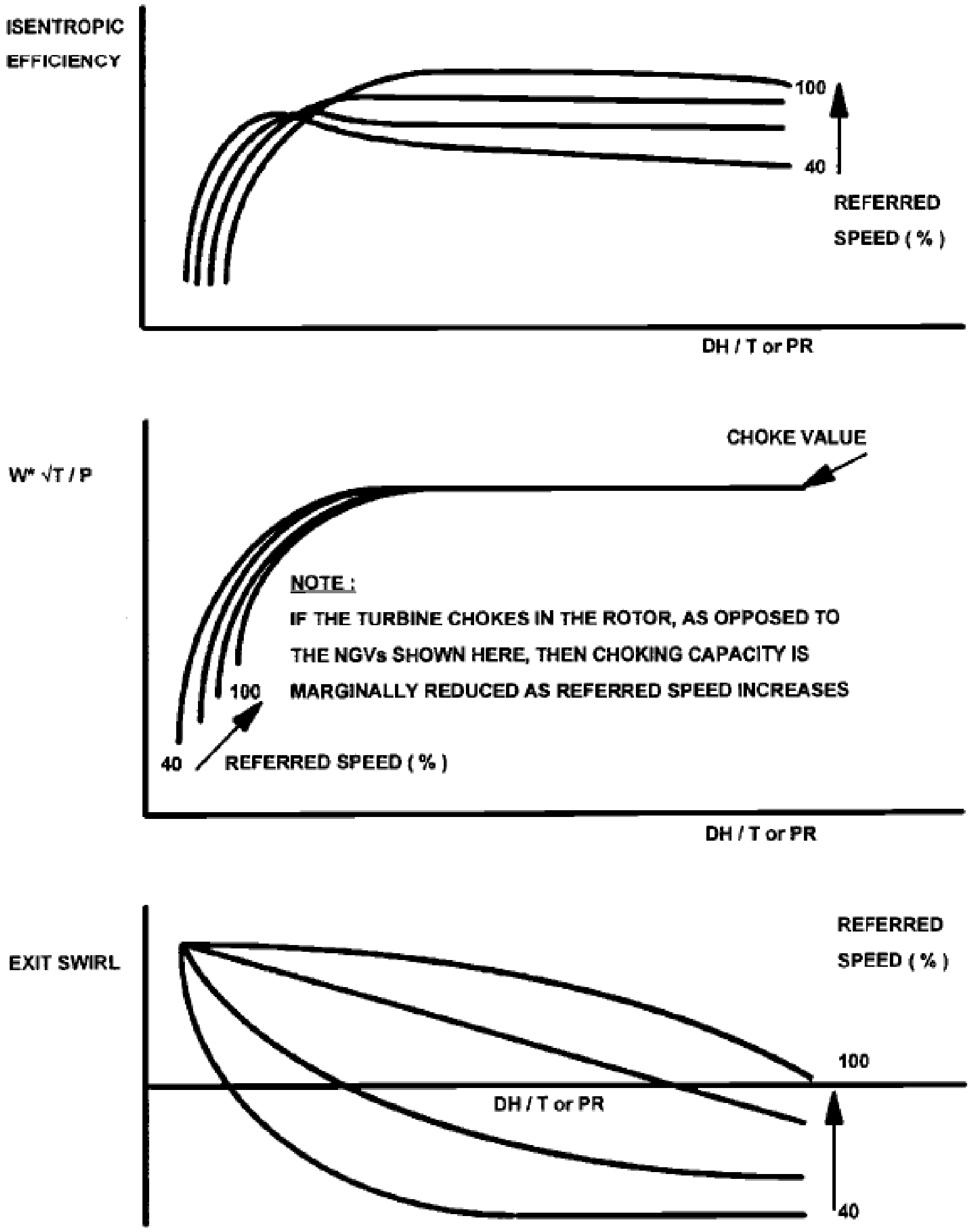
- Combustion efficiency versus loading and fuel air ratio
- Combustor cold loss pressure coefficient by flowing the rig without fuel being metered
- Combustor rich and weak extinction boundaries
- Combustor ignition boundaries
- Combustor wall temperatures using thermal paint and/or thermocouples
- OTDF and RTDF using traversing thermocouple rakes or thermal paint
- Emissions levels using a cruciform probe with a good coverage of sampling points at the exhaust

If the rig cannot achieve full engine pressure then it must be set up to the same inlet W $\sqrt{T}/P$ as the engine condition under consideration. However since the absolute pressure, and hence loading are different then care must be taken in interpreting results. Quartz viewing windows are of tremendous value. Also cold tests using water and air in perspex models of the combustor are an invaluable tool in deriving satisfactory aerodynamics.

4.3 Axial flow turbines – off design performance

4.3.1 The turbine map

Once the turbine geometry has been fixed at the design point then the turbine map may be generated to define its performance under all off design conditions. The most common form of map, sometimes called the characteristic or chic, is presented in Fig. 5.30.



5.30 The turbine map.

Capacity (referred flow), efficiency and exit swirl angle are plotted for lines of constant referred speed versus the work parameter (dH/T or CP.dT/T). For each referred speed line there is a maximum flow capacity which cannot be exceeded no matter how much CP.dT/T is increased. This operating regime is termed choke. For the map shown in Fig. 5.30 the choking capacity is the same for all referred speed lines. This is usually the case when choking occurs in the NGV, should it occur in the rotor blades then these lines separate out with choking capacity reducing marginally as referred speed is increased due to decreased density in the rotor throat. Limiting output or limit load is the point on the characteristic beyond which no additional power results from an increased expansion ratio. Here the shock wave moves from the rotor throat to its trailing edge, hence its aerodynamics are not affected by downstream pressure.

Ignoring second-order phenomena such as Reynolds number effects, and for a fixed inlet flow angle the following applies.

- For a fixed turbine geometry the map is unique.
- The operating point on the turbine map is dictated by the components surrounding it as opposed to the turbine itself.
- Each operating point on the map has a unique velocity triangle, expressed as Mach number.
- Expansion ratio, CPdT/T and efficiency are related, hence in fact any two of the three parameters may be used as the ordinates for the map.

Impact on the map of linearly scaling a turbine design

If scaling ‘down’ results in a small turbine then it may not be possible to scale all dimensions exactly, such as tip clearance or trailing edge thickness leading to a further loss in capacity pressure ratio and efficiency at a speed.

Reynolds number and inlet temperature effects

Reynolds number is strictly also a fourth dimension for a turbine map. Capacity and efficiency are both marginally reduced at a referred speed and CPdT/T. However due to the high pressures and temperatures in a turbine, Reynolds number rarely falls below the critical value to then have an effect As for a compressor, changes in turbine geometry due to changes in absolute temperature have only a tertiary effect and are usually ignored.

Change in the working fluid

When the map is plotted in terms of dimensionless parameters, then to a first order, and for a fixed inlet flow angle, it is unique for all linear scales and working fluids. The turbine map will normally be generated in terms of referred parameters as per Fig. 5.30 using gas properties for dry air. In reality these properties will be modified by the presence of combustion products, and possibly by humidity or water or steam injection. Most engine off design performance models use a map for dry air and deal with any change in gas properties.

4.3.2 The turbine rig test

Turbine rig tests, prior to engine testing, are only carried out for the highest technology engines. This is because of the cost and complexity of the rig to deliver representative inlet conditions, which requires a large heater and compressor with independent control. The turbine output power is absorbed by a water brake or dynamometer, hence referred speed may be held constant and an outlet throttle valve varied to map the speed line.

4.4 Ducts (exhaust nozzles) – off design performance

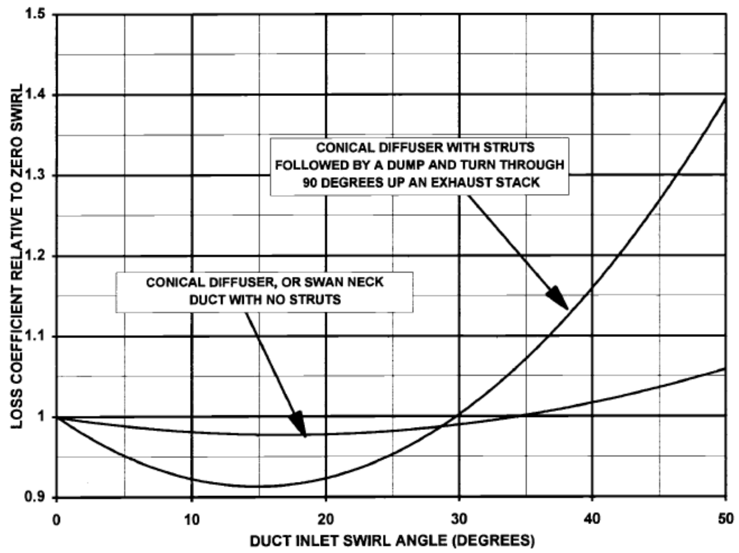
4.4.1 Loss coefficient (lambda)

Once the duct geometry has been fixed by the design process then the characteristic of lambda versus inlet swirl angle is fixed. The only exception to this rule is if dramatic flow separation occurs such that the effective geometry is significantly modified.

Inlet swirl is usually constant throughout the operational envelope for ducts downstream of compressors or fans. This is because, in general, the last component is a stator which will have a constant exit flow angle, unless it is operated so severely off design that it stalls. Hence it is usually only after turbines where there is any significant variation in swirl angle at off design conditions. In general exit swirl angle only changes dramatically at off design conditions for the last turbine in a turboshaft engine, where exhausting to ambient produces larger changes in expansion ratio. Exit swirl angle changes may be even larger in power generation as the power turbine must operate synchronously, hence changes of up to 308 between base load and synchronous idle are typical. It is essential to account for this in performance modelling, as well as in the aerodynamic and mechanical design of the duct. The latter is of particular concern for high cycle fatigue if vanes are present which may be aerodynamically excited.

Chart 5.15 shows the typical variation in lambda with inlet swirl angle for duct types which commonly occur downstream of turbines. The optimum swirl angle is of the order of 158. Also lambda rises rapidly for higher swirl angles for the hot end drive configuration of industrial engine exhaust. An improvement is to model the strut loss separately, as a 'bucket' of lambda versus inlet swirl angle. This will be non-symmetrical if the strut leading edge angle is not zero, as incidence and turning losses will not be minimized simultaneously.

Chart 5.15 Effect of duct inlet swirl on loss coefficient.



4.4.2 Pressure loss – all ducts except aero-engine intakes

As for the design point, pressure loss at off design may be found from Formula F5.13.4 with the loss coefficient being determined. This requires the duct area also to be input into the engine off design performance model such that with the known flow conditions W√T/AP may be calculated. Total to static pressure ratio may then be found via Q curve so that percentage pressure loss is calculated via Formula F5.13.6. Solving for total to static pressure ratio involves iteration and hence is cumbersome.

F5.13.4 Duct total pressure loss (kPa) = fn(lambda, inlet dynamic head (kPa))

$$\mathrm{Pin} - \mathrm{Pout} = \mathrm{LAMBDA} * (\mathrm{Pin} - \mathrm{PSin})$$

F5.13.6 Duct total pressure loss (%) = fn(lambda, inlet total and static pressures (kPa))

$$(\mathrm{Pin} - \mathrm{Pout})/\mathrm{Pin} = \mathrm{LAMBDA} * (1 - (1/(\mathrm{Pin}/\mathrm{PSin}))) * 100$$

For a given geometry it can be shown that $(W\sqrt{T}/P)^2$ is approximately proportional to inlet dynamic head divided by inlet pressure (Formula 5.14.1). To reduce computation in off design engine performance models it is common practice to use formula F5.14.2 as opposed to F5.13.6 to compute duct pressure loss. The pseudo loss coefficient, or alpha, is directly proportional to lambda and all the rules described earlier apply equally to it. For a given duct geometry alpha is calculated from lambda at the design point via Formula F5.14.3. Hence in engine off design performance models, total pressure loss may be easily calculated from Formula 5.14.2 once inlet conditions are known, without recourse to the iteration described above. However, often Mach number values are required for information, and such simplification is not possible. Mach number must then be calculated iteratively from the duct inlet conditions and area.

F5.14.1 Duct flow capacity squared is approximately proportional to dynamic head

(Win * SQRT(Tin)/Pin)^2 ∝ (Pin − PSin)/Pin

F5.14.2 Duct total pressure loss (%) = fn(alpha, inlet capacity (kg $\sqrt{}$K/s kPa))

DP/P = ALPHA * (Win * SQRT(Tin)/Pin)^2

F5.14.3 Pseudo loss coefficient alpha = fn(lambda, total pressure (kPa), static pressure (kPa), mass flow (kg/s), total temperature (K)

ALPHA = (LAMBDA * (1 − (1/(Pin/PSin))) * 100)/(Win * SQRT(Tin)/Pin)^2

Generally duct inlet Mach numbers, and hence percentage pressure loss, reduce as an engine is throttled back. Exceptions occur when the downstream capacity does not fall, such as for bypass ducts and combustor entry ducts.

4.4.3 Ram recovery factor – aero-engine intakes

For subsonic intakes, ram recovery at off design conditions is calculated in the same fashion as for other ducts using either lambda or alpha. However for supersonic intakes there is additional loss of total pressure across the shock system. Formula F5.14.4 is a first pass working rule for the pressure ratio across the shock. The pressure loss in the downstream section must be derived and the two values multiplied together to give an overall exit pressure. If needed, the overall ram recovery factor can then be calculated from Formula F5.13.9. At a flight Mach number of 2, typically 8–10% of free stream total pressure will be lost in the intake system.

F5.14.4 Pressure ratio for supersonic intake shock system = fn(free stream total pressure (kPa), flight Mach number)

P1 = P0 * (1 − 0.075 * (M − 1)^1.35)

(i) This is purely for the shock system. Other duct pressure losses will occur within the intake.

F5.13.9 Ram recovery factor = fn(ram total pressure (kPa), ambient pressure (kPa), intake exit pressure (kPa))

RRF = (P1 − Pamb)/(P0 − Pamb)

4.4.4 Specific features of propelling nozzles

Propelling nozzle CD and CX at off design conditions may be derived from Charts 5.13 and 5.14. In engine off design performance models these may be loaded in tabular form and linear interpolation employed for a known value of propelling nozzle expansion ratio. Alternatively a polynomial fit may be utilized.

For variable area nozzles the control schedule must also be included in the engine off design performance model such that area can be derived for a given operating point.

Chart 5.13 Propelling nozzle discharge coefficients.

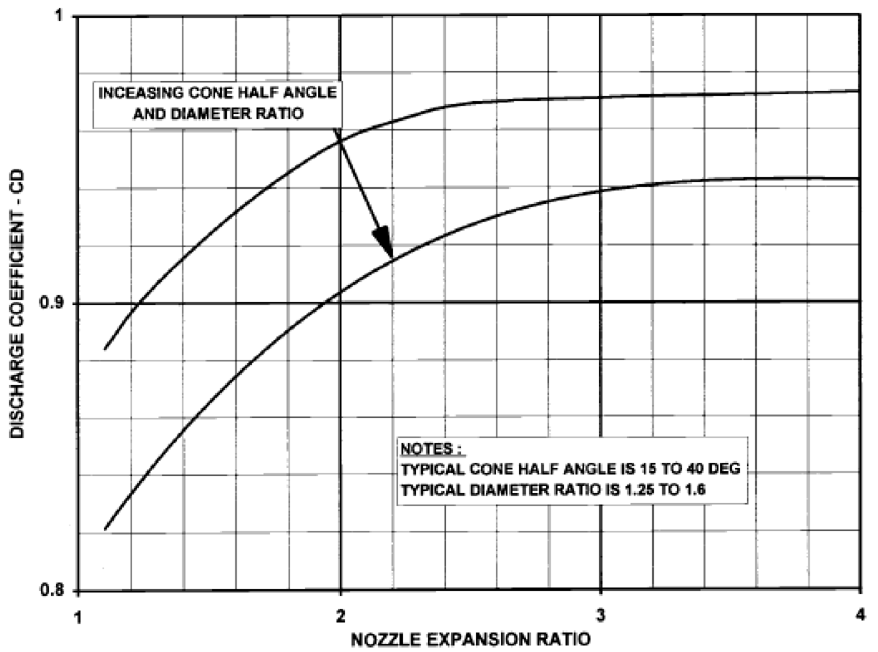
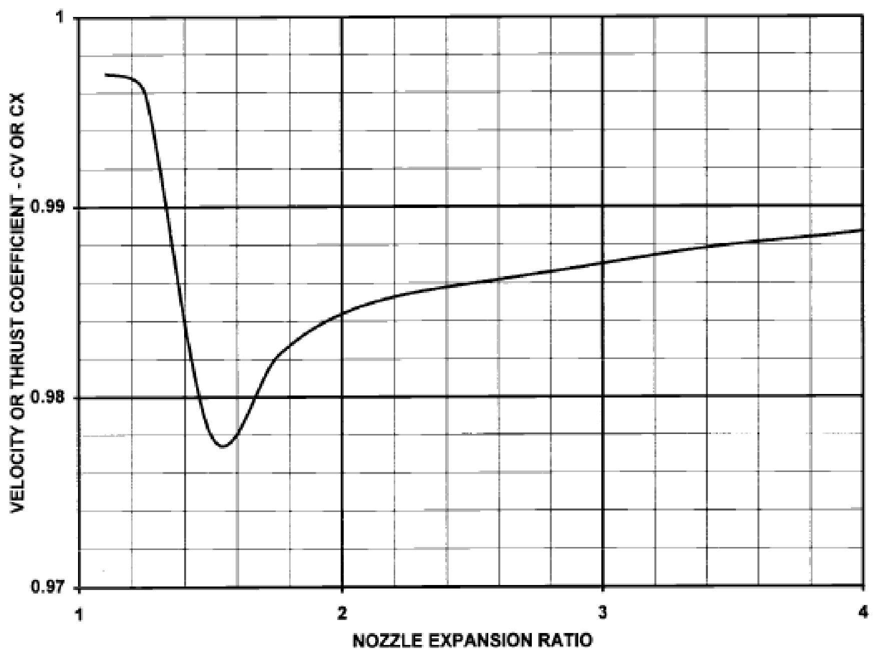


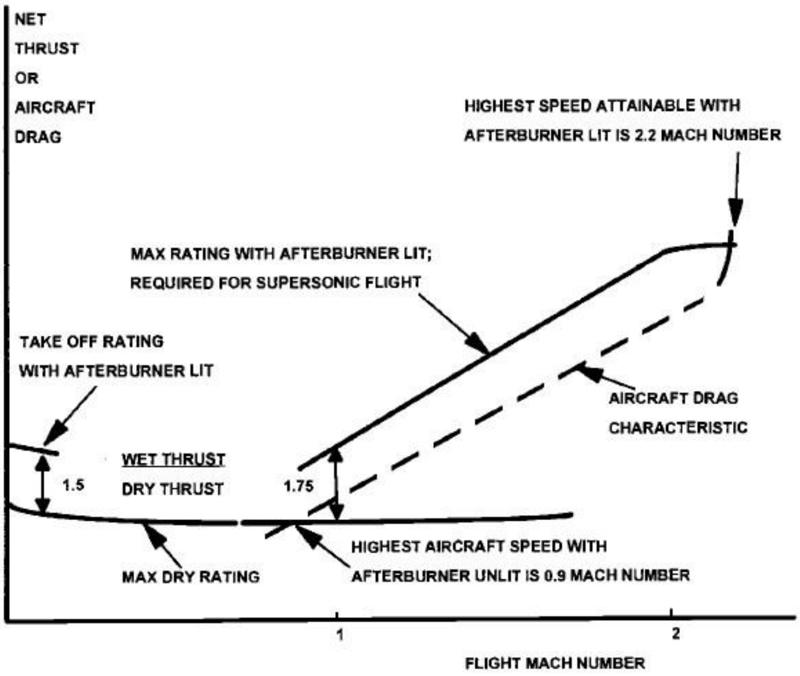
Chart 5.14 Propelling nozzle thrust coefficient.



4.5 Afterburners – off design performance

4.5.1 Operation

Figure 5.41 shows both engine net thrust and aircraft drag versus flight Mach number for typical military fighter operation. The afterburner is operative for takeoff, but to maintain good SFC it is not used or indeed required for subsonic flight, except in combat. However a flight Mach number of 0.9 is the highest attainable dry as here aircraft drag exceeds engine maximum dry thrust. Hence at this Mach number the afterburner is lit and the additional thrust enables fast aircraft acceleration through the sound barrier. The maximum flight Mach number is now 2.2 where aircraft drag again exceeds engine thrust.



Notes:
Thrust gains shown are indicative, actual values depend upon engine cycle.
With afterburner lit SOT is held at the same value as at Mach 0.9 unlit.
Limiting attainable aircraft Mach numbers shown are for this illustration only.

Fig. 5.41 Typical afterburner operation for a low bypass ratio military mixed turbofan.

The afterburner will usually have a maximum wet, and a minimum wet rating. These ratings relate to the degree of afterburning and for both the gas generator may be at different ratings. The following comments apply to the extreme combinations.

Minimum gas generator:

- Minimum wet: this is used on approach where in emergency the pilot may need to slam the throttle to go around. This gives approximately 25% of the maximum wet rating with the gas generator at full throttle.
- Maximum wet: this is not commonly used.

Maximum gas generator:

- Minimum wet: this is used in combat situations, or for low supersonic Mach number flight. This gives approximately 90% of the maximum wet rating.
- Maximum wet: this is used for takeoff and at the highest flight Mach numbers attainable.

4.5.2 Variable area propelling nozzle

To avoid compressor surge problems it is essential to have a variable area propelling nozzle downstream of an afterburner. This is because when the afterburner is lit the dramatic increase in propelling nozzle temperature would rematch the gas generator pushing compressors to surge. To maintain the same gas generator operating point the variable nozzle area must be increased with the square root of afterburner exit temperature. The most common control system strategies monitor referred parameter relationships dry, and then modulate propelling nozzle area to maintain these relationships when wet. Turbine expansion ratios and compressor pressure ratios are the parameters most commonly used as they are highly sensitive to changes in nozzle temperature enabling the control swiftly to change nozzle area.

At high nozzle pressure ratios and overlarge nozzle is of benefit in increasing thrust, via increased engine air flow; this is over restoring. At low nozzle pressure ratio a smaller nozzle area is better, to raise it, known as under restoring. Both routes are limited by compressor surge margins.

When operating dry the propelling nozzle area must be a few per cent larger than if the same engine were not fitted with an afterburner. This is to maintain the same gas generator operating point since the additional afterburner pressure loss and fuel flow increase the referred flow of the jet.

4.5.3 Temperature rise, efficiency, pressure losses and wall cooling

As at the design point, Formulae F3.37–F3.41 may be used to first-order accuracy at off design conditions to determine temperature rise once inlet temperature and fuel air ratio are known. Should the afterburner exit temperature be less than 1900K then dissociation is unlikely and these calculations are rigorous. Reference 44 shows how to calculate temperature rise rigorously with dissociation. The lowest temperature rise of around 200^0 C occurs at the minimum wet rating with the gas generator throttled back.

F3.37–F3.40 Fuel air ratio = fn(combustor inlet and exit temperatures (K))

For fully rigorous calculations:

F3.37 FAR = DH/(LHV * ETA34)

(i) DH must be calculated from Formulae F3.15, F3.26 and F3.27.

For calculations to within 0.25% accuracy with kerosene fuel which has an LHV of 43124 kJ/kg:

F3.38A FAR1 = 0.10118 + 2.00376E-05 * (700 – T3)
FAR2 = 3.7078E-03 – 5.2368E-06 * (700 – T3) – 5.2632E-06 * T4
FAR3 = 8.889E-08 * ABS(T4 – 950)
FAR = (FAR1 – SQRT(FAR1^2 + FAR2) – FAR3)/ETA34

For calculations to within 0.25% accuracy for diesel or kerosene fuel with an LHV other than 43124 kJ/kg:

F3.38B FAR = F3.37 * 43124/LHV

For calculations to within 1% accuracy with the sample natural gas, CPs at the mean temperature must be evaluated from Formulae F3.24 and F3.25, and:

F3.39 FAR = F3.36 * 43124 * Cpgas/(LHV * CPliquid)

For calculations to within 5% accuracy CP may be taken as that at the mean temperature:

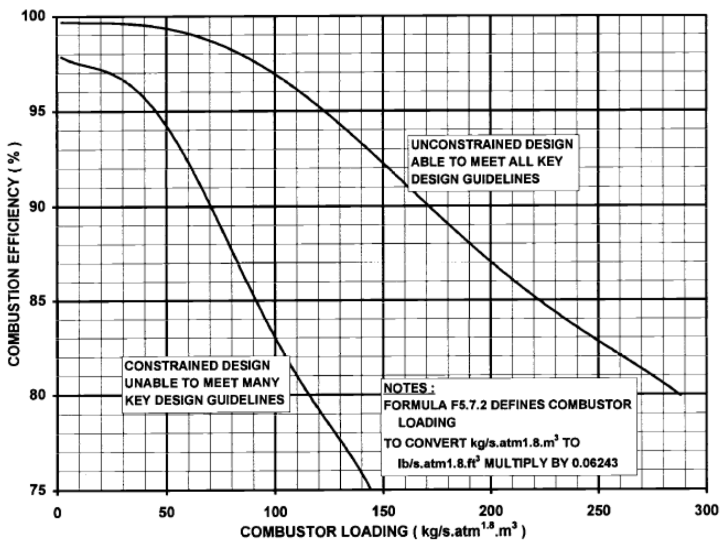
F3.40 FAR = CP * (T4 – T3)/(ETA34 * FHV)

F3.41 Combustor exit temperature = fn(inlet temperature (K), fuel air ratio) – iterative

```
T4 = 1000
START:
T4previous = T4
FARcalc = F3.37 to F3.39
IF ABS((FAR – FARcalc)/FAR) > 0.0005 THEN
T4 = (T4previous – T3) * FAR/FARcalc + T3
GOTO START:
END IF
```

Again chemical efficiency may be correlated at off design conditions versus afterburner loading. For first-order calculations, around 7% should be deducted from the levels given on Chart 5.5. If an engine programme is committed then this characteristic must be determined from rig testing. Efficiency may fall to as low as 30% at the minimum wet rating with the gas generator throttled back.

Chart 5.5 Combustion efficiency versus loading.



The cold and hot pressure loss coefficients, and hence pressure losses, are determined as for a conventional combustor using Formulae F5.7.9 and F5.7.10. The percentage of air used for afterburner cooling remains constant at off design

F5.7.9 Combustor cold loss (kPa) = fn(cold loss factor, inlet air flow (kg/s), inlet temperature (K), inlet pressure (kPa))

DPcold = Kcold * P31 * (W31 * SQRT(T31)/P31)^2

F5.7.10 Combustor hot loss (kPa) = fn(hot loss factor, inlet air flow (kg/s), inlet temperature (K), exit temperature (K), inlet pressure (kPa))

DPhot = Khot * P31 * (T4/T31 − 1) * (W31 * SQRT(T31)/P31)^2

conditions.

4.5.4 Stability

In practical operation an afterburner will never encounter rich extinction. However at rich mixtures approaching stoichiometric an audible instability called afterburner buzz may occur. Buzz is noise generated by the combustion process and is more prevalent at higher afterburner pressures and lower afterburner Mach numbers. If the afterburner is continuously operated with buzz present then mechanical damage is likely. It is one of the practical design phenomena that limit the achievable afterburner exit temperature. Weak extinction must be designed against for low afterburner fuel air ratios. The lower limit to the minimum wet rating with the gas generator throttled back is typical of the restrictions imposed by weak extinction.